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NSWCDD/TR-01/30

A SEMIEMPIRICAL METHOD FOR PREDICTING AERODYNAMICS OF TRAILING EDGE FLAPS

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JUNE 2001

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY (Leave blank)

2. REPORT DATE

June 2001

3. REPORT TYPE AND DATES COVERED

Final

4. TITLE AND SUBTITLE

A Semiempirical Method for Predicting Aerodynamics of Trailing Edge Flaps

5. FUNDING NUMBERS

6. AUTHOR(s)

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7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

Commander
Naval Surface Warfare Center
Dahlgren Division (Code G23)
17320 Dahlgren Road
Dahlgren, VA 22448-5100

8. PERFORMING ORGANIZATION REPORT NUMBER

NSWCDD/TR-01/30

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

11. SUPPLEMENTARY NOTES

12a. DISTRIBUTION/AVAILABILITY STATEMENT

Approved for public release; distribution is unlimited.

12b. DISTRIBUTION CODE

13. ABSTRACT (Maximum 200 words)

An Improved Semiempirical Method has been developed to estimate the static aerodynamics of configurations that use a trailing edge flap for control. The method is based on deflecting the full aft-located lifting surface an amount that allows the normal force coefficient to be equal to that generated by the deflected flap. A transfer in pitching moments and a modified axial force coefficient is derived to complete the set of static aerodynamics. The method is derived using theoretical methods that are a part of the 1998 version of the Naval Surface Warfare Center aeroprediction code and two sets of experimental data. Comparison of the improved method to available data shows the method to give satisfactory results over the practical range that trailing edge flaps are contemplated for use. However, additional wind tunnel data is needed to refine and expand the applicability of the method. This is particularly true for transonic Mach numbers and for supersonic Mach numbers where the angle of attack and control deflection are of the same sign.

14. SUBJECT TERMS

aerodynamics, Aeroprediction Code, Mach number, trailing edge flap

15. NUMBER OF PAGES

57

16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORTS	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
UNCLASSIFIED	UNCLASSIFIED	UNCLASSIFIED	UL

NSN 7540-01-280-5500

Standard Form 298 (Rev 2-89)
Prescribed by ANSI std. Z39-18
298-102

FOREWORD

An idea that has been considered to provide control to some advanced guided projectile concepts is to deflect the rear part of the tail surface as opposed to the entire tail surface. This concept offers an advantage over deflecting the entire tail surface in terms of control volume, weight and cost. However, to estimate the aerodynamics of these types of aerodynamic concepts with the aeroprediction code requires hand calculations and engineering judgment over much of the range of flight conditions of interest. As a result, new technology has been developed to be integrated into the next version of the aeroprediction code which will allow aerodynamics of trailing edge flaps to be computed in an automated and more accurate manner than currently available approximate methods. This report documents this new aerodynamic prediction methodology for trailing edge flaps.

The work described in this report was supported by the Office of Naval Research through the Surface Weapons Systems Technology Program managed at the Naval Surface Warfare Center, Dahlgren Division (NSWCDD) by Mr. Robin Staton. Tasking from this program was provided by Mr. Roger Horman and Mr. John Fraysse. The authors express appreciation for support received in this work.

Approved by:

DANNY BRUNSON, Head
Weapons Systems Department

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1.0 INTRODUCTION

Conventional approaches to control weapons include aerodynamic controls where either the canards, wings, or tails are deflected a given amount to provide the required maneuverability to intercept a target. More recently, as weapons have attempted to hit targets flying at high altitudes where the atmosphere is quite thin, side jet thrusters are being considered to replace or compliment aerodynamic controls. These side jet thrusters can also be effective in cases where very short airframe time constants (the time it takes for a control system to generate about 63 percent of the required maneuverability) are required. For both the conventional aerodynamic or propulsive means of control, cost and control hardware weight and volume become prohibitive for some applications. These lower cost applications tend to be associated with stationary or slowly moving targets where maneuverability requirements are low, but improvements in accuracy over a ballistic weapon are necessary.

One idea that has been considered to meet the lower cost, lower maneuverability control concept is to deflect a part of a wing or tail surface as opposed to the entire surface. The portion of the tail surface considered for deflection is at the tail or wing trailing edge. Figure 1 is an illustration of a typical concept being considered where a part of the trailing edge portion of the tail fin is being considered for the control surface as opposed to the entire tail surface. As seen in Figure 1, this trailing edge flap concept is associated with a very low drag projectile design, and given a high initial velocity, can produce a fairly long range, even without a rocket motor. For long ranges, winds and other ballistic errors can produce sizeable miss distances without some sort of corrective device. While the large tail fins of the Figure 1 concept are needed for stability at a high velocity launch, deflecting the entire tail fin a significant amount to eliminate the ballistic errors is not needed. Only a fraction of the tail surface is required to provide adequate maneuverability if the deflection occurs over a sustained period of time. The small deflected surface area results in a much lower volume, weight and cost for the control system. As seen in Figure 1, the amount of area of the trailing edge can vary depending on the requirements. Shown in the figure is a variable semispan, root chord and hinge line for the trailing edge flap.

The most recent version of the NSWC aeroprediction code (AP98)[1] distributed to users is not capable of computing aerodynamics on a concept such as that shown in Figure 1 when the trailing edge flaps are deflected. The objective of this report is to develop the methodology to allow the next version of the aeroprediction code (AP02) to compute aerodynamics on a configuration where some portion of the rear part of the aft lifting surface (either wing or tail) can be deflected to provide control. In developing this trailing edge flap aerodynamic predictive methodology, considerations of the cost to integrate the new methodology into the aeroprediction code (APC) will be a prime driver in the method chosen.

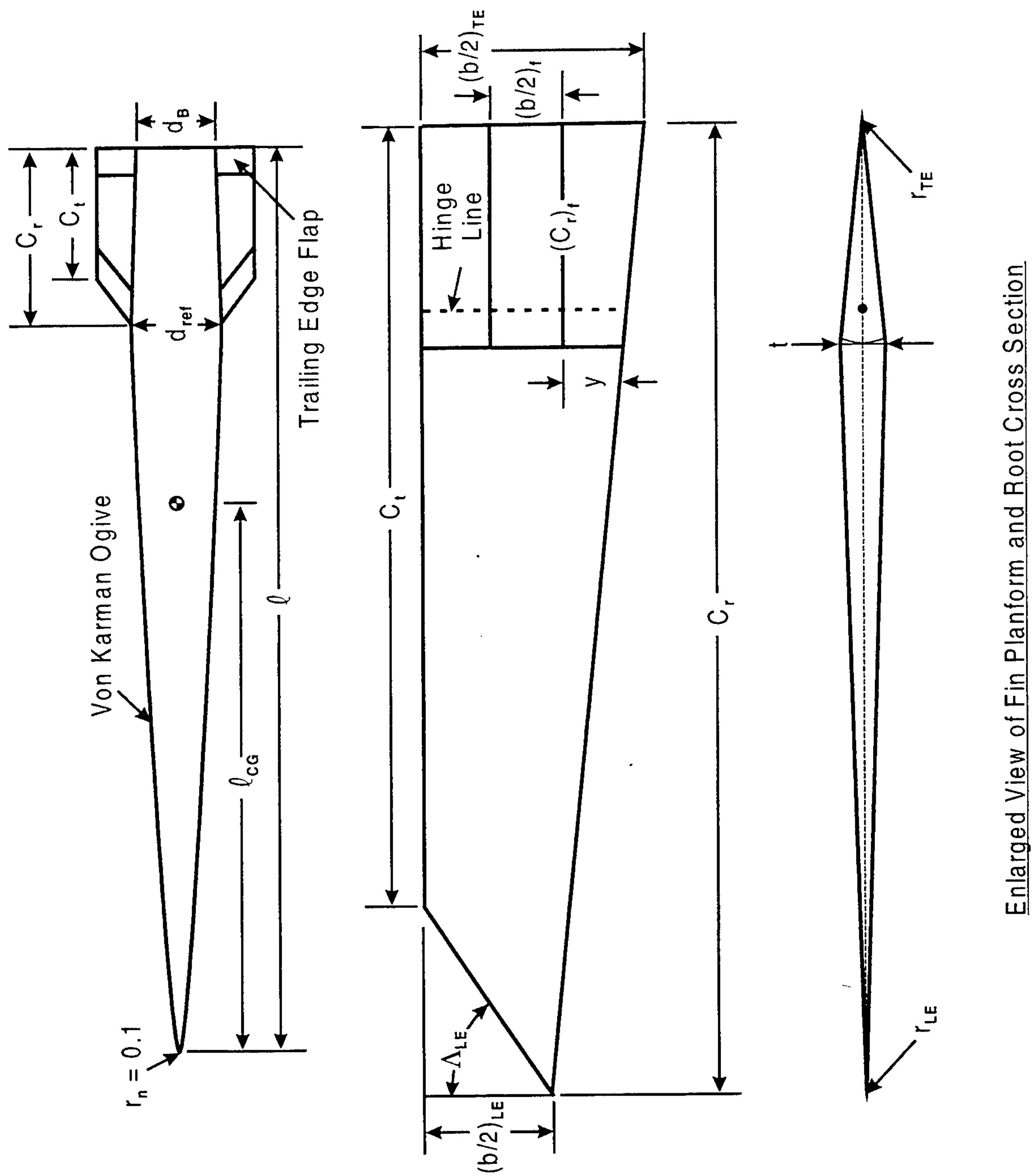


FIGURE 1. PROJECTILE CONCEPT WHERE TAIL TRAILING EDGE FLAP IS DEFLECTED FOR CONTROL

In reviewing the literature to determine approaches to use for calculating the aerodynamics of trailing edge flaps, the general approach that comes closest to that desired for use in the future AP02 is that adopted for the Missile Datcom.[2] In that approach, an equivalent value of deflection for the entire wing or tail surface is determined to reflect a given flap deflection. In other words

$$\delta_W = f(\delta_f) \tag{1}$$

The equivalent value of δ_W is determined offline using methods in the airplane DATCOM[3] at subsonic speeds and the method of Goin[4] at supersonic speeds. The advantage of an approach such as Equation (1) for codes such as Missile Datcom[2] or AP98[1] is that this is the least costly and most straightforward approach to incorporate the computation of aerodynamics of trailing edge flaps into an existing computer code. The low cost is because codes such as AP98[1] or Missile Datcom[2] are generally already set up logic wise to compute the aerodynamics of a configuration where one set of fins are deflected. Hence, if one can define what that wing deflection is in terms of some flap deflection, the codes[1,2] can be exercised to provide a set of aerodynamics that simulate a configuration with a trailing edge flap deflected by a given amount.

While the approach used by the Missile Datcom [Equation (1)] to compute aerodynamics of trailing edge flaps is the same approach that will be adopted for use here, the methods that will be used for the AP02 will differ from those[3,4] used in the Missile Datcom.[2] There are several reasons for this. First of all, the method of Goin[4] has too many limitations. Some of these limitations include requirements for supersonic leading and trailing edges of the flap hinge line, viscous effects are not accounted for, and the method does not include nonlinearities due to large flap deflections or angles of attack (AOAs). Secondly, while the method of Reference 3 takes into account some of the viscous and nonlinear effects that Reference 4 does not account for, the method itself is inconsistent with that of Reference 4.

The objective here is thus to derive an improved method to compute aerodynamics of trailing edge flaps that utilize the Equation (1) approach. The method should be similar for both subsonic and supersonic freestream Mach numbers, should not be limited to supersonic leading and trailing edges, should account (at least empirically) for viscous effects, and should account for nonlinearities associated with large flap deflections or AOAs. From a practical standpoint, the weapons that will use the trailing edge flaps for control will typically fly at fairly small trim AOAs (less than 10 deg). However, flap deflections as large as ± 30 deg are not unreasonable in order to achieve the appropriate trim AOA desired. Also, from a practical standpoint, most applications will be below $M_\infty = 2.0$. However, the method should be general enough to be applied over the Mach number range of applicability of the AP98 or AP02, which is 0 to 20. On the other hand, the method will not be validated over this large Mach number range due to limited experimental data.

