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VORTEX MANEUVER LIFT FOR SUPER-CRUISE CONFIGURATIONS

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16. Abstract

Research is being conducted to determine the vortex maneuver-lift characteristics for high performance fighter aircraft. The generation of vortex lift for maneuver of super-cruise aircraft may be particularly important if this supersonic aircraft is to have maneuver performance similar to current subsonic-transonic fighters. This paper reviews some of the theoretical and experimental research conducted at the NASA Langley Research Center to investigate the subsonic vortex-lift producing capabilities for two classes of Super-Cruise designs: a close-coupled wing-canard arrangement and a slender wing configuration. In addition, several analytical methods are discussed for estimating critical structural design loads for thin, highly swept wings having separated leading-edge vortex flows.

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INTRODUCTION

Other papers in this conference have discussed some aerodynamic design philosophies used to develop Super Cruise fighter aircraft (references 1 to 3). These studies demonstrate that Super Cruise concepts are characterized by thin wings with high sweepback angles. Wings of this type produce separation-induced vortex flows at angles of attack corresponding to maneuver conditions. The vortex lift forces associated with these flow conditions may be particularly important for the Super Cruise aircraft if it is to have maneuver characteristics similar to those of current subsonic-transonic fighters. Therefore, this paper reviews some theoretical and experimental research conducted at the NASA Langley Research Center for the purpose of investigating the subsonic vortex-lift producing capabilities for two classes of Super Cruise designs: close-coupled wing-canard configurations and a slender wing configuration. (See figure 1) In addition, several analytical methods are discussed for estimating critical structural design loads for thin, highly swept wings having separated leading-edge vortex flows.

SEPARATION-INDUCED VORTEX FLOWS

Prior to the data presentations for the Super Cruise configurations, it is desirable to comment on separation-induced vortex flows and vortex-lift technology. Figure 2 presents turn rate, a measure of maneuverability, as a function of Mach number and illustrates typical maneuver boundaries for a fighter aircraft. The limit denoted by $C_{L\max}$ is an aerodynamic boundary at subsonic and transonic speeds and reflects the maximum usable lift for the fighter. This includes buffet onset as well as stability limitations on maneuver performance. There are many aerodynamic concepts which can help improve the maneuver capability, among them is the utilization of the important leading-edge vortex flows. Wings which utilize leading-edge vortex flows to achieve high maneuver lifts can have low structural weights and design simplicity compared with other high-lift approaches; while some drag penalty can be encountered for flat wings with vortex flows, this drag penalty can be essentially eliminated by combining wing camber with the leading-edge vortex.

Figure 3 shows a photograph of an F-16 to illustrate one type of vortex system which exists in a current aircraft design. The photograph shows that the flow separates from the wing's leading-edge extensions, or strakes, and forms a discrete pair of vortices which pass over the wing. This vortex system produces sizeable increments of vortex lift at maneuver attitudes. A similar situation can exist with thin, highly swept wings, as shown in figure 4. This figure shows the effect of Mach number on the lift coefficient obtained for a 74° delta wing at $\alpha = 16^{\circ}$. The estimated vortex lift contribution is seen to be relatively constant through the

Mach number range. This apparent insensitivity to Mach number is important because the experimental data on the Super Cruise configurations were obtained at low Mach numbers of 0.2 and 0.3. Based on this information, the vortex-lift trends obtained for the Super Cruise configurations at low Mach numbers should be essentially the same at high subsonic and transonic speeds. These studies will be extended to transonic speeds in future wind-tunnel tests.

The theory used to calculate the vortex lift for the simple delta wing (in figure 4) is referred to as the Leading Edge Suction Analogy (ref. 4). As seen in figure 5, the theory has been extended to account for leading-edge and side-edge vortex flows for a variety of configurations and Mach numbers. These efforts are providing improved aerodynamic design and analysis theories (ref. 5) which account for separation-induced vortex flows associated with high performance aircraft. This theory is used throughout the remainder of the paper to compare, where it is possible, with the Super Cruise experimental results.

CLOSE-COUPLED WING-CANARD CONFIGURATION

The first Super-Cruise configuration to be examined utilizes the close-coupled wing-canard arrangement presented in reference 6. Lift coefficient is shown in figure 6 as a function of α and illustrates the synergetic effect which occurs for this configuration. The square symbols represent the data for the basic wing body, and the circle symbols represent the total configuration, i.e., the canard, wing, and body. It is obvious that adding the canard results in large lift gains. A better understanding of how these lift benefits are achieved can be obtained by examining the data represented by the diamond symbol. These data were obtained by adding the lift loads on the canard and forebody, not in the presence of the wing, to the loads on the wing and aft body, not in the presence of the canard. It was possible to do this because the data were obtained using two strain gage balances, one located in the model's aft section to measure total configuration loads and the other located in the model's forward section to measure only canard loads. Comparing the circle and diamond data at high angles-of-attack shows that the lifts on the complete configuration are much higher than those obtained by adding the two parts. This synergetic effect is due to the beneficial interference which results when the wing and canard are in close proximity. In particular, the canard's presence helps control the leading edge vortex that forms on the wing, thus promoting vortex lift. The vortex flow theoretical predictions shown on figure 6 assume that the wing develops full vortex lift. This assumption is borne out by the good agreement with the data.

The drag polars for the three configurations of figure 6 are presented in figure 7. The synergistic effect is also noted here, where the favorable interference produces improvements in the drag polar. These trends are predicted reasonably well by the attached flow theory (no leading-edge suction) and the vortex flow theory. A third analytical curve is presented to indicate the theoretical minimum drag achievable for attached flow and full leading-edge suction. This was not possible for the current data because the wing and canard had sharp leading edges and no twist or camber.

Figures 8 and 9 show the effect on C_L and on drag- due-to-lift of adding wing twist and camber to this wing-canard design. It should be noted that the data represented by the triangle and diamond symbols are for a wing design C_L of 0 and are the same data that are presented in figure 7. It is seen in figure 8 that the increase in lift obtained at $\alpha = 0^o$ by increasing the wing design C_L is generally maintained throughout the angle-of-attack range. In addition, the stall angle of attack remained approximately the same for the three configurations with the canard on. These data, therefore, suggest that vortex lift exists on all three wing-canard configurations. It is noted that wing design C_L refers to the design condition of the isolated wing, i.e., not in the presence of the canard.

The data in figure 9 show the effect of adding the canard, which, as has already been shown, results in reductions in induced drag and significant gains in lift. Further reductions in induced drag are obtained by increasing the wing design C_L to 0.35 and 0.70. In fact, this change in design C_L leads to induced drag levels which approach the theoretical minimum represented by $1/\pi A$. Reasonably low drag values are maintained well above the design lift coefficient suggesting some leading-edge thrust recovery as the leading-edge vortex reduces the pressure on the cambered leading edge.

SLENDER WING CONFIGURATION

This portion of the paper presents some unpublished data recently obtained by Washburn (ref. 7) for a slender wing Super-Cruise configuration. A sketch of the configuration is shown in figure 10. The configuration is similar to one of the Rockwell Super-Cruise designs reviewed by Shrout, et.al, (ref. 1) and Goebel, et.al., (ref. 3). The major differences between the NASA and Rockwell configurations occur on the wings outer panel. The planform geometry for the NASA model was defined using the theoretical leading-edge suction distribution shown on the right side of the figure. The design goal was to maintain a constant suction coefficient as far outboard as possible; a constant C_S distribution tends to two-dimensionalize the flow over the wing and provide full benefits of the sweep effect for cruise efficiency. This criterion resulted in a minimum angle (48°) on the outer wing panel which is lower than that on the Rockwell version.

The purpose of this study was to evaluate the effect of wing tip dihedral on the off-design aerodynamic performance at high angles-of-attack. In order to evaluate the vortex lift effects, the wing had no camber or twist and had symmetrically beveled leading edges. This provides aerodynamic characteristics at high angles-of-attack which are dominated by the forces associated with the separation-induced vortex flows.

The lift characteristics of the configuration with various wing tip geometries are shown in figure 11. The anhedral wing tips result in slightly higher lifts than the flat tips, which, in turn, have higher lifts than the dihedral wing tips. Comparing the data with the two theories that are shown, attached flow representing no vortex lift and vortex flow representing full vortex lift, it is seen that this slender wing configuration does develop some vortex lift. In fact, at $\alpha = 28^{\circ}$ the anhedral configuration achieves about 67 percent of the vortex lift theoretically available, while the dihedral configuration achieves about 46 percent. The loss of lift from the estimated full vortex-lift level is due partially to the trailing-edge notch effect (ref. 5), to early vortex breakdown on the outer panel (ref. 7), and to differences between the vortex system's behavior and the behavior assumed by the theory.

