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SUMMARY

A limited study in the use of theoretical methods to calculate the high speed aerodynamics of arrow wing supersonic cruise configurations has been conducted. The study consisted of correlations with existing wind tunnel data at Mach numbers from 0.8 to 2 7, using theoretical methods to extrapolate the wind tunnel data to full-scale flight conditions, and presentation of a typical supersonic data package for an advanced supersonic transport application prepared using the theoretical methods. A brief description of the methods and their application is given.

Basically, three theoretical methods were used to calculate high speed aerodynamics: (1) a group of in-house Langley analysis programs, (2) a computational system for aerodynamic design and analysis of supersonic aircraft (Boeing program), and (3) a generalized vortex lattice program (VORLAX). The first two methods are purely supersonic methods while the last applies to both subsonic and supersonic speeds. In general, all three methods had excellent correlation with wind tunnel data at supersonic speeds for drag and lift characteristics and fair to poor agreement with pitching moment characteristics. The VORLAX program had excellent correlation with wind tunnel data at subsonic speeds for lift and pitching moment characteristics and fair agreement in drag characteristics.

INTRODUCTION

Rapid advances in computers and numerical techniques have led to increasing reliance on theoretical methods to generate aerodynamic performance numbers for candidate airplane configurations. The generation of aerodynamic performance numbers for a complete aircraft configuration is the ultimate test of theoretical methods for calculating aerodynamic characteristics. Aerodynamic performance numbers require that the complete configuration be accurately represented and that any significant aerodynamic interference between aircraft components also be accounted for In addition, aerodynamic performance numbers are of an absolute nature. That is, an undesirable method to arrive at aerodynamic performance numbers is to be forced to continuously increment from a data base which less adequately represents the airplane as the design is refined.

Experimental data bases tend to be sparse for complicated and/or refined configurations. This discussion is not intended to imply that wind tunnel or flight data is not desirable or necessary. However, wind tunnel data directly applicable to the configuration under study is usually not available until late in the design process and even then the available data may be inadequate. Hence, the reliance on theoretical methods. An alternate method is to be able to rely on theoretical calculations to accurately predict aerodynamic performance of a configuration independently of a requirement for an experimental data base.

The object of the report is to present a description of some of the methods used by NASA Langley Research Center to calculate high speed aerodynamic performance; to present an experiment theory correlation for a supersonic cruise vehicle; to present an extrapolation of tunnel data to full scale using the same theoretical methods; and to present a typical aerodynamic data package for advance supersonic technology application prepared by using the theoretical methods discussed. The high speed experiment-theory correlation covers the Mach number range from 0.8 to 2.7. It was desirable to do a correlation at lower Mach numbers, (M = .4 to .6), but no experimental data was available on the configuration. A typical data base problem also exists in supersonic region where there is a data gap between Mach 1.2 and 2.3. The configuration on which the experiment-theory correlation and extrapolation to full scale is performed is the SCAT 15F-9898 which was designed and tested in the late 1960's. The configuration had a design Mach number of 2.70 and incorporated a 74 degree swept warped wing with a reflexed trailing edge and four engine nacelles mounted below the reflexed portion of the wing. The wind tunnel test data on this configuration is presented as an Appendix. The typical aerodynamic data package is for a current in-house reference supersonic aircraft designated AST-105. The AST-105 is the latest in-house study configuration used to measure and understand the benefits of advanced technology from the Supersonic Cruise Aircraft Research (SCAR) program on this type aircraft.

SYMBOLS

$\bar{c}$	wing mean aerodynamic chord
C_A	nacelle-on-wing interference-axial force coefficient, $\frac{\text{Axial Force}}{qS}$
C_D	drag coefficient, $\frac{\text{Drag}}{qS}$
$C_{D\,i}$	drag-due-to-lift coefficient
$C_{D_{form}}$	subsonic profile drag coefficient
$C_{D_{friction}}$	skin friction drag coefficient
$C_{D_{min}}$	minimum drag coeffient

$C_{D_{roughness}}$	roughness drag coefficient
$C_{D_{wave}}$	zero-lift wave drag coefficient
C_L	lift coefficient, $\frac{\text{Lift}}{qS}$
$C_{L_{MD}}$	lift coefficient for minimum drag
C_m	pitching-moment coefficient, $\frac{\text{Pitching Moment}}{qS\bar{c}}$
C_{m_o}	pitching moment coefficient at $C_L = 0$
C_N	nacelle-on-wing interference-normal force coefficient, $\frac{\text{Normal Force}}{qS}$
$\frac{\partial C_m}{\partial C_L}$	longitudinal stability parameter at $C_L = 0$
i_{tail}	horizontal tail incidence angle with respect to the wing reference plane
K_1, K_2	drag-due-to-lift parameters $C_{D_1} = K_1 + K_2 (C_L - C_{L_{MD}})^2$
L/D	lift-drag ratio
$(L/D)_{max}$	maximum lift-to-drag ratio
M	Mach number
q	free-stream dynamic pressure
S	wing reference area
x	longitudinal station
α	angle of attack, deg
δ_{L_6}	deflection of flap on wing tip with deflection measured normal to leading edge (positive for leading edge down), deg