2.0 ANALYSIS

To most efficiently implement the methodology for computing the aerodynamics of a weapon concept that is controlled by trailing edge flaps, we will seek the definition of the equivalent wing deflection that yields the same normal force, pitching moment and trim AOA as that obtained by deflecting the trailing edge flaps. In mathematical terms,

$$N_{W(B)} + N_{B(w)} = N_f \, f_1 \tag{2}$$

$$M_{W(B)} + M_{B(W)} = N_f \, f_1 \left[(X_{CP})_f - X_{ref} \right] \tag{3}$$

$$(\alpha_{TR})_W = (\alpha_{TR})_f \tag{4}$$

In reality, if Equations (2) and (3) are satisfied, Equation (4) will automatically be satisfied. We thus must define the relationships that allow Equations (2) and (3) to be satisfied.

Notice that in Equations (2) and (3), the wing-body normal force and pitching moments are equated to the normal force and pitching moment coefficients of the flap alone (with no interference effects present) times an empirical constant. There are a couple of reasons for this. First, when the entire wing is deflected it will have carryover normal force onto the wing. This means the equivalent control deflection of the entire wing will be lower than if no carryover normal force were present. Secondly, while there will be some interference carryover normal force onto the flap from the wing or body, this extra normal force can be lumped into an empirical term, f_1, which will be defined later.

Equation (2) can be expanded as

$$\left[C_{N_{W(B)}} + C_{N_{B(w)}} \right]_{\delta_W} Q \, A_{ref} = \left[C_N \right]_{\delta_f} f_1 \, Q \, A_{ref}$$

$$\text{or} \tag{5}$$

$$\left(C_{N_\alpha} \right)_W \left[k_{W(B)} + k_{B(W)} \right] \delta_W = \left(C_{N_\alpha} \right)_f f_1 \, \delta_f$$

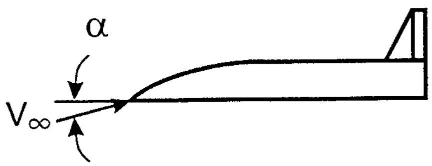
The empirical factor, f_1, of Equation (5) accounts for several physical phenomena. These include boundary layer buildup and separation of the flow ahead of the flap on the wing surface; flap thickness effects; effects of the slot created between the wing and flap when the flap is deflected; and interference effects of the flap onto the body or wing, or the wing or body onto the flap. The factor f_1 will be determined empirically based on experimental data for wings which have trailing edge flaps. Figure 2 attempts to pictorially and mathematically show the representation of a trailing edge flap deflection by deflecting the full wing.

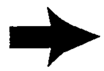
To determine f_1, we equate the right hand side of Equation (5) to the change in normal force coefficient at some AOA due to a control deflection δ_f. That is

Physical Representation

Reality

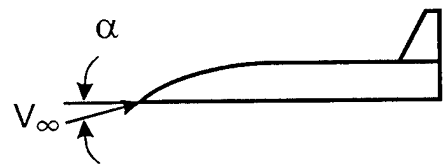
Body-Tail With T.E. Flap



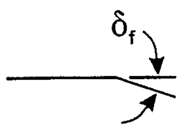


Approximation

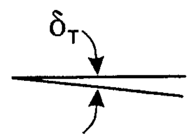
Body-Tail



Tail With T.E. Flap



Tail With No Flap



Mathematical Representation

$$\left(C_{N_\alpha}\right)_f f_1 \, \delta_f =$$

$$\begin{pmatrix}\text{Normal force coefficient} \\ \text{of flap deflected an amount } \delta_f\end{pmatrix} =$$

$$\delta_T \left(C_{N_\alpha}\right)_T \left(k_{T(B)} + k_{B(T)}\right)$$

$$\begin{pmatrix}\text{Normal force coefficient of} \\ \text{tail deflected on amount } \delta_T\end{pmatrix}$$

Physics f_1 Accounts for

- Boundary layer buildup and separation ahead of flap
- Slot between wing and flap
- Interference effects of wing/body on flap and vice versa due to flap deflection

- Viscous effects completely different (boundary layer, no separation bubble)
- No slot present
- Interference effects automatically included

FIGURE 2. PHYSICAL AND MATHEMATICAL REPRESENTATION OF TRAILING EDGE FLAP DEFLECTION BY FULL WING DEFLECTION

$$f_1 = \frac{\left(\Delta C_{N_f}\right)}{\left(C_{N_\alpha}\right)_f \delta_f f_2} \tag{6}$$

ΔC_{N_f} of Equation (6) is the additional normal force coefficient created by a flap deflection δ_f. $\left(C_{N_\alpha}\right)_f$ is the theoretical normal force coefficient slope for the flap of given aspect ratio and taper ratio at a given Mach number and AOA. This theoretical value is determined by the methods in the AP02 for a flap only (no wing ahead of it). The AP02 methods include linearized theories at low AOA or control deflection combined with empirical approaches at higher AOA. These methods in the AP98 or AP02 are fairly general and can calculate aerodynamics on supersonic or subsonic leading edge wings or flaps at low AOA. Also, aerodynamics can be computed for Mach numbers 0 to 20 and AOAs to 90 deg. Hence, the theoretical methodology for computing $\left(C_{N_\alpha}\right)_f$ is fairly general. The value of $\left(C_{N_\alpha}\right)_f$ is actually computed using a secant slope for a given AOA. This value of $\left(C_{N_\alpha}\right)_f$ is then multiplied by the given flap deflection, δ_f as seen in Equation (6). The numerator of Equation (6) is based on experimental data, which accounts for various physical phenomena of a flap in conjunction with a wing, which a wing alone does not have. Hence, the empirical factor f_1 is generated by the ratio of experimental data for a flap on a wing to a theoretical wing alone solution.

The factor f_2 in the denominator of Equation (6) is used to account for the fact that the theory in the AP02 which defines the lift curve slope of an entire wing deflected an amount δ at a given AOA may not accurately predict the increment in normal force generated by a flap. The factor f_2 is expected to be near one at supersonic speeds. However, at subsonic speeds, wind tunnel data suggests the theoretical predictions of additional normal force generated by a flap are higher than what the theory suggests. This inaccuracy of the theory arises from using the secant slope for $\left(C_{N_\alpha}\right)_f$ versus using the local slope at a given value of α. At supersonic speeds, use of the secant slope does not appear to be a problem. However, subsonically, the C_N versus α curve levels out at around 25 to 30 deg AOA, so an additional increase in α brings increasingly less increase in C_N. Using a secant slope for $\left(C_{N_\alpha}\right)_f$ versus the local tangent gives a value of $\left(C_{N_\alpha}\right)_f$ which is too large and therefore a value of f_1 which is too low. The parameter f_2 therefore corrects for this weakness. One could change the overall AP02 code to use local versus secant slopes. However, this would be a very costly and time consuming process, and it was much more cost effective to define the factor f_2 to take care of this correction.

In Equation (6), it is assumed both the numerator and denominator are based on the same reference area A_{ref}. If $\left(C_{N_\alpha}\right)_f$ is calculated based on a wing alone solution for the flap, then the Equation (6) must be multiplied by A_{ref}/A_f to have consistent reference areas.

To define the empirical factor f_1, two data bases will be used.[5,6] Reference 5 contains data for a canard-body-tail configuration (see Figure 3) with trailing edge flaps. Data is available for Mach numbers 1.5 to 4.63, AOAs −2 to about 30 deg (except for $M_\infty = 1.5$ where some data

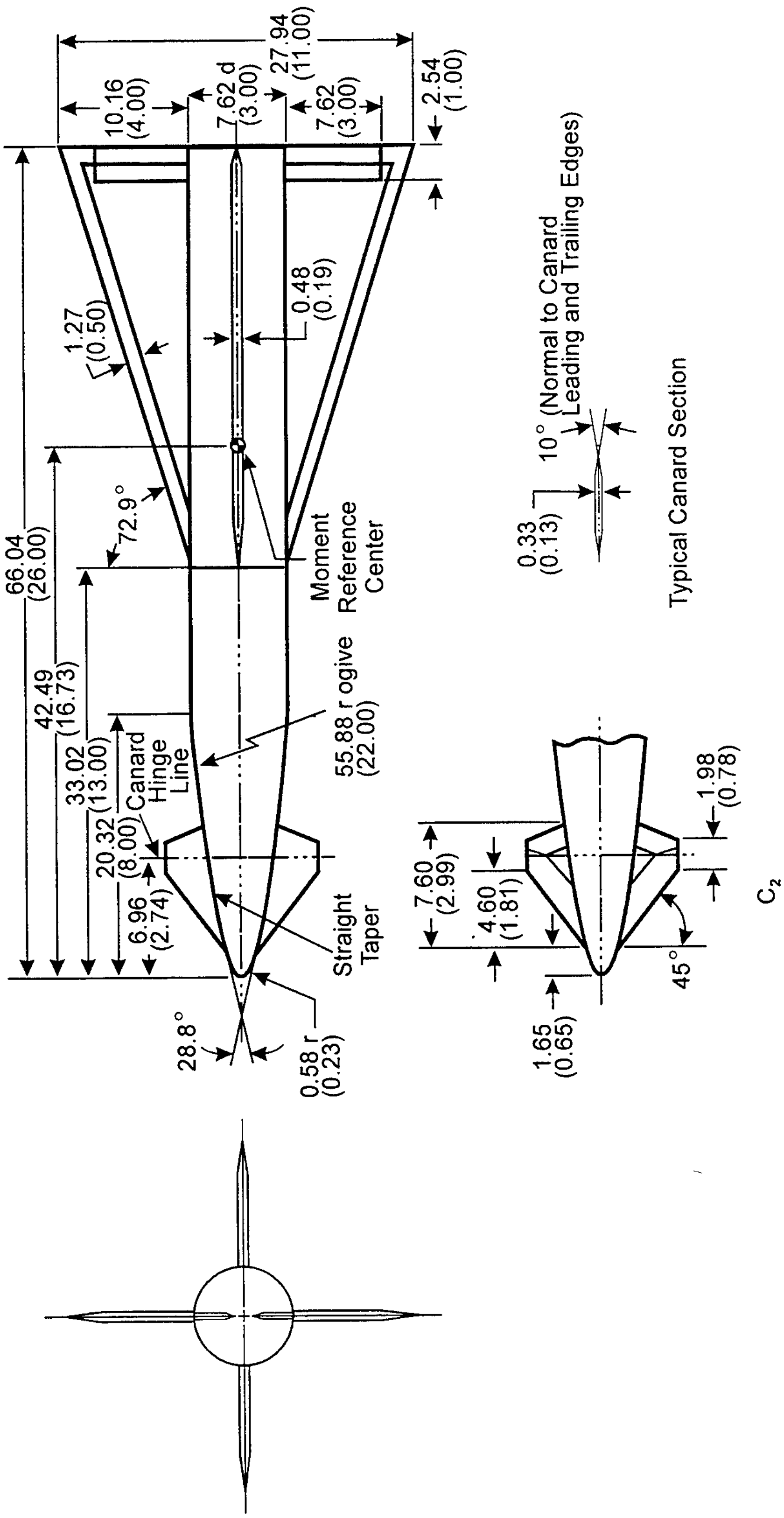


FIGURE 3. DRAWING OF THE MODEL USED FOR SUPERSONIC TESTS[5]
[ALL LINEAR DIMENSIONS IN CENTIMETERS (INCHES).]

is available only to about 15 deg AOA), and control deflections 0 to −30 deg. Unfortunately, no positive values of δ_f were available in Reference 5, probably since a negative value of δ_f is required for trim to occur when α is positive.

Reference 6 contains data for low Mach numbers ($M_\infty = 0.3$ to 0.5) for several different configurations. These configurations included an elliptical and circular cylinder-shaped bodies with either a delta or sweptback rectangular wing. The wings could have either a leading or trailing edge flap. The configuration of most interest here is the delta wing with trailing edge flaps on a circular cylinder body (see Figure 4). Data is available to 40 deg AOA for flap deflections of ±10 and ±30 deg. Hence, Reference 6 will complement the supersonic data of Reference 5.

For Mach numbers in between $M_\infty = 0.4$ and $M_\infty = 1.5$, the following procedure will apply for computing f_1. For Mach numbers below $M_\infty = 0.8$, the value of f_1 computed at $M_\infty = 0.4$ will be assumed to apply. For Mach numbers between $M_\infty = 1.5$ and 0.8, linear interpolation will be used to compute f_1 based on the values of f_1 at $M_\infty = 1.5$ and 0.8.