Oil flows patterns were obtained on the upper surface of the wing and are shown in figure 12 to illustrate the type of vortex system developed by this planform. These photographs were obtained for the configuration with dihedral wing tips and are shown for angles of attack of 8°, 16°, and 24°. A well-defined flow pattern can be seen on the inner portion of the planform which is indicative of a pair of strong vortices formed from the highly-swept portion of the model. At the lower angles-of-attack, an additional vortex system is evident on the lower-swept outer panel. Increasing the angle of attack results in breakdown of the outer panel vortex, which causes a subsequent loss of lift. This is, at least partly, a consequence of the planform being designed for constant leading-edge suction distribution for cruise considerations. Recalling the lift data, it would appear that the anhedral wing tips (tips down) provide a favorable pressure gradient on the upper surface which promotes the stability of the outer panel vortex. However, the dihedral tips (tips up) provide an adverse gradient which results in premature vortex breakdown. Wentz (ref. 8) measured the vortex breakdown characteristics on a 48° delta wing and showed that vortex breakdown at the trailing edge occurred at $\alpha = 10^{\circ}$.

Figure 13 shows the effect of wing tip geometry on induced drag and, as noted, the anhedral wing tips result in a lower induced drag in comparison to the other wing tip geometries. It is recalled that for sharp leading-edged wings, induced drag is equal to $C_L \tan \alpha$. With this in mind, the drag results are consistent with the lift results presented in figure 11.

The attached flow theory will full leading-edge suction is presented for reference. The effects of wing tip geometry on the pitching moment are shown in figure 14. The center of gravity location is similar to that used by Rockwell in their design studies and results in an unstable vehicle with about a 7 percent static margin for the test Mach number of 0.2. The data show that at the higher lift coefficient, the wing with the dihedral tips has the largest C_m, and with the anhedral tips has the lowest C_m. In addition, pitchup occurs at a C_L of about 0.3 for the configuration with flat or dihedral wing tips, while the use of anhedral tends to delay pitchup to a C_L of about 0.5. The disagreement between these measured data and the theories is because the theories estimate more loading on the aft portion of the configuration than is actually present.

CRITICAL STRUCTURAL DESIGN LOADS

In the design of advanced aircraft, the effects of separation-induced vortex flows are becoming of increased importance, and adequate theoretical methods of accounting for these flows are urgently needed. While this capability is desired even for cruise point design (ref. 10), there are at least three other design areas where the separation-induced vortex flows play a predominant roll: critical wing structural design loads, vortex lift for takeoff and landing, and vortex lift for high-speed maneuverability.

Since Super Cruise configurations have thin highly-swept wings, the wing flow field at the critical structural design load conditions is usually characterized by leading-edge vortex flows. The importance of these flows is illustrated in figure 15 which shows chordwise pressure distributions on the upper and lower wing surfaces at the 80 percent semispan location. The results are for $C_L \approx .2$ and $M = .85$. Theoretical estimates were made using the Flexstab computer program and Boeing's TEA-230 program. As noted, neither theory adequately estimates the pressures. This is because the theories assume attached flow conditions, but the upper surface pressure distribution indicates that a leading edge vortex flow has formed at this location on the wing.

Recently, Boeing has developed, under contract with Langley Research Center, a Free Vortex-Sheet Method for calculating the pressure field on wings having leading edge vortex flows (ref. 9). The theory is applied to a highly swept delta wing at $\alpha = 14^{\circ}$ in figure 16. The solid line represents the theoretical estimates using the Boeing method and are in good agreement with the data. Two other theories are shown at the most rearward location on the wing to indicate how the suction pressures have been overpredicted previously.

FUTURE STUDIES

Figure 17 shows some of the future studies that are planned for close-coupled wing-canard and slender-wing configurations. The current studies will be extended to high subsonic speeds using the Langley 7 x 10-foot wind tunnel. In particular, it is desirable to verify that the vortex lift trends established for low subsonic Mach numbers are similar at transonic maneuver conditions. In addition, more sophisticated wind tunnel models will be used to determine the optimum combinations of attached and vortex flow conditions. One concept that will be tested consists of a rounded leading edge for low drag cruise performance with a device that provides an "effectively" sharp leading edge for generating vortex maneuver lifts. The dynamic stability characteristics of these configurations will be studied at high angles-of-attack, and parametric pressure tests will be conducted to help verify the Free Vortex-Sheet Method for estimating pressures on wings having separation-induced vortex flows.

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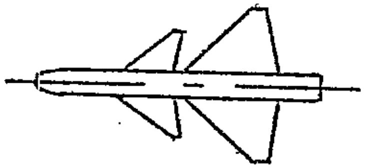
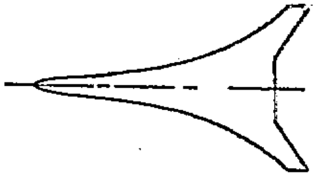
OBJECTIVES

- Experimental and Theoretical Studies of Vortex Lift at Subsonic Maneuver Conditions for:
    - Close - Coupled Wing - Canard Configurations



    - Slender Wing Configuration



- Analytical Methods for Estimating Critical Design Loads

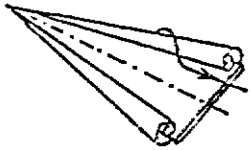


Figure 1

MANEUVER BOUNDARIES OF TYPICAL FIGHTER AIRCRAFT

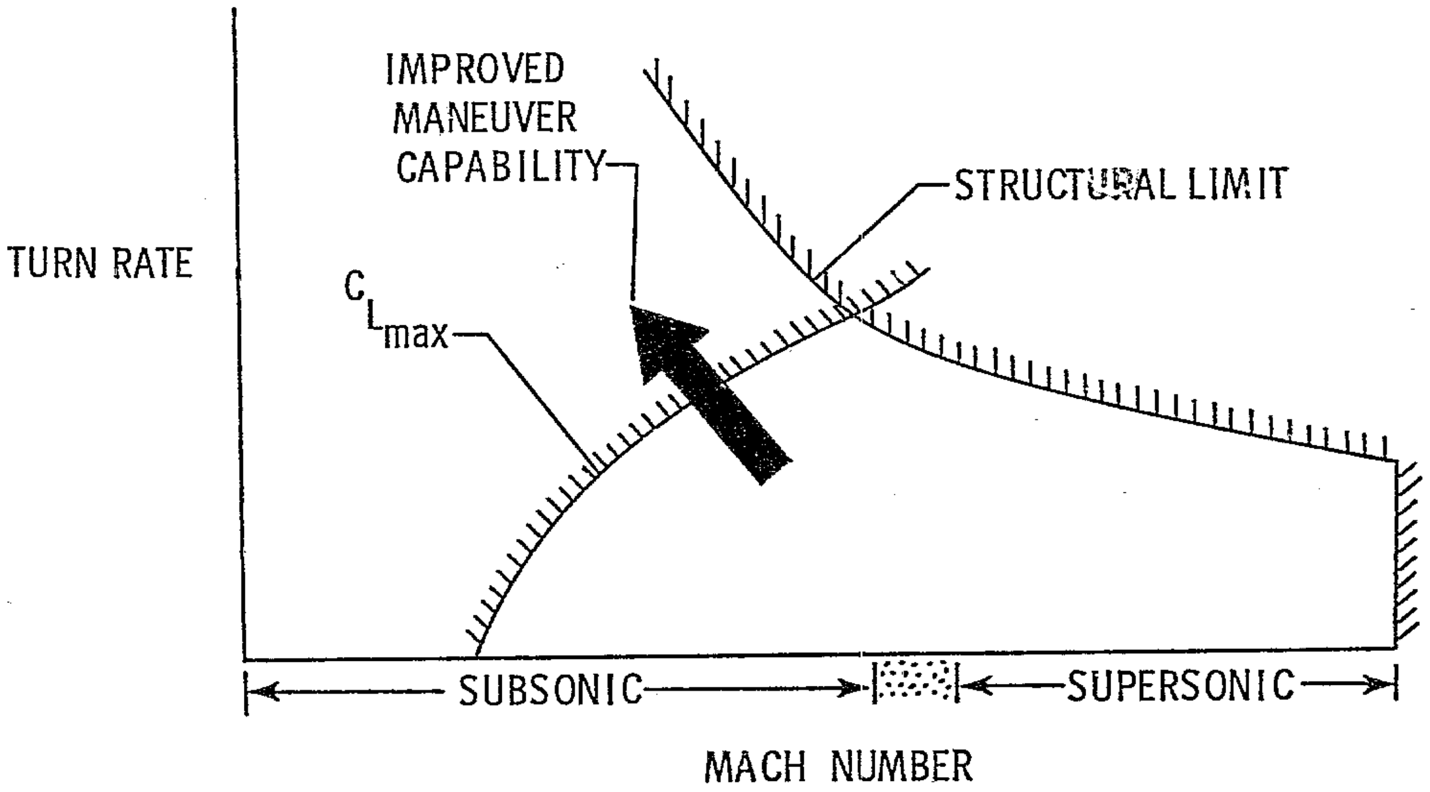


Figure 2

F-16 STRAKE VORTEX
(Aviation Week and Space Technology)