Λ leading-edge sweep angle, deg

Model Component Designation

B	fuselage
E	engine nacelles
H	horizontal tail
V	vertical tails-centerline and wing mounted
$W_{65^{o}\ tip}$	wing with $\Lambda = 65^{o}$ tip
$W_{60^{o}\ tip}$	wing with modified $\Lambda = 60^{o}$ tip

DESCRIPTION OF THEORETICAL METHODS USED

The theoretical methods used to evaluate aerodynamic forces and moments on complete airplane configurations consist of a set of compatible computer codes that utilize linear theory The drag analysis has been performed as illustrated in figure 1. At subsonic speeds, an induced drag coefficient which includes interference effects between wing/nacelles is added to the skin friction coefficient and an empirical profile or form drag coefficient which is expressed as a percentage of the skin friction coefficient. At supersonic speeds, drag-due-to-lift which includes different degrees (according to the method used) of interference effects between components is added to the skin friction and wave drag coefficients. Differences in how interference effects are handled will be discussed as each method is discussed. This composite system of supersonic drag analysis which mixes far-field and near-field methods is discussed in reference 1. The lift and pitching moment characteristics are computed simultaneously with the induced drag or drag due-to-lift calculations and also include different degrees (according to method used) of interference effects between components. All of the computer codes employed require a numerical description of the configuration that can be defined from a standard geometry deck. A description of the geometry modeling technique is presented in reference 2 The use of a standard geometry deck to calculate both subsonic and supersonic aerodynamic characteristics is very desirable. The individual methods employed in calculating the aerodynamic characteristics are discussed in the following sections.

Skin Friction

Both wind tunnel and flight skin friction drag coefficient at subsonic and supersonic speeds has been computed using the T' method described in reference 3. The configuration drag coefficient for a given Mach number, stagnation temperature, and Reynolds number per unit length for wind tunnel test conditions or for a given Mach number - altitude (standard or the +10^{o}C hot day is also specified) flight condition is computed by representing the various configuration components by appropriate wetted areas and reference lengths assuming smooth flat plate, adiabatic wall, boundary layer conditions Transition location from laminar to turbulent boundary is specified for wind tunnel skin friction and it is assumed to occur at the leading edge of each component for full scale flight conditions Configuration components, such as the wing or tail which may exhibit significant variations in reference length, are further subdivided into strips for a more accurate determination of the friction drag.

Subsonic Profile Drag

In addition to the subsonic friction drag, there is also a pressure or separation - drag component originating along the afterbody of airfoils or fusiform bodies. This component does not lend itself to theoretical analysis and is evaluated by the empirical methods of reference 4. Each component is assigned a form factor which increases the skin friction by a percentage (usually between 3 to 5 percent)

Generalized Vortex Lattice Method

A generalized vortex lattice program identified as VORLAX was used exclusively to calculate the subsonic induced drag and as one of three methods to calculate supersonic drag-due-to-lift. The VORLAX method presented in reference 5 is applicable to complete aircraft configurations at both subsonic and supersonic flight conditions. The computational capabilities of the program include determination of (1) surface pressure or net load coefficient distribution, (2) aerodynamic force and coefficients, (3) surface warp (camber and twist) design in order to support a given pressure distribution, (4) longitudinal/lateral stability derivatives, (5) ground and wall interference effects, and (6) flow field properties. Both symmetric/asymmetric configurations and/or flight conditions can be considered. Assumptions basic to the method require attached flow, small perturbations, all subsonic or supersonic (no mixed transonic) flow, straight Mach lines, and rigid vortex wake. Techniques for simulating nonzero thickness lifting surfaces and fusiform bodies are also implemented.

The basic element of the method is the swept horseshoe vortex whose trailing legs have both bound and free segments. Associated with each horseshoe vortex is a control point at which the local boundary condition is applied. The horseshoe vortices provide a velocity field which is used to generate the coefficients of a system of linear equations relating the unknown vortex strengths to the appropriate boundary conditions. Solution of this system of linear equations results in the calculation of the configuration aerodynamic characteristics. If the design mode is desired instead, a straight-forward matrix multiplication is used to determine the surface slope distribution required to support the given pressure distribution.

For the experimental/theoretical correlation section of this report, the configuration was represented to the VORLAX program as cambered planar surfaces with engine nacelles (no thickness and fuselage volume). Induced drag or drag-due-to-lift, lift, and pitching moment coefficients which included wing/nacelle interference were calculated.

Zero-Lift Wave Drag

The far-field wave drag program uses the supersonic area rule concept to compute the zero-lift wave drag of an arbitrary configuration as described in reference 6 The program calculates equivalent bodies of revolution by passing a number of cutting planes inclined at the Mach angle through the configuration for several different aircraft roll angles. The wave drag of each equivalent body is determined from the von Karman slender body theory which relates the wave drag to the freestream conditions and the equivalent body area distribution. The discrete equivalent body wave drag values are integrated around the configuration to obtain overall wave drag.

Nacelle Interference Effects

Interference loads imposed on the wing by the four nacelles located below the wing at the trailing edge have been computed using a modified version of the method described in reference 7. The program uses modified linearized theory to compute the loads imposed on a warped wing suface by nacelles located either above or below the wing. The method of reference 6 was modified to allow for the lower supersonic Mach number cases in which the interference flow field from the nacelles spills over the wing leading edge or tip and simultaneously effects the upper and lower wing surfaces. These nacelle on wing interference coefficients are used directly in the lift analysis discussed in the following section to obtain the lift, drag-due-to-lift, and pitching moment characteristics with the nacelle interference effects included.