Figures 5 and 6 give the values of f_1 determined by using References 5 and 6 to find values of $(\Delta C_N)_f$ and Reference 1 to compute a value of $(C_{N_\alpha})_f$ at a given AOA. Figure 5 is when α and δ are of opposite signs, which is the practical case for trim when the aft located control surface is deflected. Figure 5 applies for $M_\infty \geq 1.5$ and for values of α and δ of the same sign when α is numerically small. No data has been found to ascertain the validity of Figure 5 when α and δ are the same sign and α is greater than a small value. For $M_\infty > 4.63$, the value of f_1 at $M_\infty = 4.63$ will be assumed. Also Figure 5 holds for values of δ_f up to −30 deg, based on the Reference 5 data.

Figure 6 gives values of f_1 for $M_\infty = 0.4$ for values of α up to 30 deg and for values of δ_f of ±30 deg. Figure 6 values of f_1 utilize the values of f_2 from Figure 7. Figure 7A presents the most practical case for tail-located trailing edge flaps since α and δ_f must be of opposite signs for trim to occur. Figures 7B and 7C present results for f_2 when α and δ_f are of the same sign. Figure 7B is for $\delta_f = 10$ deg and Figure 7C is for $\delta_f = 30$ deg. Linear interpolation of the Figures 7B and 7C will occur for values of δ_f other than 10 or 30 deg.

Knowing f_1 from Figures 5 or 6, Equation (5) can be rewritten as

$$\delta_W = \left[\frac{(C_{N_\alpha})_f \; f_1}{(C_{N_\alpha})_W \left(k_{W(B)} + k_{B(W)}\right)} \right] \delta_f \tag{7}$$

The way Equation (7) is utilized within the AP02 is as follows:

1. For a given flap size, $(C_{N_\alpha})_f$ is computed from the wing alone solution in the AP02 at a given M_∞, α, AR and λ. This value of $(C_{N_\alpha})_f$ is then related to A_{ref} versus A_f.

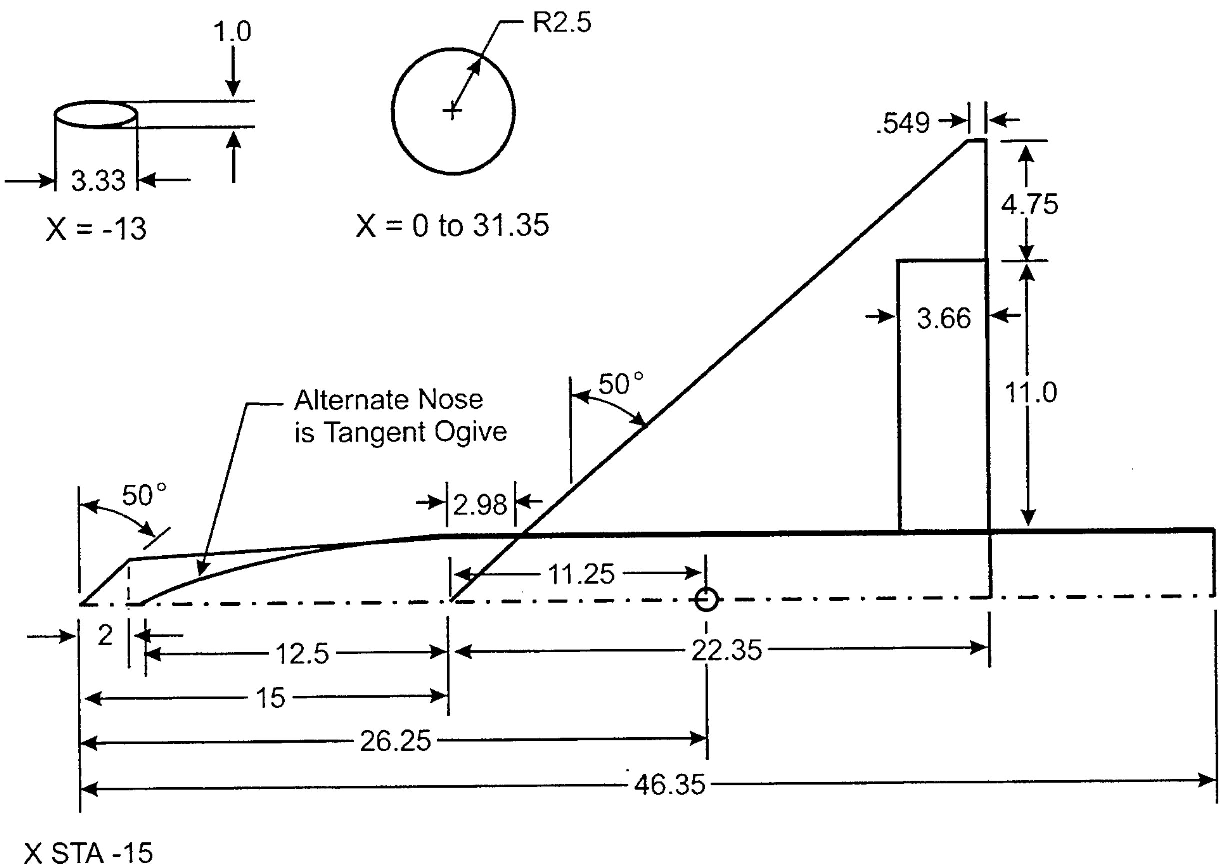


FIGURE 4. DELTA WING PLANFORM USED FOR SUBSONIC TESTS[6]
(ALL DIMENSIONS IN INCHES)

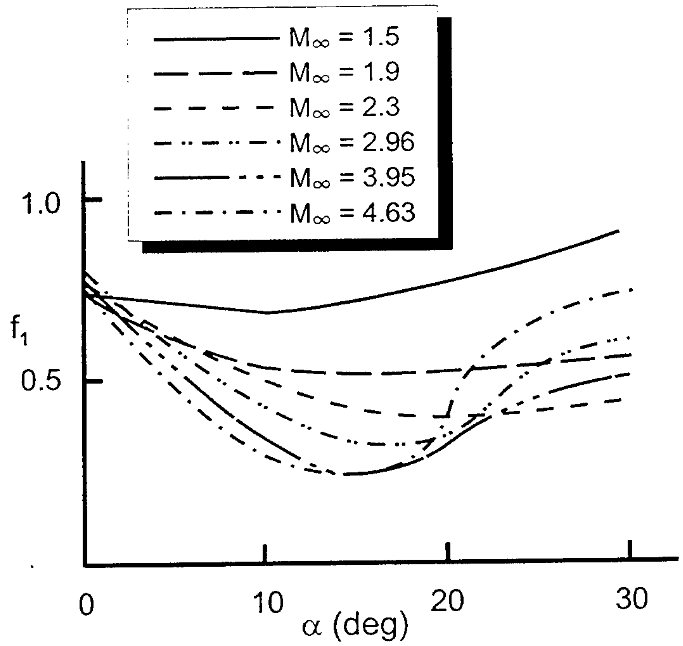


FIGURE 5. VALUE OF PARAMETER f_l AT SUPERSONIC SPEEDS BASED ON REFERENCE 5 DATA AND AP98

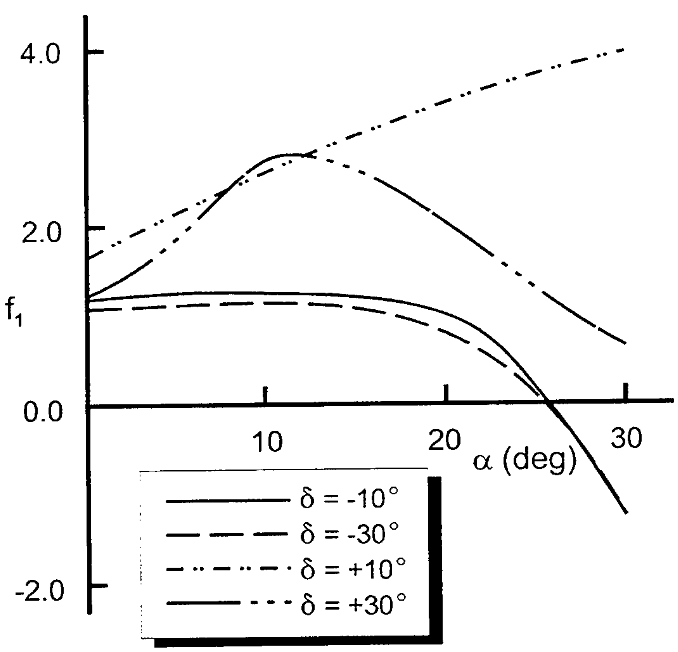


FIGURE 6. VALUE OF PARAMETER f_l AT SUBSONIC SPEEDS BASED ON REFERENCE 6 DATA AND AP98

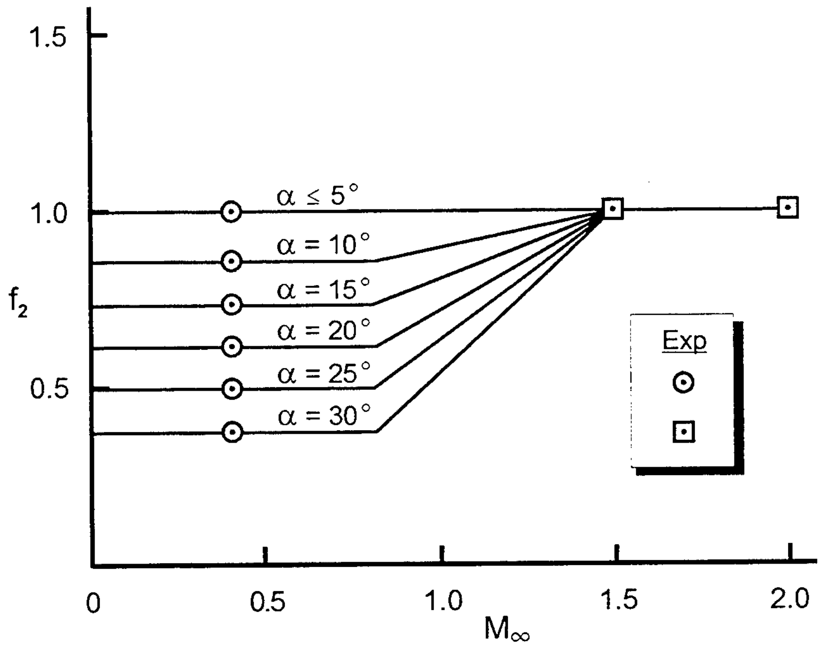


FIGURE 7A. FACTOR WHICH CORRECTS FOR USE OF SECANT VERSUS TANGENT IN NORMAL FORCE CURVE SLOPE (α AND δ OF OPPOSITE SIGNS)

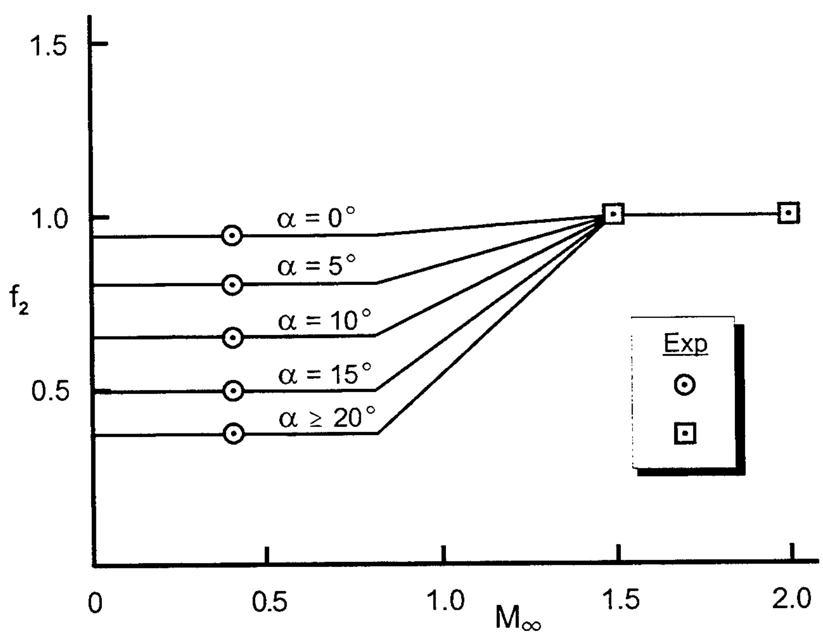


FIGURE 7B. FACTOR WHICH CORRECTS FOR USE OF SECANT VERSUS TANGENT IN NORMAL FORCE CURVE SLOPE ($\delta_f = 10$ DEG)