Figure 3

EFFECT OF MACH NUMBER ON VORTEX LIFT FOR A 74° DELTA WING

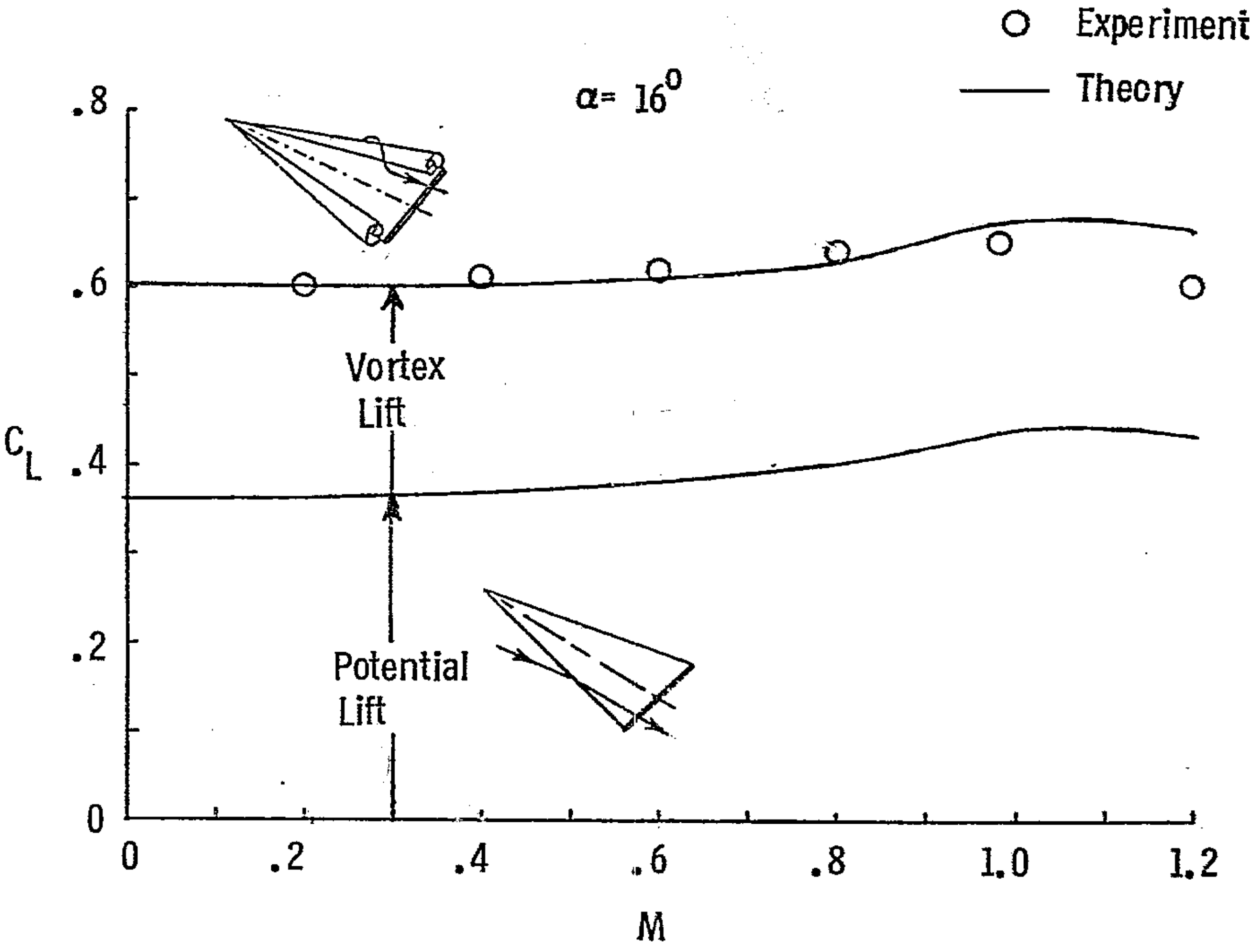
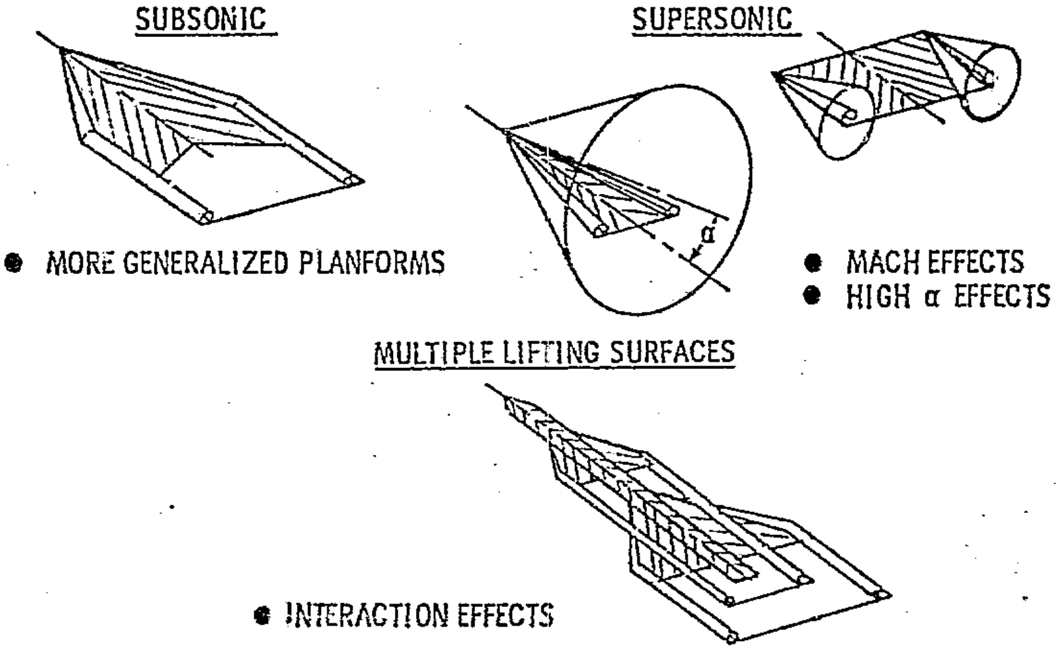


Figure 4

3-D SEPARATED FLOW THEORIES

- VORTEX FLOW MANEUVER LIFT
- CRITICAL WING DESIGN LOADS



SUBSONIC

- MORE GENERALIZED PLANFORMS

SUPERSONIC

- MACH EFFECTS
- HIGH α EFFECTS

MULTIPLE LIFTING SURFACES

- INTERACTION EFFECTS

Figure 5

CLOSE - COUPLED WING-CANARD . . .