Lift Analysis

The wing lifting characteristics, drag-due-to-lift, and pitching moment behavior were computed using the method described in reference 8. Based on linearlized supersonic wing theory, the method breaks an arbitrary planform arrangement into a mosaic of "Mach-box" rectilinear elements which are assumed to lie in the horizontal ($z = 0$) plane. These grid elements are then employed to numerically evaluate the linear theory integral equation which relates the lifting pressure at a given field point to the wing surface slopes in the region of influence of that field point. The overall force coefficients for the camber surface at incidence are obtained by integrating the computed pressure distribution over the wing surface. This solution is combined using a superposition technique with a flat-wing solution per unit angle of attack to obtain the variation of the force coefficient with angle of attack. The nacelle interference effects previously discussed are incorporated with the lift, drag-due-to-lift, and pitching moments charactersitics computed by this method.

Supersonic Design and Analysis System (Boeing Program)

The Boeing Company has extended and combined all of the programs previously discussed except the VORLAX program into an integrated system of computer programs (see references 9-11). The extensions to the analysis methods are: Addition of a near field (thickness pressure) wave drag program, an improved lift analysis program which provides for separate modeling of fuselage lift and includes the interference of wing lift on nacelles, and the addition of pressure limiting terms in the lifting pressure programs to constrain the linear theory solution. A brief description of these extensions is given below.

Near-Field Wave Drag

The near-field wave drag program computes zero-lift thickness pressure distributions for an arbitrary wing-body-nacelle-empennage configuration. The distributions are integrated over the cross-sectional areas of the configuration to obtain the resultant drag force. The "Whitham" near-field method is used to define pressure distributions propagating from the fuselage or nacelles and superposition is used to calculate the interference drag terms associated with the pressure field from a component acting on the surfaces of the other components The following interference terms are included: wing on fuselage, fuselage on wing, nacelle on wings, wing on nacelle, fuselage on nacelle, nacelle on fuselage, and nacelle on nacelle. The near-field method is particularly useful in studying pressure distributions.

Lift Analysis

The lift analysis program uses the same basic technology used in the previously discussed individual programs, but includes the following additional features: The effect fuselage upwash field on the wing/canard, the effect of wing downwash on the fuselage lift distribution, and the effects of the wing pressure field acting on the nacelles The fuselage is assumed to be a body of revolution and the local surface angles of attack of the wing/canard are increased by the fuselage upwash values. If the area growth of fuselage is asymmetric (e.g. a high or low wing configuration), an approximate method is used to compute the asymmetric fuselage pressure field using the Whitham technique. In addition, an optional pressure limiting feature is provided. The permissible level of upper surface pressure coefficient that is calculated by linear theory may be set to a specified fraction of vacuum C_p.

For the experimental/theoretical correlation section of this report, the Boeing program was one of three methods used to calculate supersonic drag-due-to-lift, lift, and pitching moment characteristics of the configuration. Skin-friction and far-field wave drag coefficients were added to the drag-due-to-lift to determine the drag polars using this method.

COMPARISON OF THEORETICAL AND EXPERIMENTAL RESULTS

An experiment-theory correlation was performed on the SCAT 15F-9898 configuration in the Mach number range from 0.8 to 2.7. Drawings of the complete model configuration are shown in figure 2. The configuration had a design Mach number of 2.70 and incorporated a 74 degree swept warped wing with a reflexed trailing edge and four engine nacelles mounted below the reflexed portion of the wing. Because of the desire to do a correlation for a model component buildup, the theoretical/experimental correlation was carried out on two slightly different configurations. The configuration for the Mach 2.3 to 2 7 test had a 65° leading edge sweep on the wing tip and component build-up data was available for this configuration. The configuration for Mach .8 to 1.2 had an extended wing tip which had a leading edge sweep angle of 60° and include leading edge flaps (see fig. 2(b)). No model component build-up was available at the lower Mach numbers. Details of the wind tunnel test, corrections, and tabulated results are given in the Appendix.

Figure 3 presents the experiment theory correlation at Mach number 0.8 and 0.9. Theory in this case is the VORLAX program results for the configuration represented as a cambered planar surface plus skin friction and form drag coefficients. The VORLAX program results include wing/nacelle interference. There is excellent agreement between theory and experiment for lift and pitching moment characteristics, but only fair agreement in drag coefficient characteristics. Hopefully, representing the fuselage volume and planar surface thicknesses to the VORLAX program will improve the correlation in drag.

The correlation for theory and experiment at Mach 1.20 is presented in figure 4 for the configuration with the wing tip leading-edge flap (see figure 2(b)) deflected 0, 10, and 20 degrees The theories shown are the results of the individual supersonic programs collectively referred to as the Langley programs and of the Boeing program. The VORLAX program did not converge on a result at Mach 1.20. The Langley programs did an excellent job of predicting the aerodynamic characteristics for the undeflected flap case, while the Boeing program did only slightly worse. Experimental aerodynamic effects due to deflection of the leading-edge flap are small and except for drag, are predicted to be so. The Boeing program does a good job predicting the drag effects and the Langley program over-predicts the effect on drag coefficient of deflecting the flap, especially for the 20° flap deflection. However, linear theory programs are not expected to do very well at this high of a deflection angle.