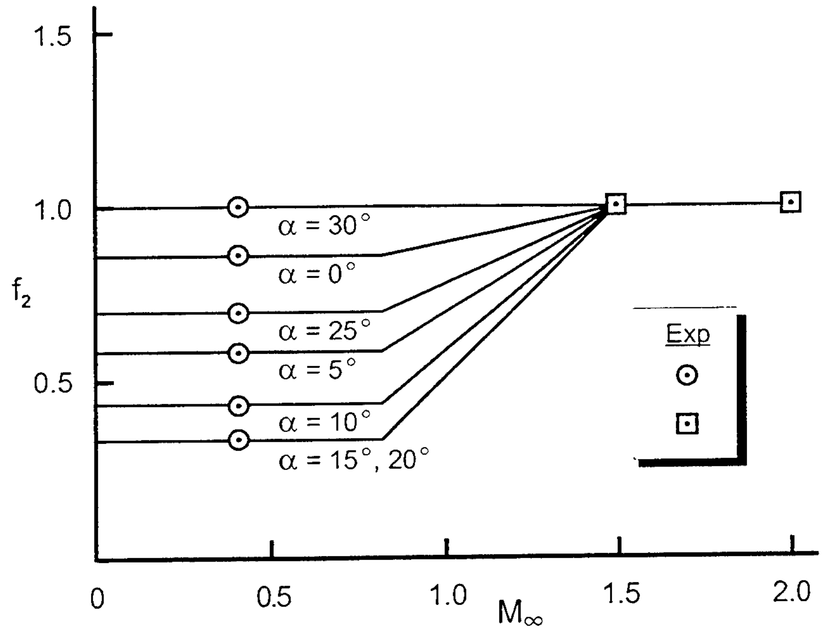


FIGURE 7C. FACTOR WHICH CORRECTS FOR USE OF SECANT VERSUS TANGENT IN NORMAL FORCE CURVE SLOPE (δ_f = 30 DEG)

2. f_l is then computed via table lookup for a given value of α, M_∞ and δ_f (if the flow is subsonic.

3. For a given wing size, $\left(C_{N_\alpha}\right)_W$ is computed from the wing alone solution in the AP02 at a given M_∞, α, AR and λ. This value of $\left(C_{N_\alpha}\right)_W$ must again be based on A_{ref}.

4. Values of $k_{W(B)}$ and $k_{B(W)}$ are then computed at a given α using the nonlinear control methodology in the AP02. This methodology uses slender body theory as a basis for low AOA estimates and wind tunnel data at high AOA to modify these estimates.

5. For a given value of δ_f, an effective value of δ_W is then computed based on Equation (7). This value of δ_W is the amount the entire wing is deflected to approximate the additional normal force of a wing due to a trailing edge flap deflection of an amount δ_f.

Equation (7) defines the equivalent fin deflection to give the same normal force that deflecting the rear part of the fin an amount δ_f would give. The normal force coefficient of the flap or fin is computed from Equation (6). That is

$$\left(\Delta C_N\right)_f = f_l \left(C_{N_\alpha}\right)_f \delta_f \tag{8}$$

The question that we must now address is the pitching moment for the flap. By deflecting the entire wing an amount δ_f defined by Equation (8), the pitching moment for the wing will be based on the center of pressure of the entire wing, not that due to the flap. Thus to obtain the correct pitching moment for the flap, where the entire wing is deflected, a change in the center of pressure must be calculated.

Most trailing edge flaps under consideration have a fairly high aspect ratio with a fairly small root chord. The initial thought by the author was to assume the center of pressure of the normal force generated by the trailing edge flap would be similar to that on a high aspect ratio wing alone. That is for subsonic flow, the center of pressure would be around the quarter chord location and then transition to the half chord location around $M_\infty = 2.0$. However, in comparing this assumed location to the experimental data of References 5 and 6, it was clear this assumption on center of pressure location was not correct. It is believed the reason for the center of pressure assumption not being correct is that the flap cannot be treated as a wing in isolation at most Mach numbers. At a Mach number of 1.5, the assumption of ¼ chord transitioning to ½ chord supersonically was a good assumption (see Figure 8). However, at other Mach numbers, assuming the center of pressure of the flap normal force could be treated similar to a high aspect ratio wing in isolation became increasingly erroneous as seen by the experimental data of Figure 8. In giving the behavior of the experimental data in Figure 8 some thought, the author believes that the physics of the flow can explain the Figure 8 experimental data. That is, as Mach number increases and the trailing edge flap is deflected, a shock is created ahead of the leading edge of the flap. This shock in turn creates a high pressure region on the wing where the flap is attached. This high pressure region is the reason for the experimental center of pressure of the flap normal force actually lying ahead of the leading edge of the flap as seen by Figure 8. The dashed line in Figure 8 is the new assumed center of pressure of the flap normal force as a function of Mach number. Notice that in Figure 8, $\left[(X_{CP})_f / C_r\right]_{avg}$ represents the average center of pressure over the AOA range from 0 to 30 deg as a fraction of the root chord of the flap.

At a subsonic Mach number of 0.4, the center of pressure also lies ahead of the flap. If the flap deflection has the same sign as the AOA, this center of pressure location is about 0.7 chord lengths ahead of the flap leading edge. If the flap deflection is of opposite sign to the AOA, the center of pressure is about 0.4 chord lengths ahead of the flap leading edge. For Mach numbers 0 to 0.8, it is assumed these values of 0.4 and 0.7 chord lengths hold constant. For Mach numbers 0.8 to 1.5, it is assumed the location of the flap center of pressure varies linearly between the values at $M_\infty = 0.8$ and 1.5.

The physics which cause the center of pressure to move ahead of the flap are believed to be different for the subsonic and supersonic cases. Supersonically, it is believed viscous effects as well as the shock structure are the dominant features. However, subsonically, it is believed the flap deflection rearranges the pressure distribution on the wing ahead of the flap as well as the viscous effects, which are present at all Mach numbers. The rearrangement of the pressure distribution on the wing ahead of the flap occurs because in subsonic flow, disturbances in the flow can feed forward, whereas supersonically they cannot, except through the boundary layer.

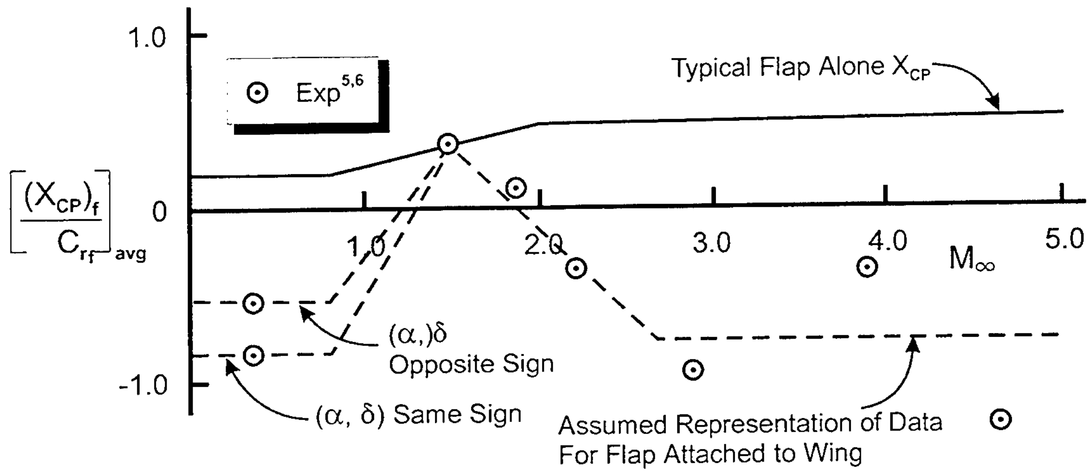


FIGURE 8. FLAP ALONE AND TRAILING EDGE FLAP ATTACHED TO WING AVERAGE CENTER OF PRESSURE OVER ANGLE OF ATTACK RANGE OF 0 TO 30 DEG FOR VARIOUS MACH NUMBERS

From a practical standpoint, the effect of the flap center of pressure shift diminishes its effectiveness somewhat in generating trim AOA. This is because the center of pressure of the normal force actually lies in front of the flap at most Mach numbers, decreasing the moment somewhat and hence the trim AOA. On the other hand, if the flap is located near the base of a fairly long body, a one to four inch shift in the center of pressure forward can be fairly small in terms of the overall moment arm. The amount of normal force created does not seem to be affected by the forward shift in center of pressure for trailing edge flaps.

The center of pressure of the trailing edge flap is therefore

$$\frac{(X_{CP})_f}{\ell_{ref}} = \frac{(X_{LE})_W + C_{r_w} - C_{r_f} f_3 - X_{ref}}{\ell_{ref}} \tag{9}$$

The term f_3 of Equation (9) is based on the empirically defined dotted lines of Figure 8. That is

$$\begin{aligned} f_3 &= +1.5 \text{ for } M_\infty \le 0.8 \text{ and } (\alpha, \delta) \text{ opposite signs} \\ &= +1.8 \text{ for } M_\infty \le 0.8 \text{ and } (\alpha, \delta) \text{ same signs} \\ f_3 &= 2.53 - 1.29\, M_\infty \text{ for } 0.8 < M_\infty \le 1.5 \text{ and } (\alpha, \delta) \text{ opposite signs} \\ &= 3.17 - 1.71\, M_\infty \text{ for } 0.8 < M_\infty \le 1.5 \text{ and } (\alpha, \delta) \text{ same signs} \\ f_3 &= -0.84 + 0.96\, M_\infty \text{ for } 1.5 < M_\infty \le 2.7 \\ &= 1.75 \qquad \text{for} \qquad M_\infty > 2.7 \end{aligned} \tag{10}$$

Using Equations (9) and (10), the change in pitching moment created by the fact the wing is deflected to simulate the trailing edge flap deflection is then

$$(\Delta C_M)_f = -\frac{\Delta C_{N_f}}{\ell_{ref}}\{[(X_{CP})_f - (X_{CP})_W] + [(X_{CP})_W - X_{CG}]\} \tag{11}$$

Equation (11) represents the pitching moment coefficient of any configuration where the trailing edge flap deflection is approximated by deflecting the full wing. The first term of Equation (11) represents the difference in the center of pressure between the flap and wing whereas the second term represents the center of pressure of the wing normal force term relative to a reference location which is here taken to be the center of gravity of the vehicle. Of course, the center of pressure of the wing is computed in the AP02 using linear theory methods at low AOA and transitions to the centroid of the wing planform area at high AOA.

The major focus in the analysis for estimating the aerodynamics of trailing edge flaps has been to determine an equivalent tail deflection which will give normal force and pitching moments equal to those when the flap is deflected. No mention of axial force has been made to this point in time. The axial force coefficient will be different for an equivalent wing deflection based on a flap deflection δ_f. The flap deflection will generate an additional axial force term due to the fact δ_f will be generally much larger than δ_W. An approximate relation which can be used to calculate the increment in axial force coefficient that results from estimating the aerodynamics based on a wing deflection of δ_W versus a flap deflection of δ_f is

$$(\Delta C_A)_f \cong \Delta(C_N)_f\,[\sin|\delta_f| - \sin|\delta_W|] \tag{12}$$

ΔC_{N_f} of Equation (12) is the additional normal force contribution due to the flap. Sin $|\delta_f|$ takes the component of this normal force term in the axial direction. Sin $|\delta_W|$ subtracts off the component of axial force of the wing since this is automatically included in the AP02 calculations; to leave this term in the calculations would mean we would double account for the wing deflection axial force contribution.