AN EXAMPLE OF AERODYNAMIC SYNERGISM

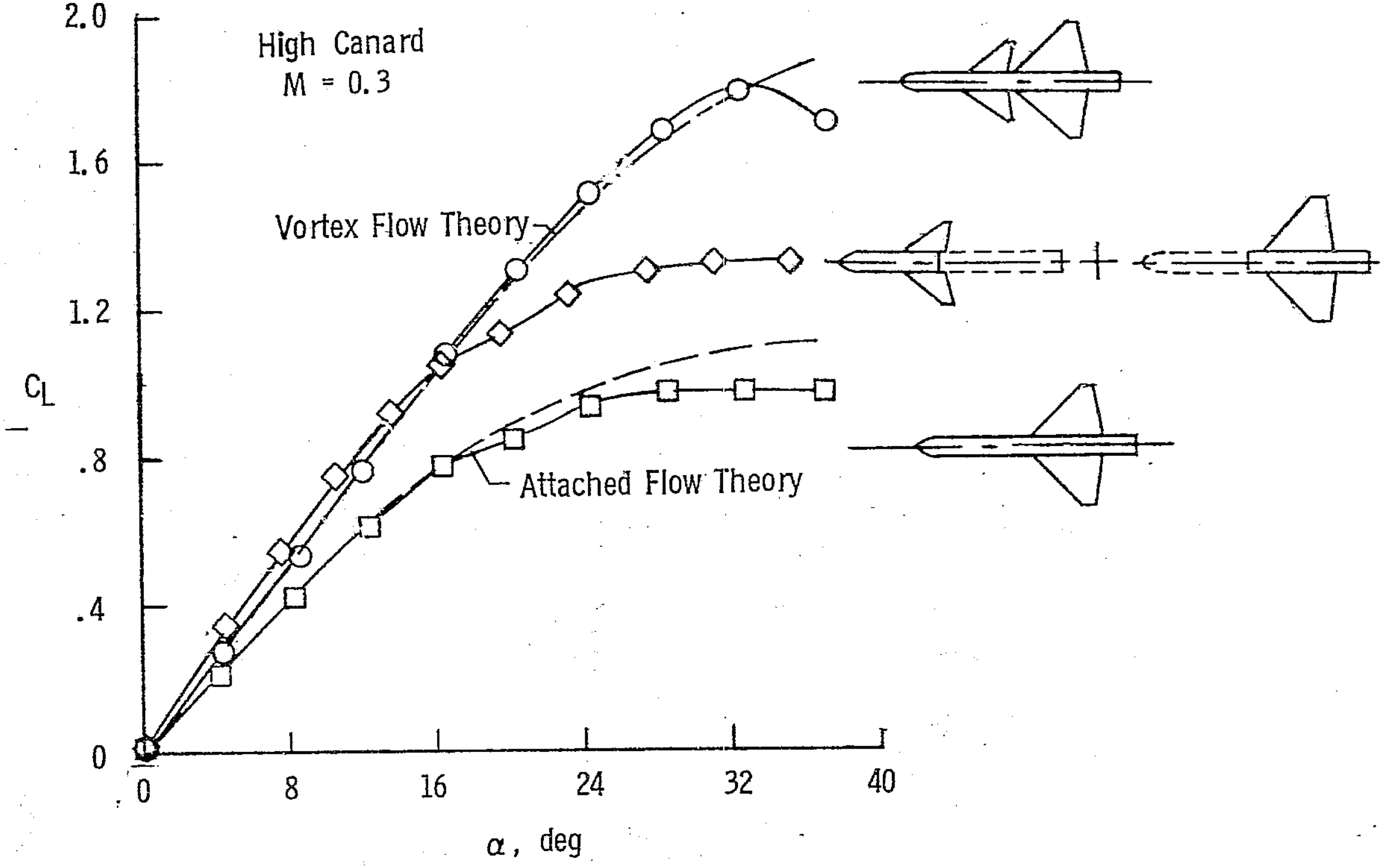


Figure 6

CLOSE - COUPLED WING-CANARD . . .
AN EXAMPLE OF AERODYNAMIC SYNERGISM

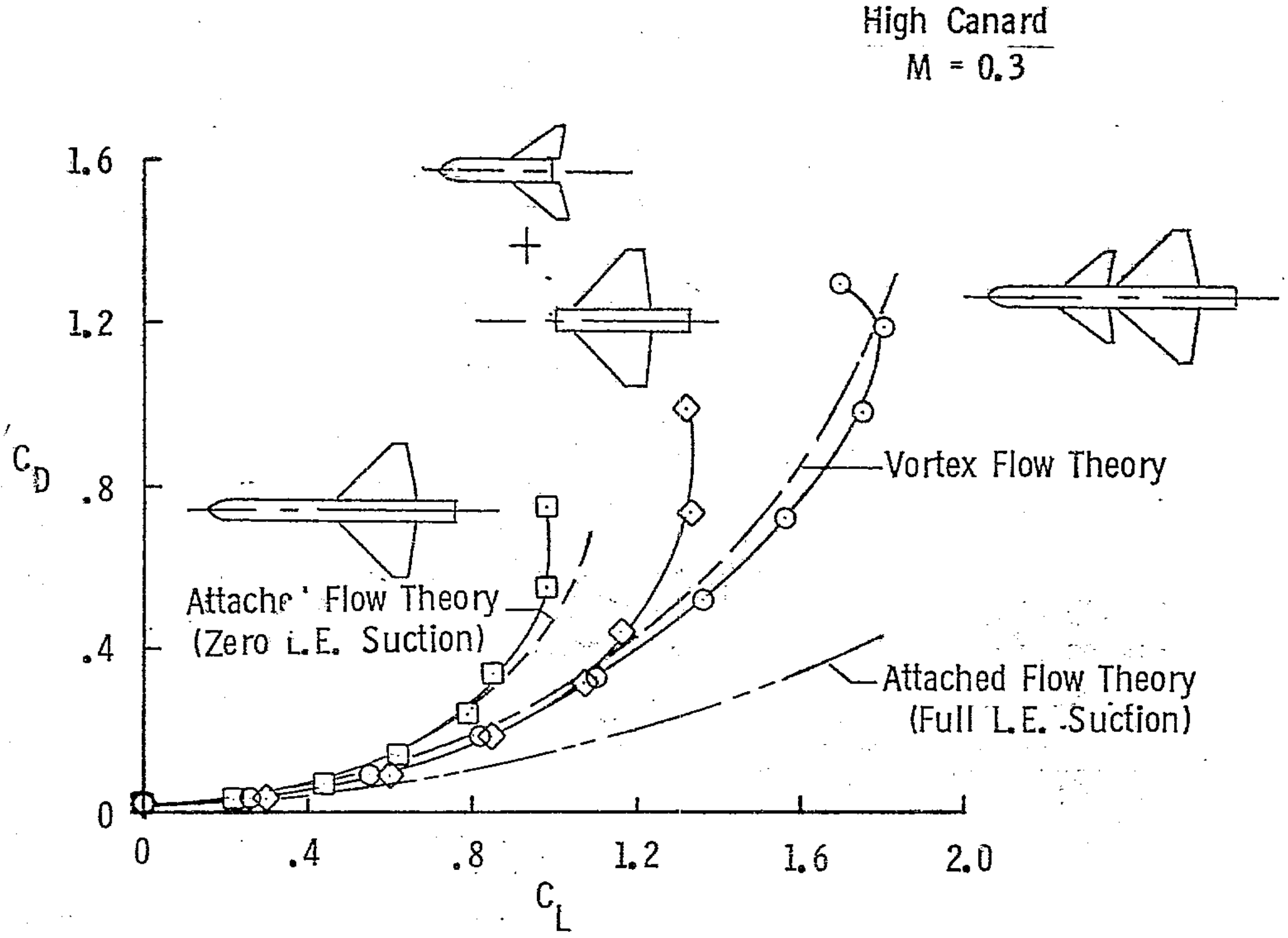