Figures 5 through 7 show the correlation between theory and experiment obtained at Mach number 2.30 and 2.70 for the three theoretical methods used. The VORLAX results are shown as four discrete calculated points instead of a curve because a sufficient number of calculations were not performed to define the drag characteristics completely. The theoretical results are carried to as high a lift coefficient as the experimental results are generally available; however, the area of primary interest is usually lift coefficients from 0.0 to approximately 0.15.

The data in figures 5 through 7 are for a component build-up of the SCAT 15F-9898 configuration. Figure 5 presents data for the wing-fuselage, the four engine nacelles are added in figure 6, and data for the complete configuration is presented in figure 7. The correlation of theory and experiment is excellent for drag and lift coefficient characteristics for all three theoretical methods and for any combination of components. The Boeing program defines the drag polars better at the higher lift coefficients, however, there is little difference at the lift coefficients of interest. Pitching moment characteristics for any component combination are not well predicted by either of the three theoretical methods. In general, the individual Langley programs tend to be reasonably close to predicting the zero-lift pitching moment coefficient, while the Boeing and VORLAX do a considerably poorer job.

CORRECTION TO FULL SCALE

The major corrections of wind-tunnel data to full scale airplane conditions are the skin friction coefficient correction for Reynolds number differences and the drag coefficient correction due to geometry differences between the model and airplane. The drag coefficient correction due to differences between the model and airplane is usually limited to a correction for model distortion to accomodate the balance and sting if the model is properly constructed. Proper construction entails not only representing the airplane geometry as accurately as possible, but also scaling the inlet diverter height so that airplane spillage is duplicated In addition to the major corrections, the surface roughness due to manufacturing techniques has been found to be approximately 3 and 6 percent of the skin friction coefficient at subsonic and supersonic speeds respectively. The wind-tunnel data may or may not be corrected for grit drag. If the data is not corrected for grit drag, a grit off run to get a grit on/off drag coefficient increment and sublimation photographs to correct to turbulent conditions. The additional laminar flow region associated with grit off are required. This information was available to correct the SCAT 15F-9898 data, however, if this type data is not available, the method of reference 12 may be used to predict the drag of roughness elements used in boundary layer trips. In addition to the foregoing corrections, operational items, such as air conditioning, engine bleed drag, etc , are sometimes included in the extrapolation to full scale. These operational items are not included in the aerodynamic extrapolation to full scale performed herein.

The extrapolation of wind tunnel data to full scale aircraft conditions presented in figure 8 is for the SCAT 15F-9898 at the design Mach number of 2.7. Drag coefficient increments applied to the tunnel data are presented in Table I. The airplane skin friction coefficient was calculated for an altitude of 18 288M (60,000 ft) assuming a standard atmosphere, surface emittance of 0.8, and a surface sand grain roughness of 9 microns (3×10^{-15} ft) The drag increment due to surface roughness (rivet heads, gaps, etc.) and miscellaneous surface defects was assumed to be 6-percent of the airplane skin friction drag. Model/airplane geometry differences due to model distoration to insert a balance and sting resulted in a wave drag increment to be applied to the data. Wetted area differences due to model distortion was accounted for in the respective skin friction calculations. The grit drag increment was calculated by the method previously discussed. As seen in figure 8, applying these drag increments to the wind tunnel data result in an extrapolated airplane $(L/D)_{max}$ of 9.6 as compared to 7.2 for the wind tunnel test for the baseline drag polar. Depending on the method used to trim the aircraft, i.e. center of gravity control by pumping fuel or hoizontal tail deflection, a trim drag increment may or may not need to be applied to the drag characteristics. Also, it was assumed that the wind tunnel tests correctly modeled the airplane pitching moment and lift characteristics, so no corrections to these were necessary.

TYPICAL SST APPLICATION

Theoretical lift, drag, and pitching moment characteristics of the AST-105 configuration (figure 9) have been computed for Mach numbers from 1.1 to the 2.62 cruise condition. In addition, the horizontal tail incidence angle required for maximum configuration performance has been calculated and maintained at all Mach numbers. The analysis has utilized the individual methods (Langley programs) that have been previously discussed. The data package presented is typical of that used for supersonic aerodynamics in preliminary sizing and performance evaluation.

The design Mach 2 62 equivalent area distributions developed by the wave drag program for both the fuselage and the complete configuration are shown in Figure 10. The smoothness of this curve indicates that wave drag has been minimized at the cruise Mach number. Any "bumps" in this curve indicate a potential to improve the drag characteristics by area-ruling the configuration. The configuration wave drag variation with Mach number is presented in figure 11. The skin friction analysis along the desired Mach number-altitude flight profile is presented in figure 12 where both climb and cruise conditions are illustrated. Table II presents component wetted areas and skin friction values for three representative Mach number-altitude combinations. The configuration roughness drag increment has been assumed to be six-percent of the friction drag for the Mach 2.62 cruise condition For the lower Mach number, the ratios of roughness drag to skin friction increases as Mach number is lowered toward one. These ratios are based on estimates by aircraft manufacturers for similar configurations. The resulting variation of roughness drag coefficient with Mach number is shown in figure 12.