3.0 RESULTS AND DISCUSSION

Equations (8) and (11) define the theoretical change in normal force and pitching moment coefficients due to a flap deflection. The value of ΔC_{N_f} computed by the theory is that value defined by

$$\Delta C_{N_f} = (C_N)_{\delta_{W=0}} - (C_N)_{\delta_{W\neq 0}} \tag{13a}$$

The value of δ_W in Equation (13a) is obtained from Equation (7) using the process defined earlier in the analysis section of this report. Using the values of δ_W from Equation (7) in the AP98, values of $(C_N)_{\delta_{W=0}}$ and $(C_N)_{\delta_{W\neq 0}}$ of Equation (13a) can be computed and then ΔC_{N_f} defined theoretically. This value of ΔC_{N_f} can then be compared to experimental data where ΔC_{N_f} is obtained using experimental data for $(C_N)_{\delta_{f=0}}$ and $(C_N)_{\delta_{f\neq 0}}$. That is

$$\Delta C_{N_f} = (C_N)_{\delta_{f=0}} - (C_N)_{\delta_{f\neq 0}} \tag{13b}$$

Likewise, experimentally measured values of ΔC_{M_f} can be defined as

$$\Delta C_{M_f} = (C_M)_{\delta_{f=0}} - (C_M)_{\delta_{f\neq 0}} \tag{14}$$

and compared to theoretical values computed from Equation (11). ΔC_{N_f} of Equation (11) comes from the theoretical values defined by Equation (13a). Thus comparison of ΔC_{N_f} values obtained by Equation (13a) to (12) and ΔC_{M_f} values obtained from Equation (11) to Equation (14) will allow us to determine the validity and accuracy of the new theory.

The first set of data we will consider is from Reference 5. The configuration tested in the wind tunnel is shown in Figure 3. Figures 9-14 compare theory and experiment for $(\Delta C_N)_f$ and $(\Delta C_M)_f$ at $\delta_f = -20$ deg and Mach numbers 1.5, 1.9, 2.3, 2.96, 3.95, and 4.63. Results are plotted as a function of AOA up to 30 deg. For Mach numbers 1.5 and 1.9, experimental data was not available up to 30 deg AOA, so data was shown where available. As seen in the figures, the theory does a reasonable job in matching the data for both ΔC_{N_f} and ΔC_{M_f}, except at $M_\infty = 4.63$ and $\alpha \geq 20$ deg. At these conditions the theory overpredicts the normal force and pitching moment increments somewhat. However, since this region is beyond the anticipated practical range of usage ($M_\infty < 2.0$, $\alpha < 20$, $|\delta_f| < 30$ deg), no effort will be made to try to improve upon the theory at this condition.

Also shown on the $(\Delta C_M)_f$ portion of Figures 9-14 are the results of assuming the center of pressure of the flap is based on the flap in freestream flow and with the flap attached to the trailing edge. The flap attached to the trailing edge computations take into account the center of pressure shift shown in Figure 8. Note that at $M_\infty = 1.5$, no shift is shown so the Figure 9 pitching moment results show no change between the flap alone and the flap attached. However, Figures 10-14 show a change in pitching moment between flap alone and the flap attached. As seen in Figures 10-14, using the Figure 8 results tend to show an improvement in pitching moment calculations over assuming the flap alone.

It is also worthwhile to reemphasize the fact that all the theoretical calculations shown in Figures 9-14 (as well as the figures which will follow) were computed by using the AP02 in conjunction with Equation (7) as described in the Analysis Section of the report.

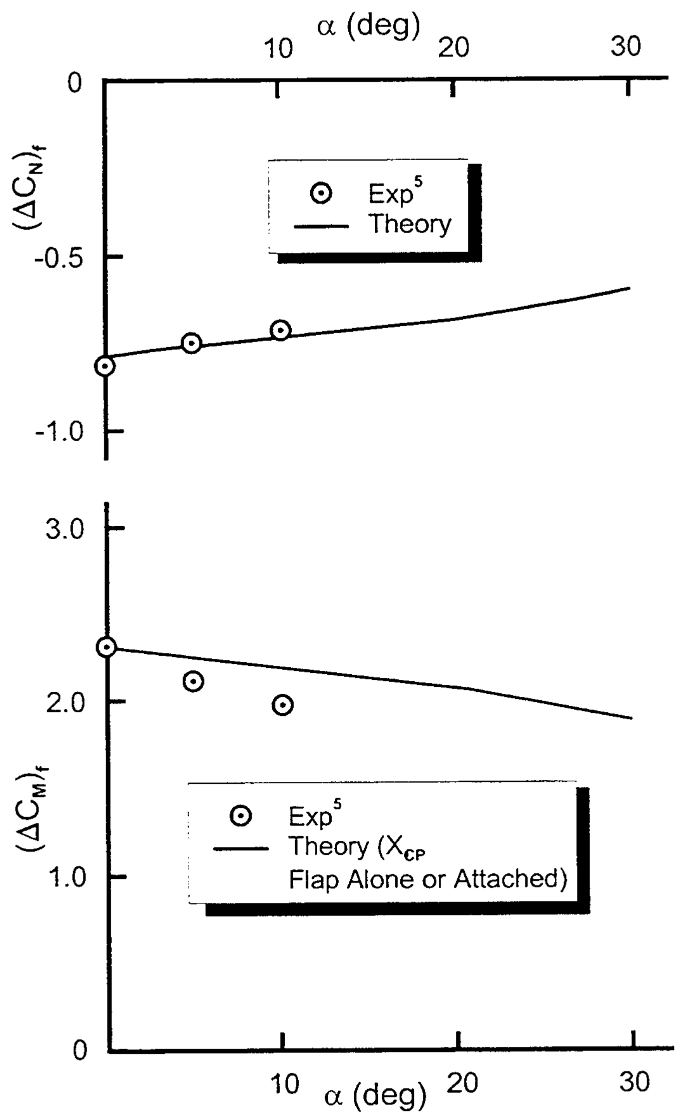


FIGURE 9. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 1.5$, $\delta_f = -20$ DEG)

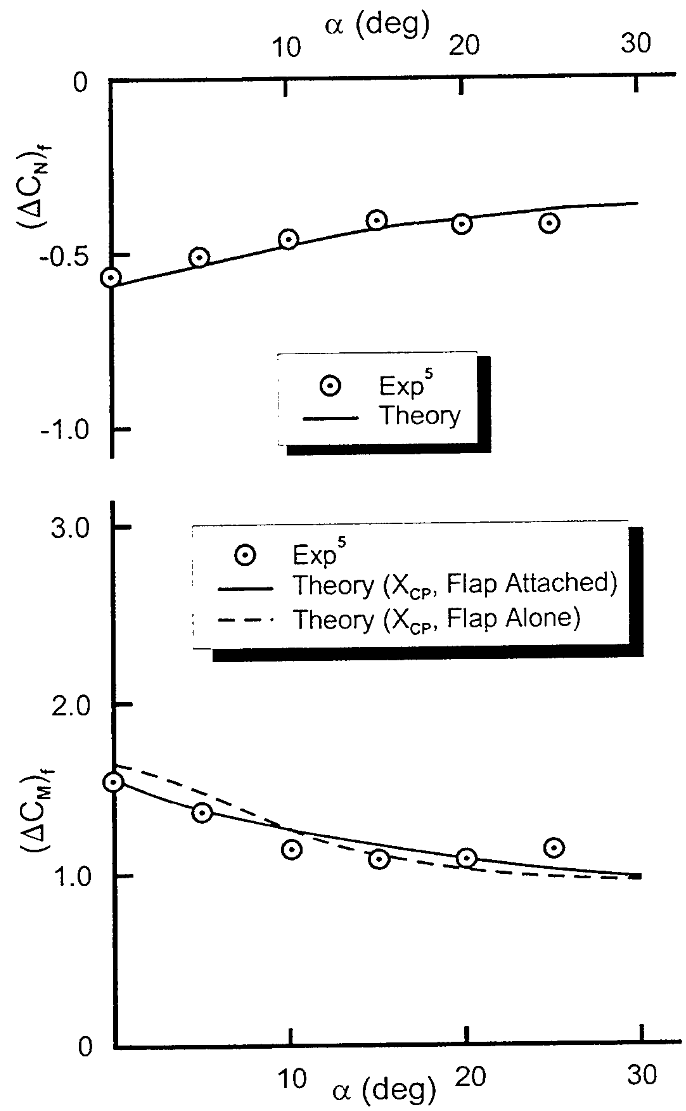


FIGURE 10. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 1.9$, $\delta_f = -20$ DEG)

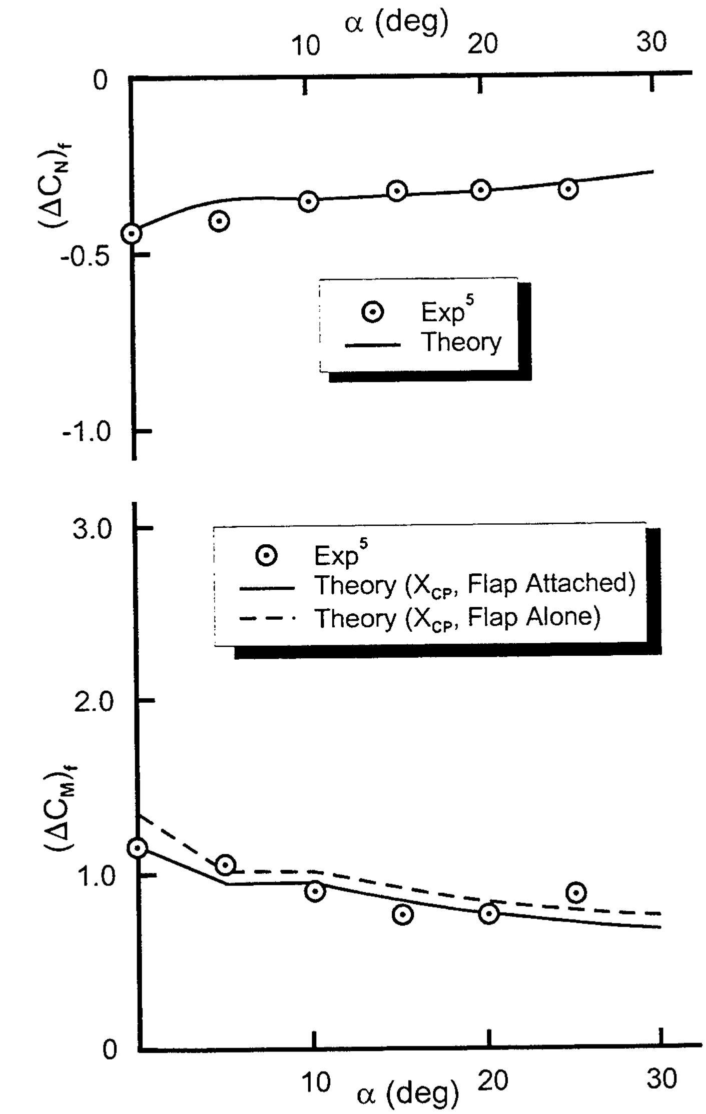


FIGURE 11. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 2.3$, $\delta_f = -20$ DEG)

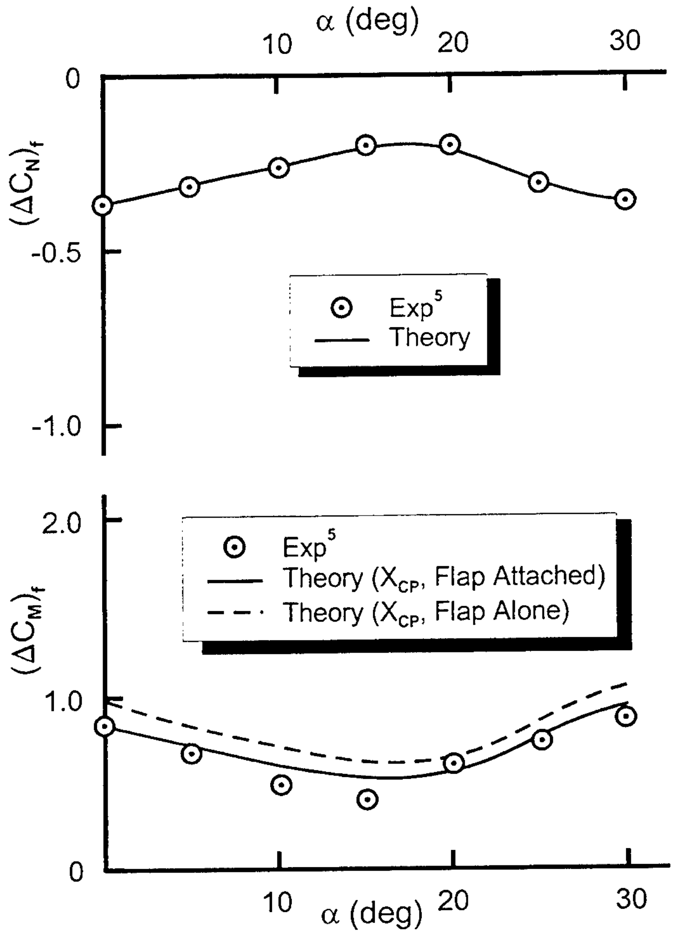


FIGURE 12. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 2.96$, $\delta_f = -20$ DEG)