Figure 7

WING DESIGN C_L

M=0.3

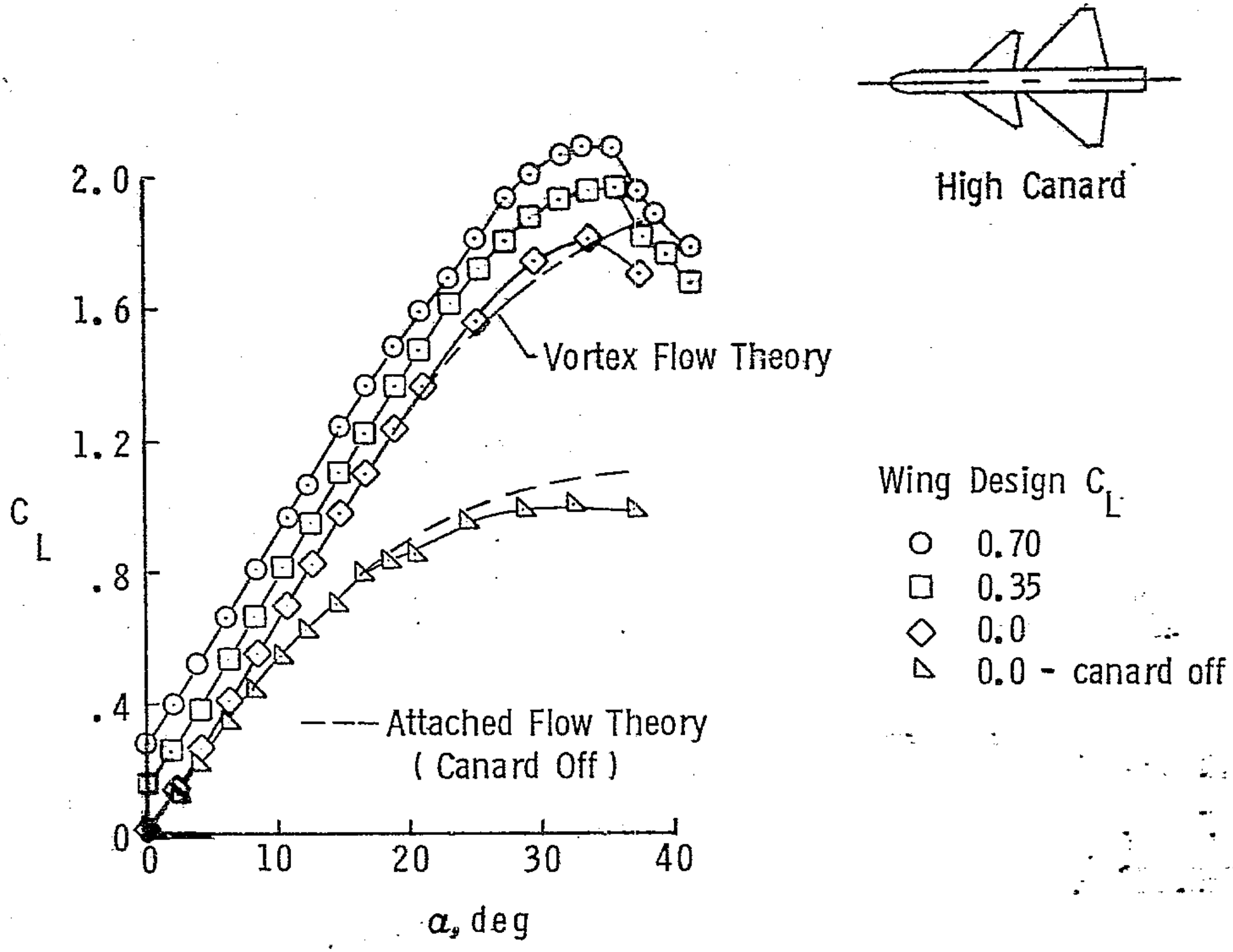


Figure 8

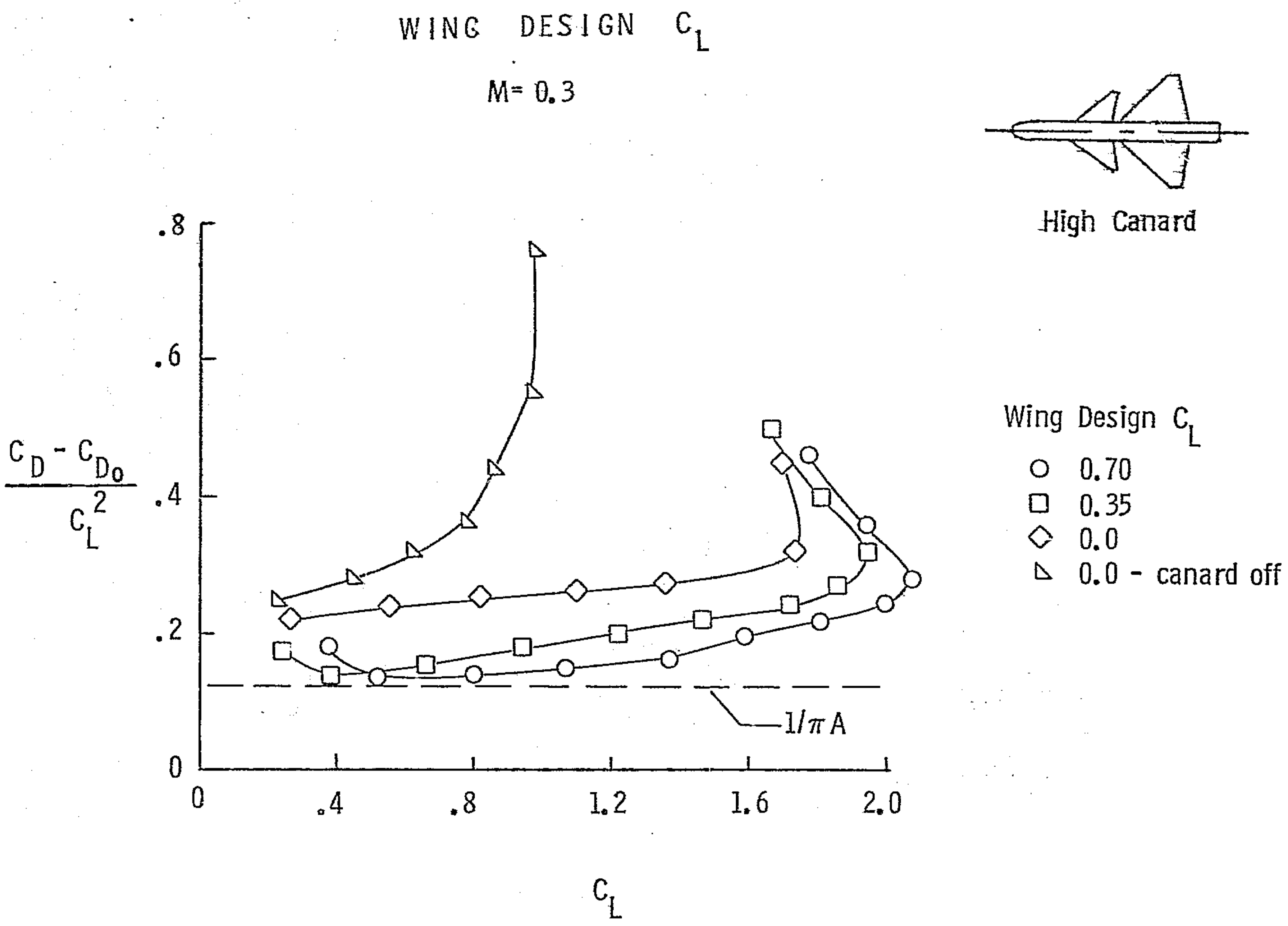
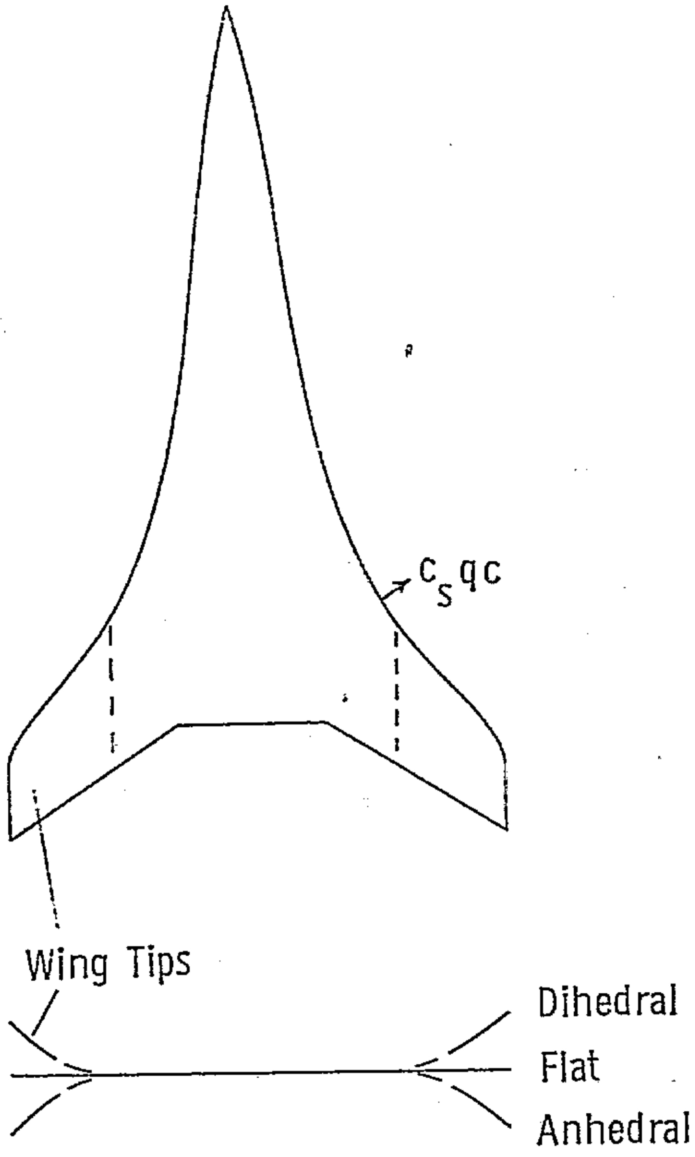