Interference loads imposed on the wing by the four nacelles located below the wing at the trailing edge have been computed by the nacelle interference program and are summarized in figure 13 as normal and axial force coefficients. The interference effects are used directly in the drag-due-to-lift analysis to obtain the lift, drag-due-to-lift, and pitching moment characteristics with the nacelle inteference effects included. If it is assumed that trim requirements for the configuration are met through suitable center-of-gravity control, then aircraft performance is optimized by flying the configuration with the horizontal tail oriented to maximize lift-to-drag ratio and each Mach number. Figure 13 presents the results of a study to determine the required tail incidence angle. As the figure indicates, the configuration $(L/D)_{max}$ is not overly sensitive to tail setting. Table III presents the horizontal tail incidence angle used to maximize the overall aerodynamics characteristics presented in the following figures. The configuration, drag-due-to-lift parameters with tail settings as indicated are presented in figure 14.

The overall aerodynamic characteristics for the AST-105 configuration are summarized in figures 15 through 19 Typical drag polars obtained by combining the various zero-lift drag items with the drag-due-to-lift characteristics (as shown in figure 1(b)) are presented in figure 15. The associated lift curves are shown in figure 16 while the $(L/D)_{max}$ variation with Mach Number derived from these lift and drag characteristics is summarized in figure 17. A start of cruise value of 9.0 is indicated by the analysis.

The pitching moment characteristics are presented in figures 18 and 19. The pitching moment characteristics have been computed using the horizontal tail incidence angles previously discussed and with center-of-gravity locations typical of an actual mission profile for the AST-105, thus both ascent and descent pitching moment characteristics are indicated.

CONCLUDING REMARKS

A limited study in the use of theoretical methods to calculate the high speed aerodynamics of arrow wing supersonic cruise configurations has been conducted. The study consisted of correlations with existing wind tunnel data at Mach numbers from 0.8 to 2.7, using theoretical methods to extrapolate the wind tunnel data to full-scale flight conditions, and presentation of a typical supersonic data package for an advanced supersonic transport application prepared using the theoretical methods. A brief description of the methods and their application is given.

Basically, three theoretical methods were used to calculate high speed aerodynamics (1) a group of in-house Langley analysis programs, (2) a computational system for aerodynamic design and analysis of supersonic aircraft (Boeing program), and (3) a generalized vortex lattice program (VORLAX). The first two methods are purely supersonic methods while the last applies to both subsonic and supersonic speeds. In general, all three methods had excellent correlation with wind tunnel data at supersonic speeds for drag and lift characteristics and fair to poor agreement with pitching moment characteristics. The VORLAX program had excellent correlation with wind tunnel data at subsonic speeds for lift and pitching moment characteristics and fair agreement in drag characteristics.

APPENDIX

WIND TUNNEL DATA

Drawings of the model are shown in figure 2. Detailed geometric characteristics of the baseline (65^o tip) model are presented in Table A-I. The model scale was 0.015, which represents a full-scale supersonic transport aircraft configuration approximately 91.44m (300 ft) in length.

The model incorporated a slender cambered body with a 74^o swept wing planform which was designed for a cruise lift coefficient of 0.08 at a Mach number of 2.7. The wing had a subsonic leading edge except in the region of the tip where the leading edge angle was decreased to 65^o on the basic configuration and 60^o on the subsonic-transonic modified configuration. The modified wing tip was equipped with movable leading edge flaps (figure 2(b)).

A small horizontal tail (figure 2(c)) was mounted aft on the fuselage to provide longitudinal pitch control. Two vertical tails (figure 2(d)) were mounted on the outboard wing panels for directional stability and to improve the air flow in the region of the wing tip. A fuselage mounted vertical tail (figure 2(e)) together with a ventral fin (figure 2(f)) was also included to provide directional control.

Four engine nacelles (figure 2(g)) were located below the wing near the wing trailing edge to simulate the engine installation. The wing trailing edge was reflexed upward in the region of the engine nacelles in order to essentially cancel the lift interference from the nacelles and to improve the drag interference effects of the wing nacelle combination at cruise conditions.

Tests and Corrections

The tests were conducted in the Langley 8-Foot Transonic Pressure and Unitary Plan Wind Tunnels. The tests were made through an angle-of-attack range from about -8^o to 12^o. The nominal test conditions were as follows:

M	Stagnation Temperature, K (f)	Stagnation Pressure, kPa (psfa)	R/M (f) X 10^6
0.80	322 (120)	53.72 (1122)	6.56 (2.0)
0.90	322 (120)	51.04 (1066)	6.56 (2.0)
1.20	322 (120)	49.99 (1044)	6.56 (2.0)
2.30	339 (150)	73.35 (1532)	6.56 (2.0)
2.70	339 (150)	90.40 (1888)	6.56 (2.0)

The stagnation dewpoint temperature was maintained sufficiently low to avoid any significant condensation effects in the test section.