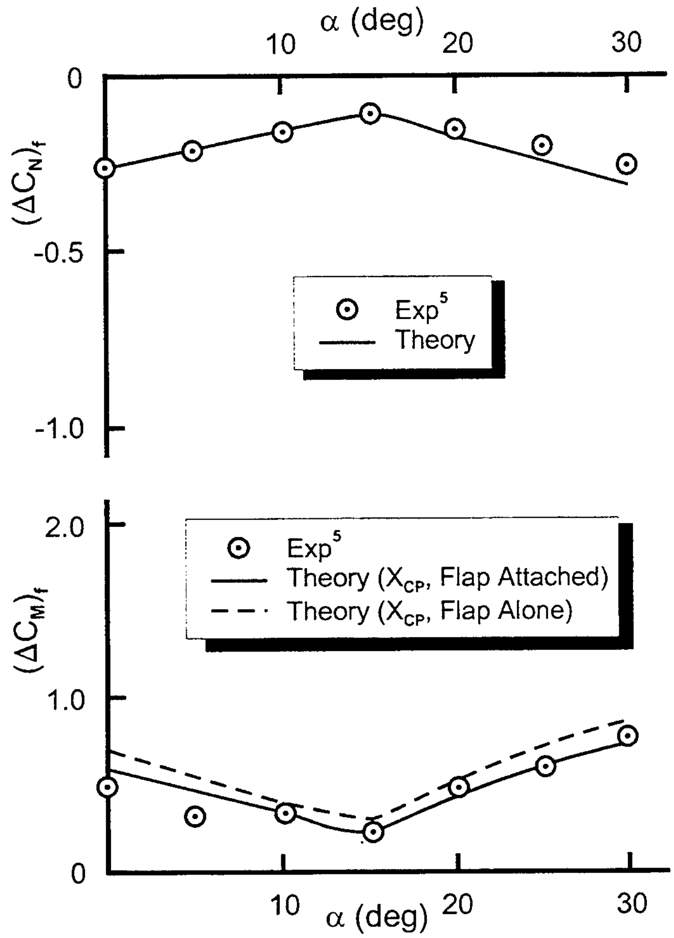


FIGURE 13. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 3.95$, $\delta_f = -20$ DEG)

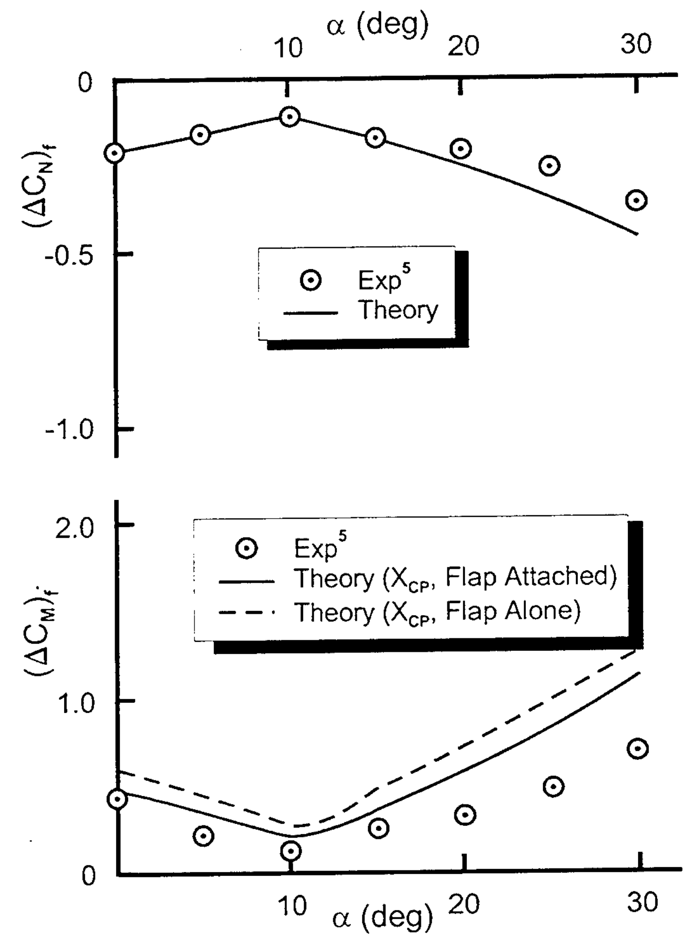


FIGURE 14. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 4.63$, $\delta_f = -20$ DEG)

Figures 15-20 compare theory and experiment for $(\Delta C_N)_f$ and $(\Delta C_M)_f$ as a function of flap deflection at AOA 10 deg and for the Mach numbers of the Reference 5 data base. Figures 15-20 are believed to be a more realistic representation of the practical case where trim is expected to occur for $\alpha \leq 10$ deg with the flap deflected as high as -30 deg. As seen in Figures 15-20, the theory and experiment are in fairly good agreement. All pitching moment data in Figures 15-20 assume the Figure 8 center of pressure shift. Note also that the theory shows a linear variation of $(\Delta C_N)_f$ and $(\Delta C_M)_f$ for $M_\infty \geq 1.5$ and δ_f to -30 deg for the small AOA of 10 deg.

Reference 6 represents the only subsonic data base the author found in the literature. The configuration tested is shown in Figure 4. The ogive of the Figure 4 configuration can be either an elliptical or a circular cylinder tangent ogive. The case upon which the change in pitching moments and normal force coefficients were determined was based on an elliptical nose. However, since the data used was $\Delta(C_N)_f$ and $\Delta(C_M)_f$, it is expected the body shape will have little impact since the same body shape is used for the $\delta_f = 0$ case as well as the $\delta_f \neq 0$ case.

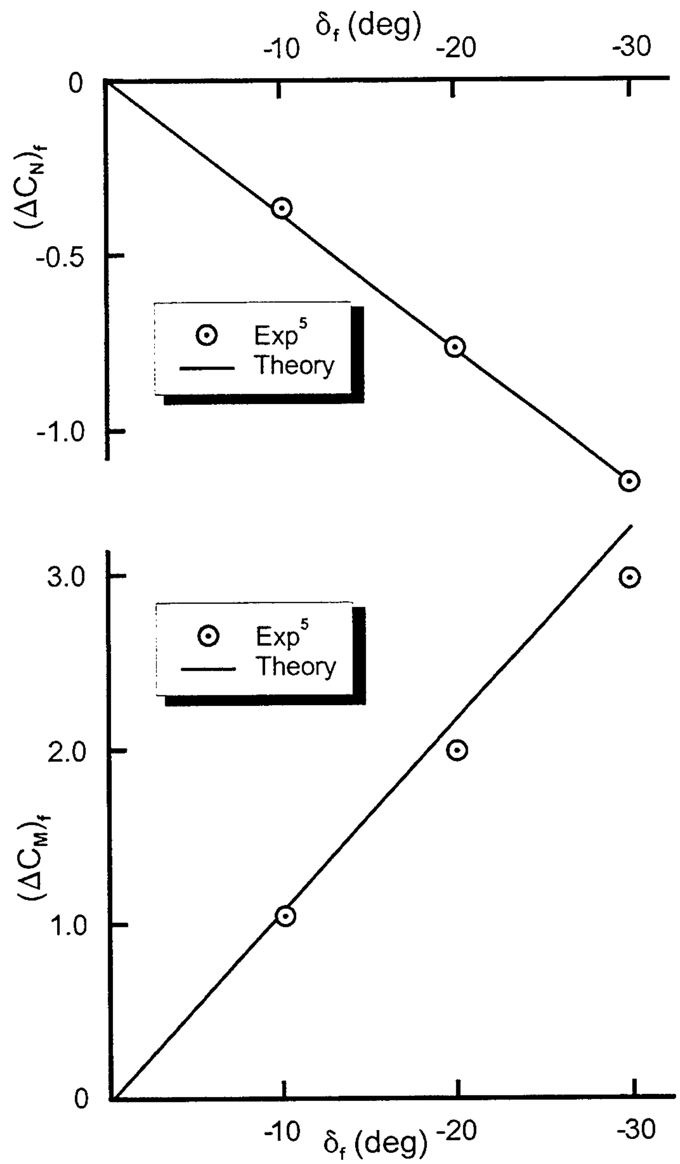


FIGURE 15. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 1.5$, $\alpha = 10$ DEG)

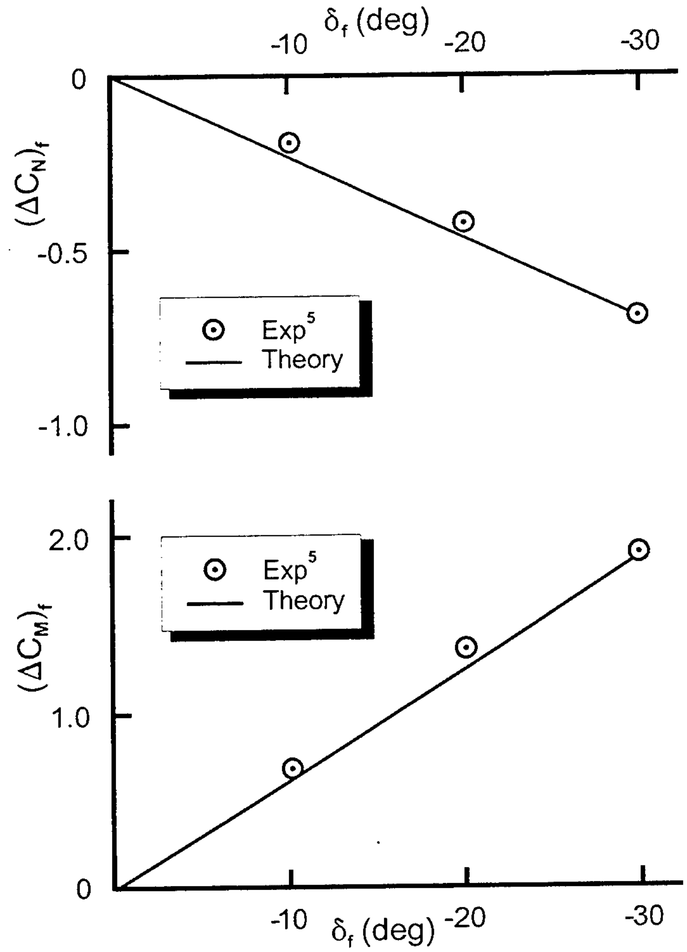


FIGURE 16. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 1.9$, $\alpha = 10$ DEG)