WING DESIGN C_L
M= 0.3
High Canard
$\frac{C_D - C_{D_o}}{C_L^2}$
.8
.6
.4
.2
0
0
.4
.8
1.2
1.6
2.0
C_L
$1/\pi A$
Wing Design C_L
○ 0.70
□ 0.35
◇ 0.0
◺ 0.0 - canard off

Figure 9

SUPER - CRUISE CONFIGURATION STUDY

- Slender Wing -



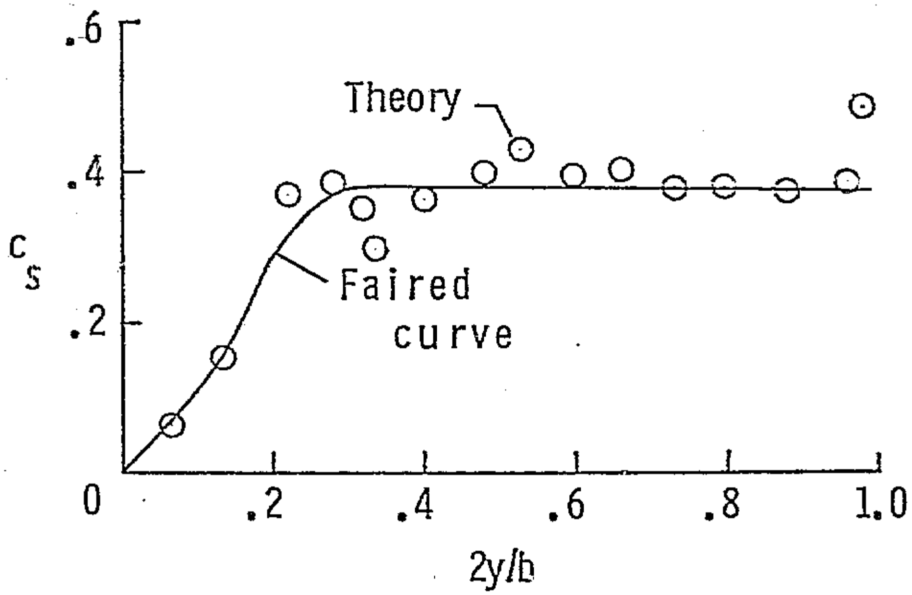


Figure 10

MANEUVER AERODYNAMICS
M = 0.2

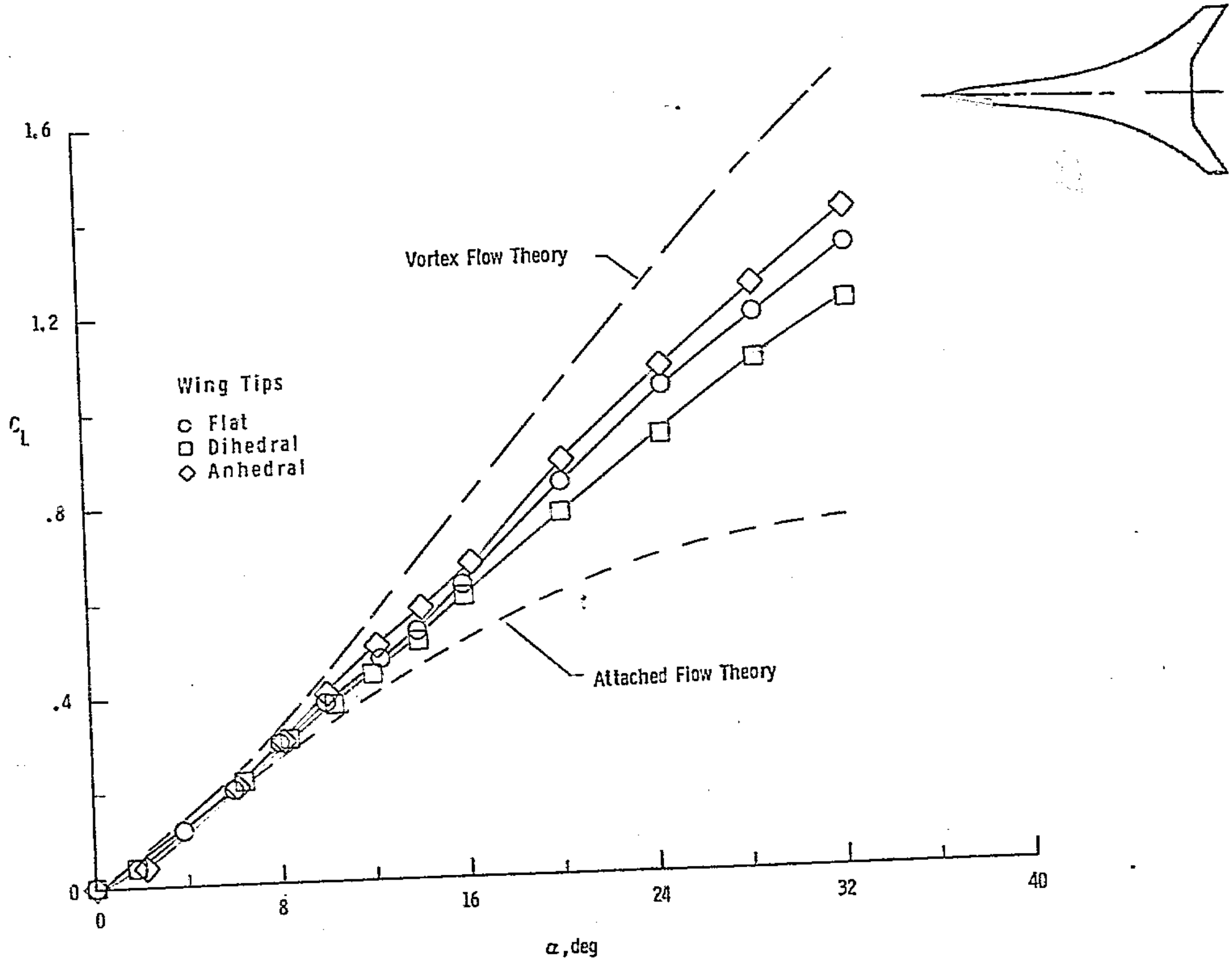
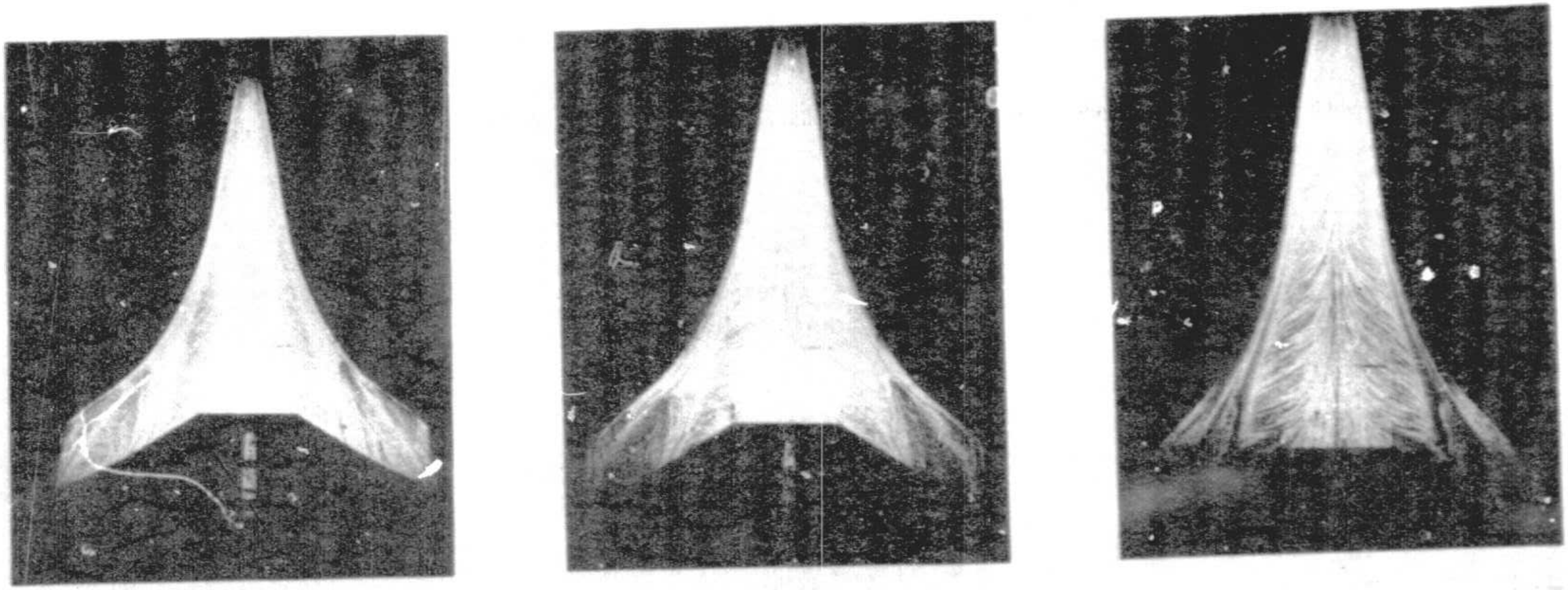