Aerodynamic forces and moments were measured by means of a six-component strain gage balance housed within the model The angles of attack were corrected for the deflection of the balance and sting under aerodynamic load and for tunnel flow angularity. The balance-chamber pressure and nacelle base pressure were measured, and the drag force was adjusted to a base pressure equal to free-stream static pressure. In addition, the drag results have been corrected for internal skin-friction drag of the nacelles.

To insure boundary-layer transition to turbulent flow at conditions between Mach 0 80 and 1.20, transition strips 0.16m (1/16 in) of No. 60 carborundum were placed in the body 3.05m (1.2 in) aft of the nose and strips of No. 80 carborundum grit were placed streamwise 1.52m (0.6 in) from the leading edge of all external surfaces and on the inside of the engine nacelles. At conditions of Mach 2.30 and 2 70, strips of No. 60 grit on all outer surfaces and No. 80 on the inner surfaces of the nacelles were located 1.02m (0.4 in) aft the leading edge, except for the fuselage where transition strips remained the same as for the lower Mach numbers. These transition strips were shown to be adequate in the conclusions of reference 13.

Presentation of the Results

Tabular listings of the data that is used in this report from the wind tunnel investigations are presented. The experimental data in the main body of this report are plots of this data, except for drag characteristics at Mach 1.20. The tabular data for Mach 1.20 had a constant nacelle base drag coefficient correction of .0021 applied to drag results, the correction should have varied linearly from 0016 at C_L = - 05 to .0022 at C_L = 0.42. The linearly varying correction was applied to the plotted data, otherwise the tabular data was used as presented below.

Tabular Data

Run No.	Mach No.	Configuration
8	0.80	W_{60^o} tip BEVH $\delta_{L_6} = 0^o$
3	0.90	
4	1.20	
10	1 20	$\delta_{L_6} = 10^o$
16	1.20	$\delta_{L_6} = 20^o$
93	2.3	W_{65^o} tip BEVH
95	2.7	BEVH
146	2.3	BE
148	2.7	BE
150	2.3	B
152	2.7	B

TABLE A-I - GEOMETRIC CHARACTERISTICS OF MODEL

Wing W		
Aspect ratio	1.63	
Span, cm (in)	57.973	(22.824)
Area, m^2 (ft^2)	0.207	(2.227)
Root chord at fuselage centerline, cm (in)	81.473	(32.076)
Tip chord, cm (in)	3.429	(1.350)
Mean geometric chord, cm (in)	48.654	(19.155)
Thickness-chord ratio, near root	0.030	
Thickness-chord ratio, tip	0.027	
Fuselage, B		
Length, cm (in)	137.160	(54.000)
Balance-chamber area, m^2 (ft^2)	.0015	(0.0158)
Horizontal tail, H		
Aspect ratio, exposed	2.42	
Span, exposed cm (in)	7.544	(2.970)
Area, exposed m^2 (ft^2)	.0047	(0.0504)
Root chord at fuselage juncture, cm (in)	10.201	(4.016)
Tip chord, cm (in)	2.258	(0.889)
Mean geometric chord, cm (in)	7.061	(2.780)
Airfoil section	Half circular arc	
Thickness-chord ratio	.030	
Centerline vertical tail		
Area, m^2 (ft^2)	.0055	(0.059)
Airfoil section	Half circular arc	
Thickness-chord ratio	0.025	
Outboard vertical tails		
Area (each), m^2 (ft^2)	.0055	(0.059)
Airfoil section	Half circular arc	
Thickness-chord ratio	0.025	
Centerline ventral fin		
Area, m^2 (ft^2)	.0022	(0.024)
Airfoil section		Wedge slab
Nacelles, E		
Length, cm (in)	17.089	(6.728)
Base area (each) m^2 (ft^2)	.00031	(0.00336)

REFERENCES

1. Carlson, Harry W. and Harris, Roy V., Jr.: "A Unified System of Supersonic Aerodynamic Analysis." NASA SP-228, 1970.

2. Craidon, Charlotte B.: "Description of a Digital Computer Program for Airplane Configuration Plots." NASA TM X-2074, 1970.

3. Sommer, Simon C. and Short, Barbara J.: "Free Flight Measurements of Turbulent Boundry Layer Skin Friction in the Presence of Severe Aerodynamic Heating at Mach Numbers from 2.8 to 7.0." NACA TN-3391, 1955.

4. Hoerner, S. F.: "Fluid-Dynamic Drag." Hoerner Fluid Dynamics, 1965.

5. Miranda, Luis R.; Elliott, Robert D ; and Baker, William M.: "A Generalized Vortex Lattice Method for Subsonic and Supersonic Flow Applications." Lockheed-California Company, LR 28112, 1977

6. Harris, Roy V. Jr.: "An Analysis and Correlation of Aircraft Wave Drag." NASA TM X-947, 1964.

7. Mack, Robert J.: "A Numerical Method for the Evaluation and Utilization of Supersonic Nacelle-Wing Interference." NASA TN D-50057, 1968.

8. Carlson, Harry W. and Miller, David S.: "Numerical Methods for the Design and Analysis of Wings at Supersonic Speeds." NASA TN D-7713, December 1974.

9. Middleton, W. D. and Lundry, J. L.: "A Computational System for Aerodynamic Design and Analysis of Supersonic Aircraft, Part 1-General Description and Theoretical Development," NASA CR-2715, 1976.