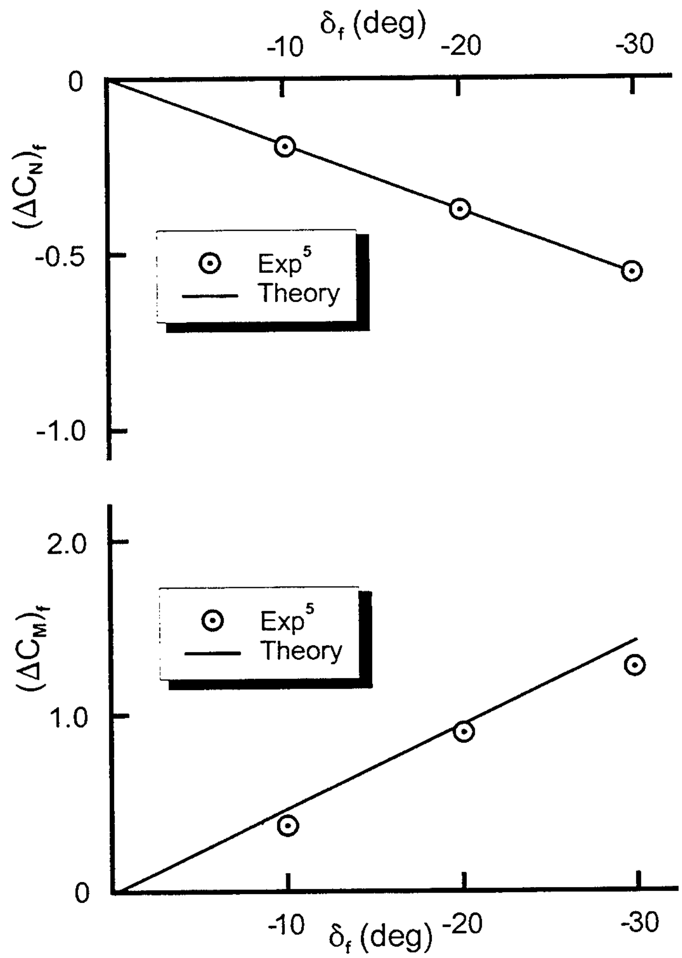


FIGURE 17. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 2.3$, $\alpha = 10$ DEG)

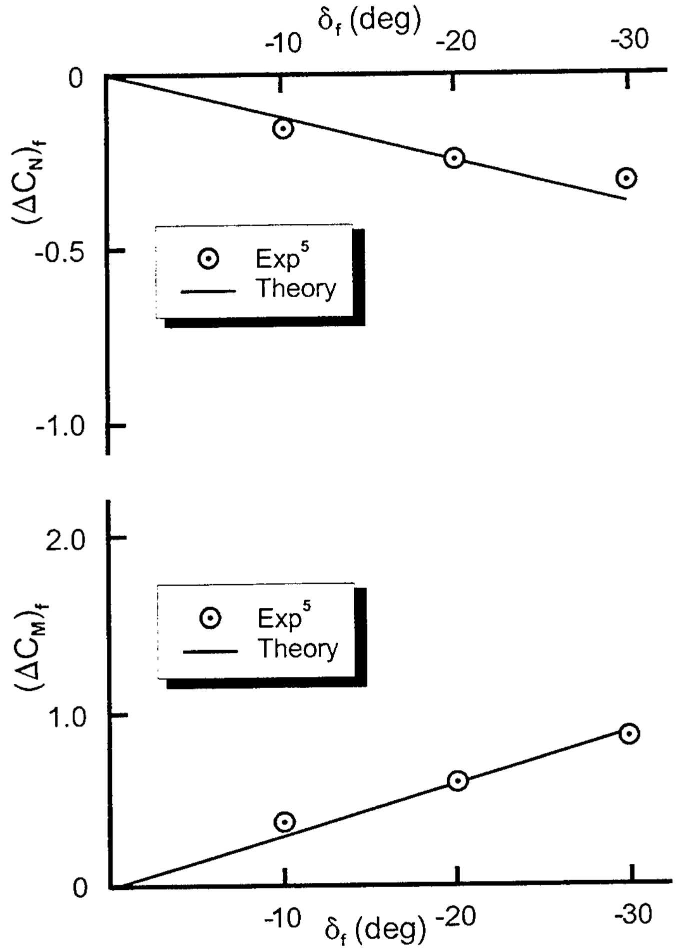


FIGURE 18. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 2.96$, $\alpha = 10$ DEG)

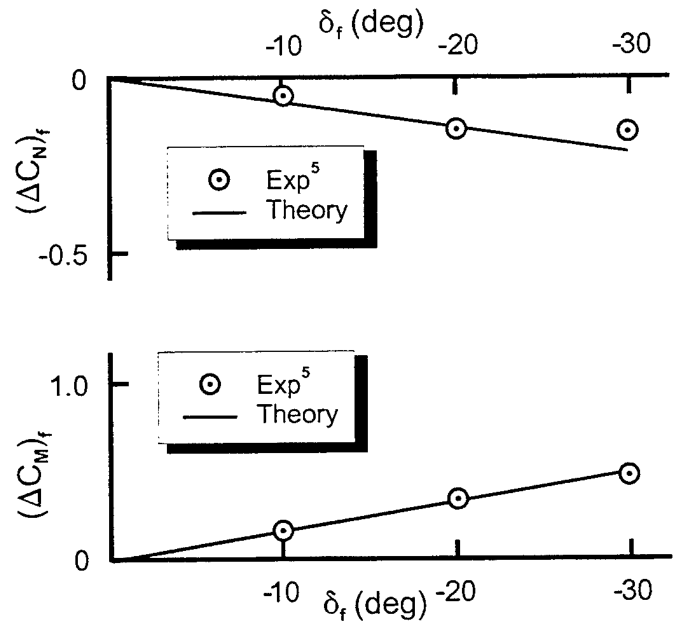


FIGURE 19. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 3.95$, $\alpha = 10$ DEG)

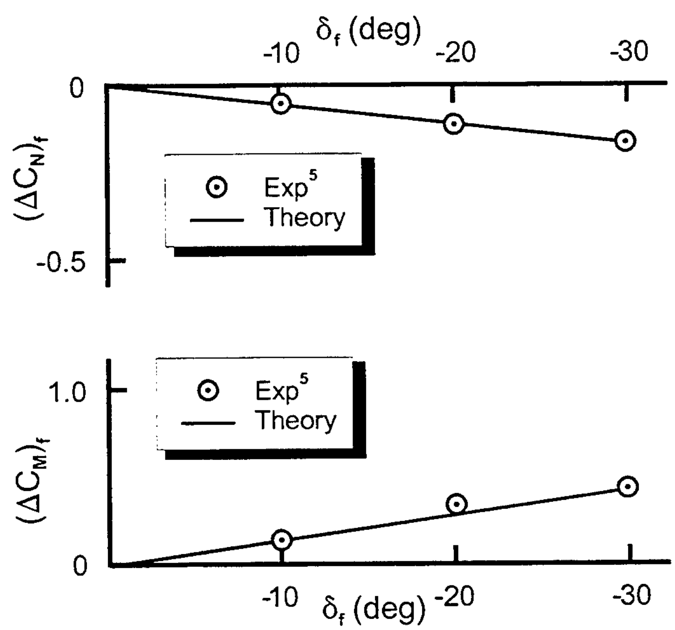


FIGURE 20. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 4.63$, $\alpha = 10$ DEG)

Reference 6 has both positive and negative values of δ_f available. Unfortunately, $M_\infty = 0.4$ was the highest freestream Mach number considered, and AOA to 30 deg were also included in the test series.

Figure 21 compares the theory and experiment for $\Delta(C_N)_f$ and $\Delta(C_M)_f$ where δ_f is negative for α to 30 deg. Note excellent agreement for $\Delta(C_N)_f$ is obtained between theory and experiment for both $\delta_f = -10$ deg and -30 deg cases. Good agreement between theory and experiment is obtained for $\Delta(C_M)_f$ for the $\delta_f = -10$ deg case up to α of about 20 to 25 deg, where the theory and experiment start to depart. For $\delta_f = -30$ deg, comparison of theory and experiment for $\Delta(C_M)_f$ is quite acceptable for α up to 20 deg. The trim AOA occurs at about 6 deg for $\delta_f = -10$ deg and at about 14.8 deg for $\delta_f = -30$ deg. In other words, good accuracy in both $\Delta(C_N)_f$ and $\Delta(C_M)_f$ can be obtained up to and slightly beyond the trim AOA, which is most critical. For α above the trim value, accuracy of $\Delta(C_N)_f$ and $\Delta(C_M)_f$ is not as important, and thus no attempt was made to try to improve the theory for these conditions.

Figure 22 gives the complimentary results to the Figure 21 case except here δ_f is positive. While trim cannot occur due to the fact α and δ_f are of the same sign and the configuration is tail controlled, it is still of interest to see how well the theory compares to data for conditions where trim is not possible. As seen in Figure 22, agreement between theory and experiment for both $(\Delta C_N)_f$ and $\Delta(C_M)_f$ is quite good up to an α of about 15 deg. Above α of 15 deg, both $\Delta(C_M)_f$ and $\Delta(C_N)_f$ deviate from experiment at most conditions. Again, since this is not a practical set of conditions for trim, no effect has been made to improve $\Delta(C_N)_f$ and $\Delta(C_M)_f$ for α above 15 deg and δ_f is positive.

Figure 23 compares the theory and experiment for axial force coefficient where the trailing edge flap has been deflected -10 deg and -30 deg respectively. The equivalent value of δ_W corresponding to $\delta_f = -10$ deg and -30 deg respectively is shown at the top of Figure 23 as a function of freestream mach number. Note that δ_W is only a small fraction of δ_f. The wing area is 8.67 times that of the trailing edge flap. At the bottom of Figure 23 is the axial force coefficient based on the AP02 calculations plus the value defined by Equation (12). Two cases are shown for the theory: where the wind tunnel model has a boundary layer trip and where no boundary layer trip is present. The Reynolds number for the tests was 2.5×10^6. According to Reference 5, a boundary layer trip was present. Based on comparison of theory and experiment, it appears the boundary layer trip was effective in producing a turbulent boundary layer over the surface at the lower supersonic Mach numbers. However, at the higher supersonic Mach numbers, it appears that the flow partially transitions back to laminar over much of the body and large wing for the $\delta = -10$ deg case. This relaminarization of the flow is speculated to be the reason the theory with no boundary layer trip option agrees closer to the wind tunnel data at high supersonic Mach number than does the theory which assumes turbulent flow over the entire surface of the model at all Mach numbers. If the above hypothesis of relaminarization of the flow is correct, the theory predicts the experimental data quite nicely. If this hypothesis is not correct, then the theory is high for Mach numbers 3.0 and greater.

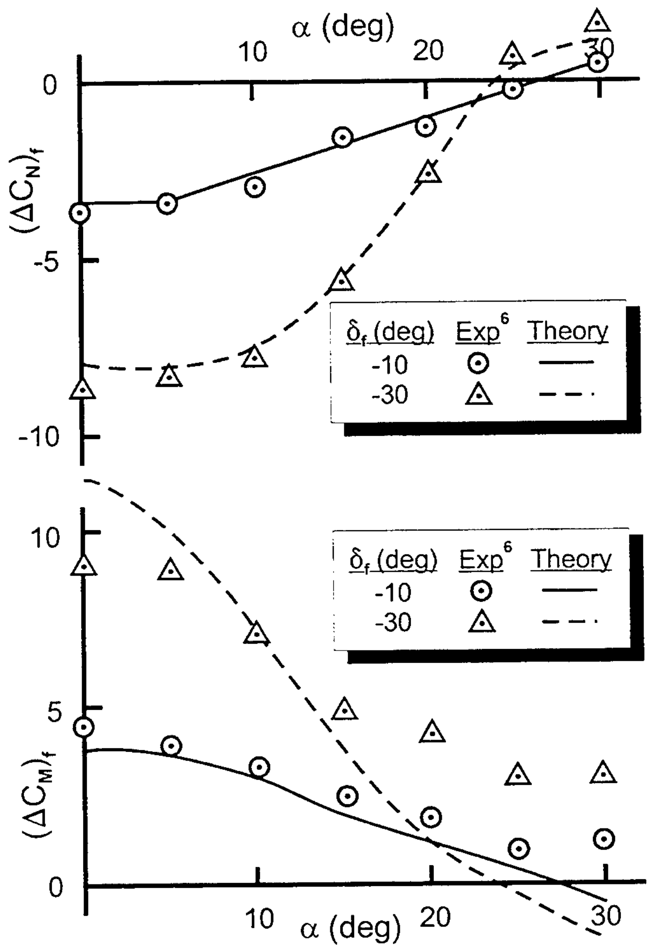


FIGURE 21. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 0.4$, δ_f NEGATIVE)