Figure 11

SURFACE FLOW PATTERNS FOR CONFIGURATION WITH DIHEDRAL



$\alpha = 8^\circ$

$\alpha = 16^\circ$

$\alpha = 24^\circ$

Figure 12

MANEUVER AERODYNAMICS
M = 0.2

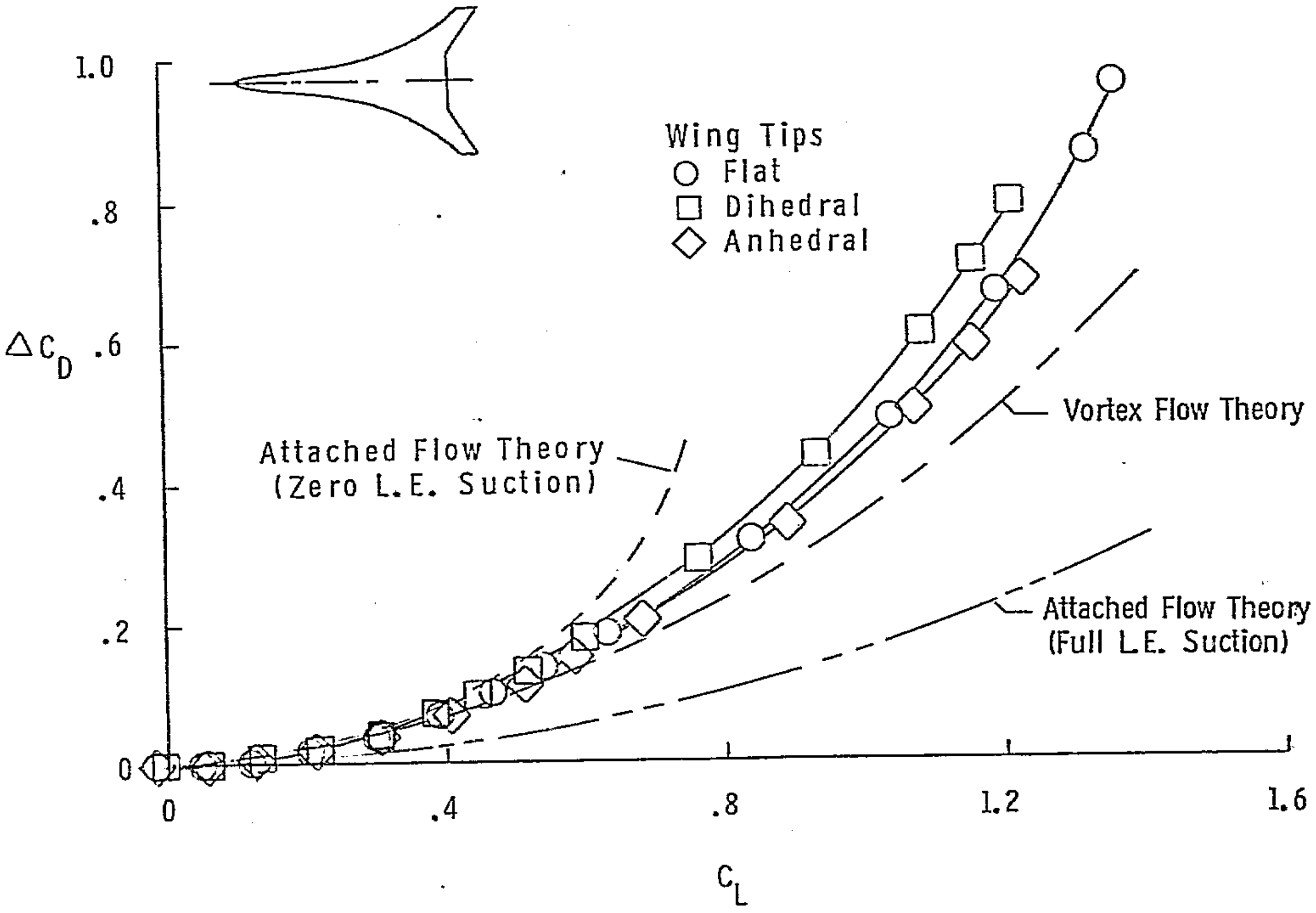


Figure 13

MANEUVER AERODYNAMICS
$M = 0.2$

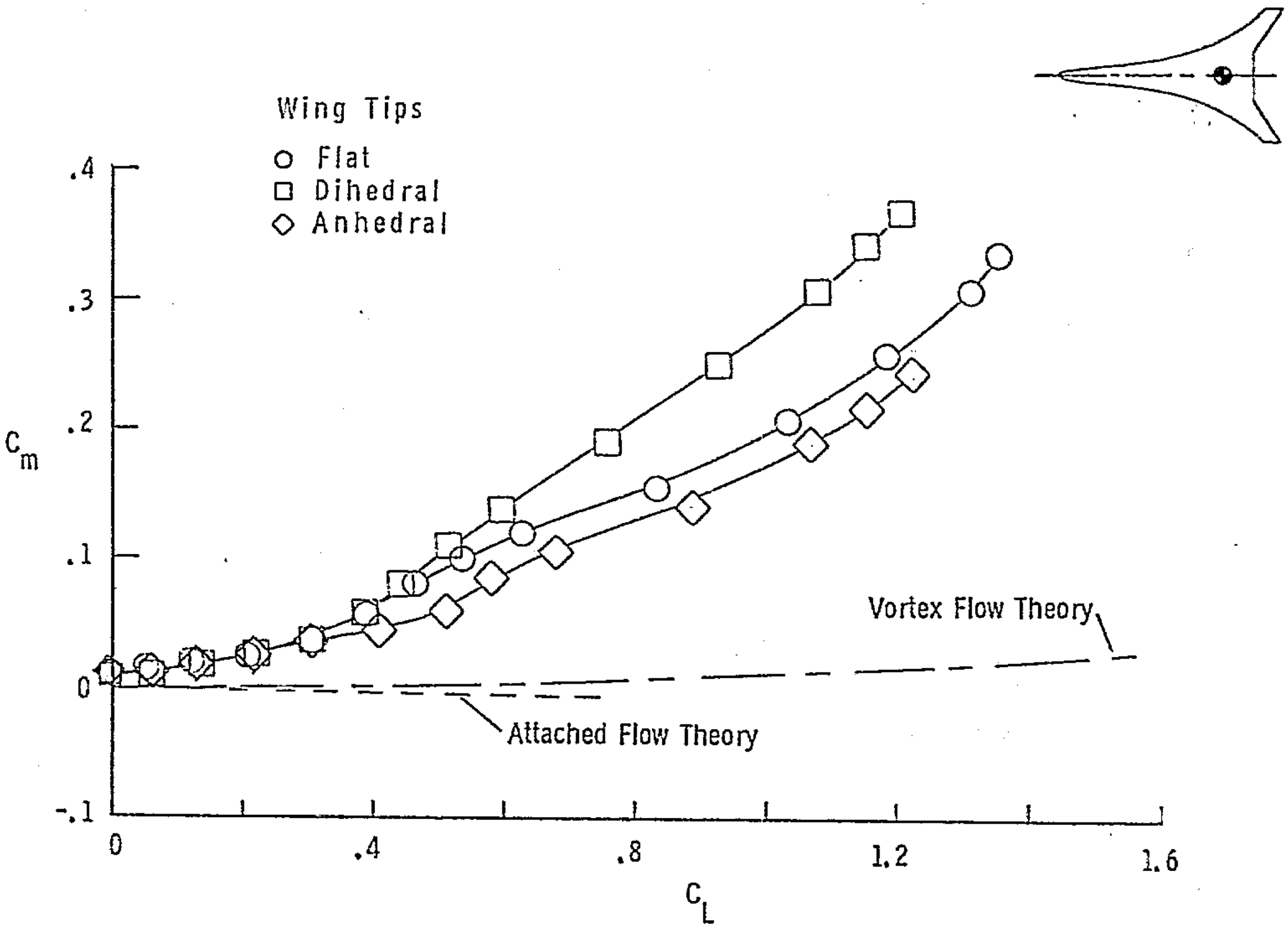


Figure 14

WING PRESSURE DISTRIBUTION FOR CRITICAL DESIGN LOADS

$\alpha = 8^0$, $M = 0.85$, $\overline{C_L \approx 0.2}$

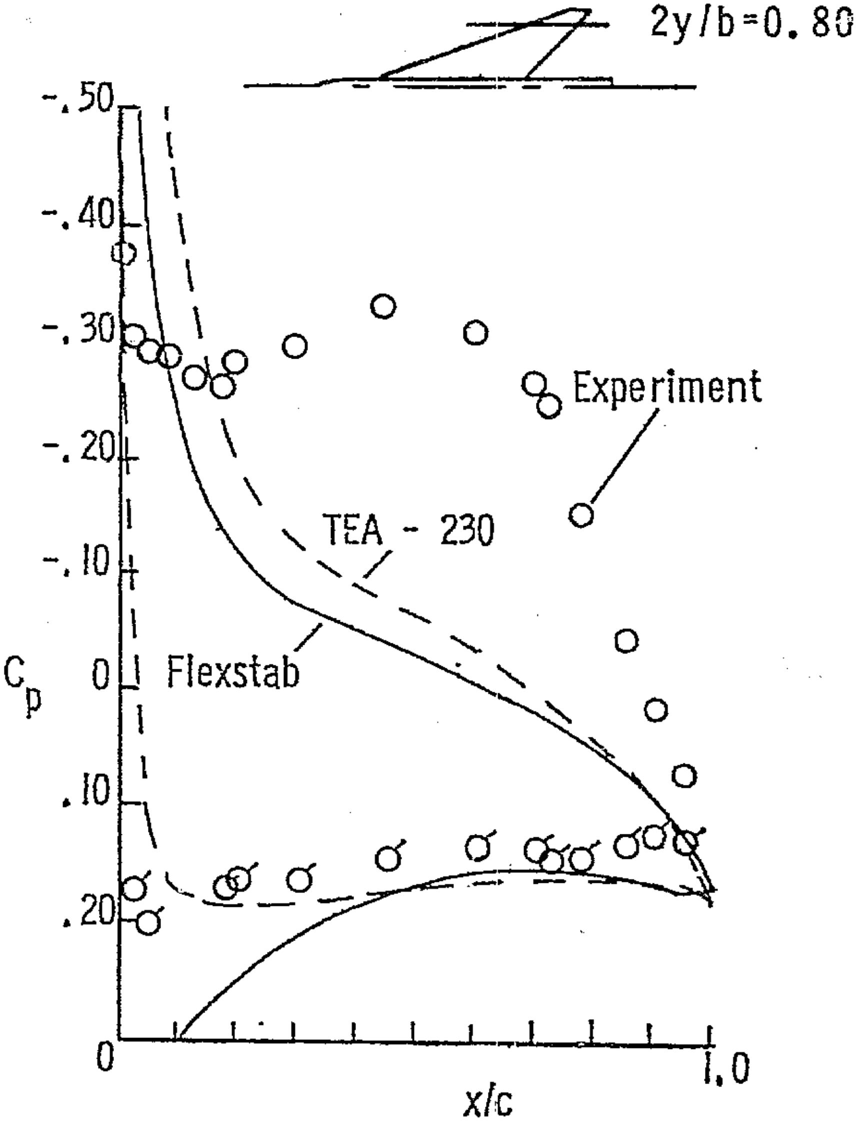