10. Middleton, W. D.; Lundry, J. L.; and Coleman, R. G.: "A Computational System for Aerodynamic Design and Analysis of Supersonic Aircraft Part 2-Users Manual." NASA CR-2716, 1976.

11. Middleton, W. D.; Lundry, J. L.; and Coleman, R. G.: "A Computational System for Aerodynamic Design and Analysis of Supersonic Aircraft, Part 3-Computer Program Description." NASA CR-2717, 1976.

12. Stallings, Robert L., Jr; and Lamb, Milton. "Effects of Roughness Size and Position of Boundary Layer Transition and on the Aerodynamic Characteristics of a 55^{o} Swept Delta Wing at Supersonic Speeds." NASA TP-1027, 1977.

13. Braslow, Albert L,; Hicks, Raymond, M.; and Harris, Roy V., Jr.. "Use of Grit Type Boundry-Layer-Transition Trips on Wind-Tunnel Models." NASA TN D-3579, 1966.

NASA LANGLEY RESEARCH CENTER 8-FT TPT SCAT 15-F

TEST 503 RUN 4 MACH 1.200 CONFIG. 1

POINT	MINF	Q	BETA	ALPHA	CN	CA	CM	CROLL	CYAW	CSIDE	CL	CD	L/D
90	1.200	417.53	.00	-5 62	-.1194	.01139	.0362	.0004	.0008	-.0024	-.1177	.01992	-5.91
91	1.203	417.58	.01	-4 25	- 0631	.01361	.0274	.0003	.0007	-.0023	-.0619	.01514	-4 09
92	1 200	417.68	.01	-3 50	-.0301	.01493	.0221	.0003	.0006	-.0020	-.0292	.01355	-2.15
93	1 201	417 66	.01	-2 79	-.0006	.01582	.0170	.0002	.0005	-.0020	.0002	.01273	.01
94	1.201	417.71	.01	-2.11	.0261	.01668	0124	-.0000	.0004	-.0021	.0267	01761	2.12
95	1 201	417.71	.01	-1 41	.0534	.01735	.0079	-.0002	.0003	-.0018	.0539	.01293	4.16
96	1.203	417.63	.01	-.73	.0789	01784	.0050	-.0003	.0003	-.0019	.0791	.01373	5.76
97	1 203	417.63	.01	-.02	.1077	.01824	-.0006	-.0004	.0004	-.0020	.1077	.01510	7.13
98	1 200	417.63	.01	.69	.1369	.01851	-.0052	-.0004	.0004	-.0020	.1367	.01705	8.02
99	1 200	417 69	01	1.39	.1652	.01974	- 0101	-.0003	.0004	- 0018	1657	.01964	8 43
100	1.200	417.63	.01	2 11	.1994	01929	-.0159	- 0005	.0002	-.0014	.1985	02351	8.45
101	1.200	417.66	.01	2.89	.2376	.02027	-.0219	-.0006	-.0001	-.0008	.2363	.02909	8.12
102	1.200	417.63	.01	3.69	.2755	.02144	- 0272	-.0007	-.0001	-.0010	.2735	.03597	7.60
103	1 200	417.63	.01	4.50	.3160	.02293	-.0318	-.0007	.0000	-.0009	.3132	.04456	7.03
104	1 203	417 63	.01	5 35	.3564	.02452	-.0355	-.0008	-.0000	-.0009	.3525	.05455	6.46
105	1.200	417.66	01	6 19	.3941	.02612	-.0387	-.0009	.0000	-.0010	.3890	.06535	5 95
106	1.200	417.60	.01	7 05	.4339	.02796	-.0416	-.0010	.0000	-.0012	.4272	.07792	5.48
107	1.200	417 56	.01	7.94	.4746	.03004	-.0442	-.0012	.0002	-.0017	.4659	.09224	5.05
108	1 200	417.60	.01	8.19	.4878	.03076	-.0457	-.0012	.0002	-.0017	.4784	.09687	4.94
109	1.201	417.71	.00	-5 66	-.1223	.01128	.0368	.0004	.0007	-.0021	-.1206	.02019	-5.97

POINT	ALPHA	CLSQ	CDC	CDB	CDI	R/FT	TEMP
90	-5.62	0139	.00111	.00210	.0010	2.00	121.4
91	-4 25	.0038	.00114	.00210	.0010	2.00	120.7
92	-3 50	.0009	00116	.00210	.0010	2 00	120 4
93	-2.79	0000	.00119	.00210	.0010	2 00	120 4
94	-2.11	0007	00118	.00210	.0010	2.00	120 4
95	-1.41	.0029	.00118	.00210	.0010	2.00	120.2
96	- 73	.0063	.00119	.00210	.0010	2.00	120.1
97	-.02	0116	00120	00210	.0010	2.00	120.0
98	.69	0187	00123	00210	.0010	2 00	119 9
99	1 38	.0274	00126	.00210	0010	2 00	119.9
100	2.11	.0394	.00130	.00210	.0010	2 00	120 1
101	2.88	.0559	00131	00210	.0010	2.00	120 1
102	3 68	0748	00132	00210	.0010	2.00	120.1
103	4 50	0981	.00133	00210	0010	2 00	120 1
104	5 35	1243	.00135	00210	.0010	2 00	120 1
105	6.19	.1513	00137	.00210	.0010	2 00	120 0
106	7.05	.1825	00140	.00210	.0010	2.00	120.1
107	7.94	.2171	.00144	.00210	.0010	2 00	120.1
108	8.19	.2299	.00150	.00210	.0010	2 00	120 1
109	-5.66	.0146	.00112	.00210	0010	2 00	120 1