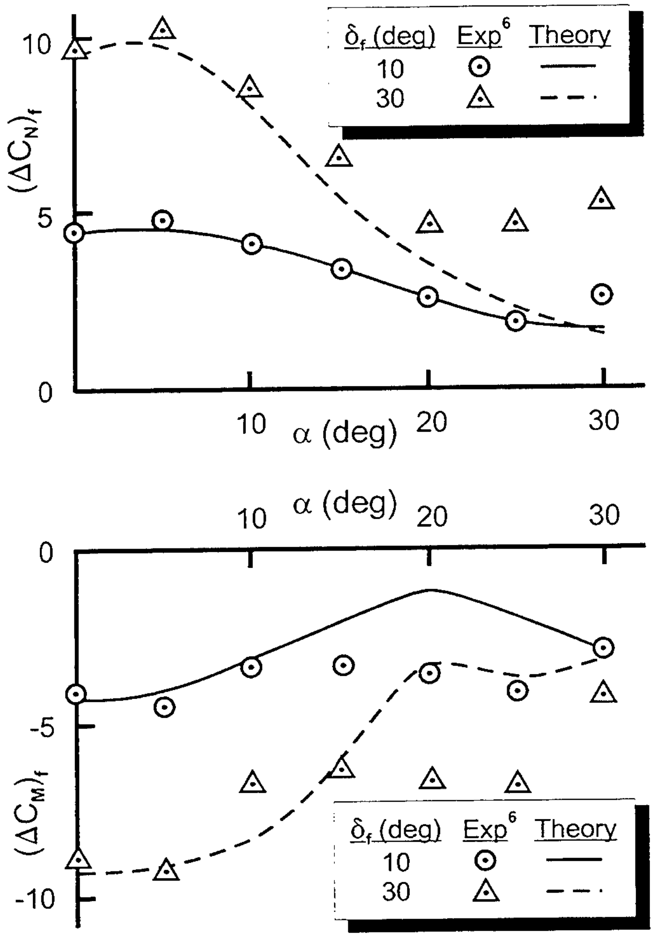


FIGURE 22. COMPARISON OF THEORY AND EXPERIMENT FOR NORMAL FORCE AND PITCHING MOMENT COEFFICIENTS OF TRAILING EDGE FLAPS ($M_\infty = 0.4$, δ_f POSITIVE)

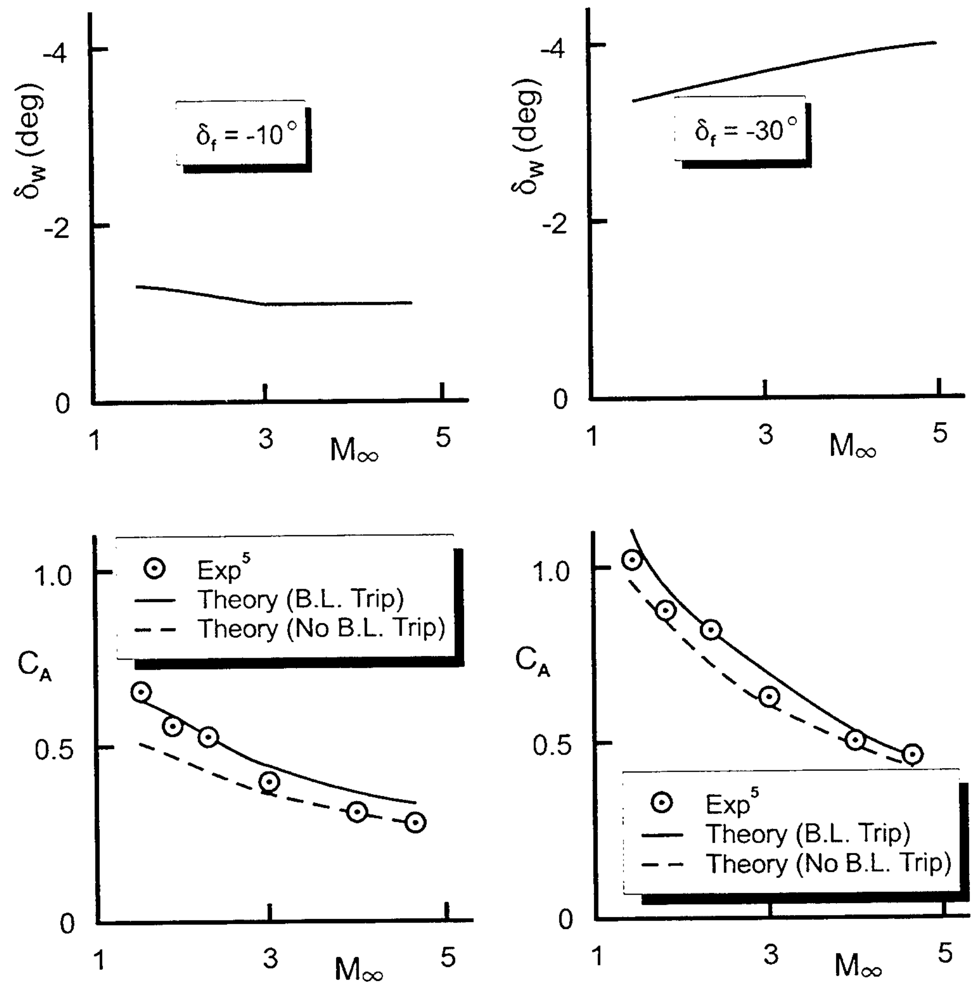


FIGURE 23. COMPARISON OF THEORY AND EXPERIMENT FOR AXIAL FORCE COEFFICIENT AT VARIOUS VALUES OF FLAP DEFLECTION AND AS REPRESENTED BY AN EQUIVALENT DEFLECTION OF ENTIRE WING AT $\alpha = 0$ DEG ($R_N/ft = 2.5 \times 10^6$)

The Reference 6 data base also contained axial force data. Unfortunately, the base drag term was subtracted out, only one fin was deflected and the numbers for no fin deflection were small and irregular. As a result, it was believed an accurate value of experimental data for the axial force would be difficult to obtain and therefore no comparisons of axial force coefficient will be shown at subsonic Mach numbers.

4.0 SUMMARY

An improved semiempirical method has been developed to estimate the static aerodynamics generated by a trailing edge flap. The method is based on deflecting the full wing or tail surface an amount that allows the normal force coefficient to be equal to that generated by the flap deflected. A transfer in pitching moments is derived to account for the difference in pitching moment when a full wing versus a trailing edge flap is deflected. Also, an approximate relationship is given which accounts for the additional axial force coefficient not accounted for based on a full wing deflected a small amount versus a trailing edge flap a larger amount.

In comparing the new semiempirical method to experimental data, the following observations were made.

1. Normal force coefficient predictions at supersonic speeds were very good except at the highest Mach numbers ($M_\infty = 4.63$) and AOA ($\alpha > 25$ deg) where the predictions were only fair.

2. Pitching moment coefficient predictions at supersonic speeds were fair to good at all conditions considered ($1.5 \le M_\infty \le 4.63$, $0 \le \alpha \le 30$, $-30 \le \delta_f \le 0$). The worst case agreement was again for $M_\infty = 4.63$ and $\alpha > 20$ deg.

3. Axial force coefficient predictions for supersonic conditions were found to be reasonable. However, the accuracy was seen to be dependent on whether the boundary layer on the wind tunnel model remained turbulent at $M_\infty \ge 2.3$ versus returning to laminar flow over the model.

4. At subsonic flow, the only data available to the author was at $M_\infty = 0.4$. For this Mach number, it was found the predictions for both normal force and pitching moment coefficients were acceptable up to and slightly past the trim AOA. For larger flap deflections, the accuracy of the predictions was acceptable at AOAs that exceeded trim conditions by about 5 deg. However, since trim and slightly past trim are of the most practical interest, this problem was not seen as a major limitation.

A linear interpolation of the empirical factors used in the derivation process was assumed between Mach numbers of 0.8 and 1.5. Also values of these factors were assumed to be constant below $M_\infty = 0.4$ and above $M_\infty = 4.63$, where no data was available.

Additional wind tunnel data is needed to refine and validate the new semiempirical model. Specifically, data is needed when the AOA and flap deflection are of the same sign at supersonic speeds. Data is needed for Mach numbers between 0.4 and 1.5 as well. However, until additional data becomes available, the model derived here uses engineering judgment to fill in these gaps and allows the model to be operational over the practical AOA, Mach number and control deflection range that trailing edge flaps are contemplated for use.

5.0 REFERENCES

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6.0 SYMBOLS AND DEFINITIONS

AOA	Angle of Attack
APC	Aeroprediction Code
AP02, AP98	2002 and 1998 versions of the APC respectively
AR	Aspect Ratio = b^2/A_W
NASA/LRC	National Aeronautics and Space Administration/Langley Research Center
NSWCDD	Naval Surface Warfare Center, Dahlgren Division
A_{REF}	Reference area (maximum cross-sectional area of body, if a body is present, or planform area of wing, if wing along)(ft^2)
A_W	Planform area of wing in crossflow plane (ft^2)
b	Wing span (not including body)(ft)
C_A	Axial force coefficient
$(\Delta C_A)_f, (\Delta C_N)_f, (\Delta C_M)_f$	Change in axial, normal and pitching moment coefficients respectively due to a flap deflection δ_f
C_M	Pitching moment coefficient (based on reference area and body diameter, if body present, or mean aerodynamic chord, if wing alone)
C_N	Normal-force coefficient
$C_{N_{B(W)}}, C_{N_{B(T)}}$	Normal-force coefficient on body in presence of wing or tail
$(C_{N_\alpha})_W, (C_{N_\alpha})_T$	Normal-force coefficient slope of wing and tail respectively
C_{N_W}	Normal-force coefficient of wing alone

$C_{N_{W(B)}}, C_{N_{T(B)}}$	Normal-force coefficient of wing or tail in presence of body
C_{N_α}	Normal-force coefficient derivative
c_r	Rood chord (ft)
c_{r_w}, c_{r_f}	Root chord of wing and flap respectively (ft)
c_t	Tip chord (ft)
cal	Caliber(s) (one body diameter)
d_B	Body diameter (ft) at base
d_{ref}	Reference body diameter (ft)
deg	Degree(s)
f_1, f_2, f_3	Empirical factors used in defining the semiempirical model for flap aerodynamics
$k_{B(W)}, k_{B(T)}$	Ratio of additional body normal-force coefficient due to presence of wing or tail at a control deflection to that of wing or tail alone at $\alpha = 0$ deg
$k_{W(B)}, k_{T(B)}$	Ratio of wing or tail normal-force coefficient in presence of body due to a control deflection to that of wing or tail alone at $\alpha = 0$ deg
ℓ, ℓ_n	Total length and nose length respectively (ft)
ℓ_{ref}	Reference length (ft)
$M_{W(B)}, M_{B(W)}$	Moment of wing in presence of body and body in presence of wing respectively (ft – lb)
M_∞	Freestream Mach number
$N_{W(B)}, N_{B(W)}$	Normal force of wing in presence of body and body in presence of wing respectively (lb)
N_f	Normal force of trailing edge flap (lb)
p	Pressure (lb/ft^2)
Q	Dynamic pressure (lb/ft^2)

r	Local body radius (ft)
r_{LE}, r_{TE}	Radius of leading and trailing edges of fin respectively (ft)
r_n	Nose radius (ft)
R_N	Reynolds number
t	Fin thickness (ft)
V_∞	Freestream velocity (ft/sec)
X_{LE}, X_{CG}	Distance from nose tip to wing leading edge or center of gravity of body respectively (ft)
X_{CP}	Center of pressure (in feet or calibers from some reference point that can be specified) in x direction
X_{ref}	Reference location along x axis for moments (ft)
x,y,z	Axis system fixed with x along centerline of body
α	Angle of attack (deg)
α_{TR}	Trim angle of attack (deg)
Λ_{LE}	Leading edge sweepback angle of fin (deg)
δ_f	Control deflection (deg) of trailing edge flap, positive leading edge up
δ_W, δ_T	Deflection of wing or tail surfaces (deg), positive leading edge up
Φ	Roll position of missile fins ($\Phi = 0$ deg corresponds to fins in the plus (+) orientation). $\Phi = 45$ deg corresponds to fins rolled to the cross (×) orientation
λ	Taper ratio of a lifting surface = c_t/c_r

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