Figure 15

DELTA WING SURFACE LOADING

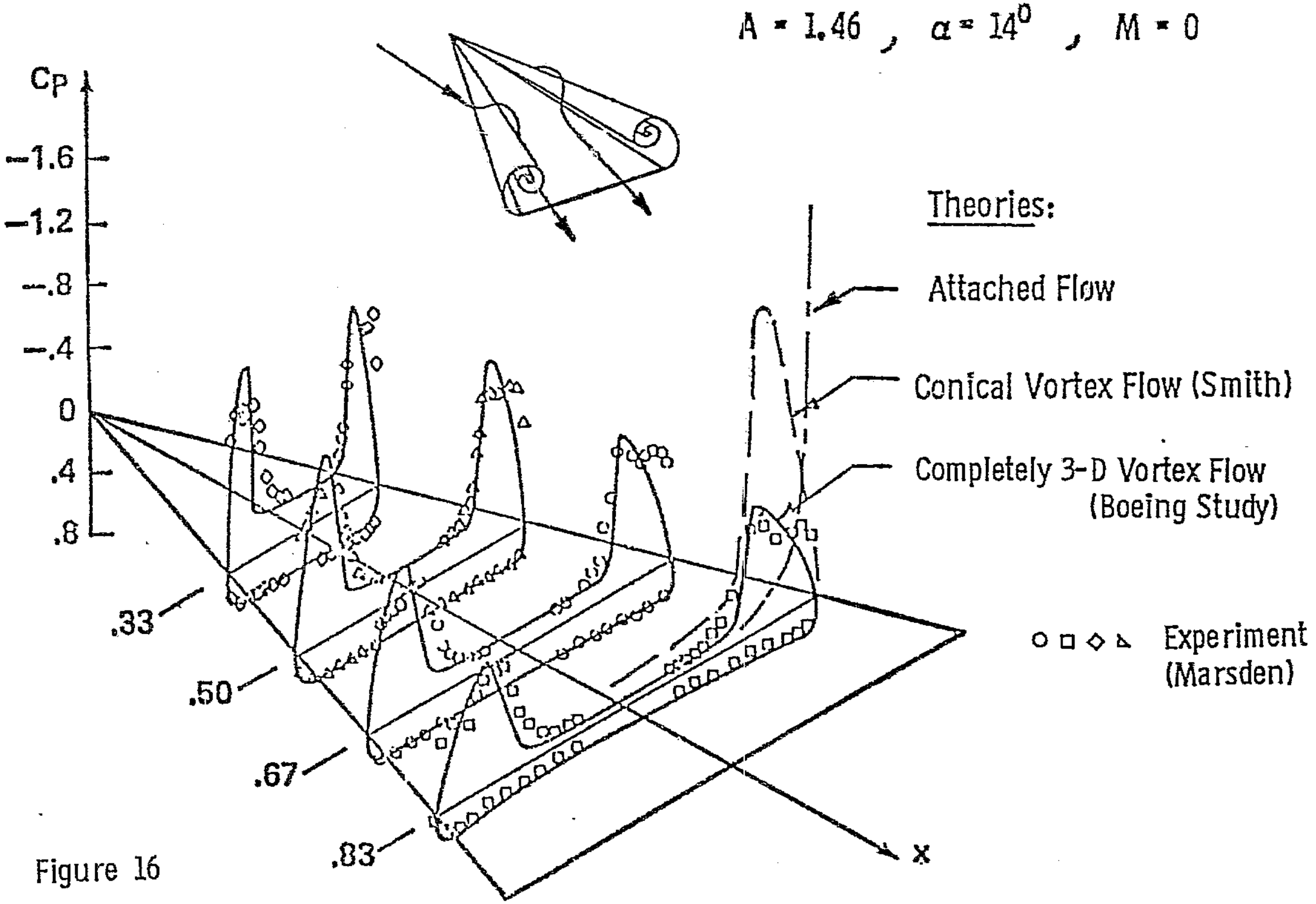


Figure 16

CLOSE-COUPLED WING-CANARD AND SLENDER-WING RESEARCH

— FUTURE STUDIES —

- EXTEND CURRENT STUDIES TO HIGH SUBSONIC SPEEDS
- DETERMINE OPTIMUM COMBINATIONS OF ATTACHED AND VORTEX FLOW CONDITIONS
- STUDY DYNAMIC STABILITY CHARACTERISTICS AT HIGH ANGLES-OF-ATTACK
- CONDUCT PARAMETRIC PRESSURE TESTS TO VERIFY FREE VORTEX-SHEET METHOD

Figure 17