NASA LANGLEY RESEARCH CENTER 8-FT TPT SCAT 15-F

TEST 503 RUN 10 MACH 1.200 CONFIG. 2

POINT	MINF	Q	BETA	ALPHA	CN	CA	CM	CROLL	CYAW	CSIDE	CL	CD	L/D
217	1 202	417 64	.00	-5.53	-.1167	.01214	.0367	.0005	0007	-.0021	-.1150	.02023	-5.68
218	1 200	417.45	.01	-4.11	-.0591	.01435	.0274	.0003	.0006	-.0020	-.0569	.01538	-3.70
219	1.201	417.50	.01	-3.39	-.0272	.01543	.0224	.0002	.0005	-.0020	-.0262	.01391	-1.89
220	1.201	417 53	.01	-2 78	-.0020	.01619	.0182	.0002	.0005	-.0019	-.0012	.01316	-.09
221	1.200	417 30	.01	-2.03	.0279	.01700	.0129	.0001	.0005	-.0020	.0285	.01290	2.21
222	1.200	417.28	.01	-1.37	.0531	.01758	.0086	-.0001	.0004	-.0019	.0535	.01320	4.05
223	1.201	417 50	.01	-.65	.0824	.01806	.0037	-.0002	.0004	-.0018	.0826	.01403	5 89
224	1.201	417.50	.01	.08	.1102	.01838	-.0009	-.0003	.0005	-.0020	.1102	01543	7.14
225	1.201	417.56	01	.73	.1370	.01854	-.0051	-.0003	.0005	-.0019	.1368	.01719	7.96
226	1 201	417 50	.01	1.46	.1679	.01869	-.0101	-.0005	.0005	-.0019	.1674	.01986	8.43
227	1 201	417 50	.01	2.18	.2016	01917	-.0157	-.0005	0005	-.0017	.2007	.02373	8.46
228	1.201	417 50	.00	2.99	.2408	.02005	-.0218	-.0007	.0004	-.0014	.2394	.02947	8.12
229	1.203	417 42	.01	3.76	.2791	.02111	-.0270	-.0007	.0004	-.0015	.2771	03626	7.64
230	1 203	417 40	00	4 60	.3183	.02239	-.0313	-.0007	.0003	-.0012	.3155	.04472	7.05
231	1 203	417 47	.01	5.45	.3576	.02384	- 0348	-.0008	.0002	-.0010	.3537	05461	6.48
232	1.203	417 32	.01	6.29	.3956	02535	-.0379	- 0009	0002	-.0012	.3904	.06543	5 97
233	1 203	417.36	.01	7.14	.4360	.02712	-.0409	-.0010	0002	-.0013	.4293	.07801	5 50
234	1.199	417.33	.01	8 03	.4781	.02934	-.0437	-.0012	0004	- 0019	.4693	.09273	5 06
235	1 203	417.30	.01	8.36	.4923	.03009	-.0447	-.0012	.0004	-.0020	.4827	.09825	4.91
236	1.203	417.31	.00	-5.53	-.1197	.01202	.0372	.0005	.0007	-.0022	-.1180	.02051	-5.75

POINT	ALPHA	CLSQ	CDC	CDB	CDI	R/FT	TEMP
217	-5 53	.0132	.00113	.00210	0010	2 00	121.4
218	-4.11	0032	00117	.00210	.0010	2 00	121.0
219	-3 39	0007	.00117	00210	.0010	2 00	120 9
220	-2 76	0000	00117	.00210	.0010	2.00	120 6
221	-2.03	0009	.00118	.00210	0010	2.00	120 5
222	-1.37	.0029	.00118	.00210	.0010	2.00	120 4
223	-.65	0069	.00118	00210	.0010	2 00	120 4
224	.08	0121	.00121	.00210	.0010	2 00	120.1
225	.73	0187	00124	00210	.0010	2 00	120.1
226	1.46	0280	00127	00210	.0010	2 00	120 1
227	2.18	0403	.00130	.00210	.0010	2 00	120 1
228	2.99	0573	00133	.00210	0010	2.00	120.1
229	3 76	0768	00133	00210	.0010	2.00	120 1
230	4 60	.0995	.00135	.00210	0010	2.00	120.1
231	5 45	1251	00136	.00210	0010	2.00	120 0
232	6.29	1524	00137	.00210	.0010	2.00	120.1
233	7.14	.1843	00139	.00210	.0010	2.00	120.1
234	8 03	2203	00143	.00210	.0010	2 00	120.1
235	8.36	2330	00147	00210	.0010	2 00	120 1
236	-5.58	.0139	00113	.00210	.0010	2.00	120 1

TEST 503 RUN 8 MACH .800 CONFIG. 1

POINT	MINF	Q	BETA	ALPHA	CN	CA	CM	CROLL	CYAW	CSIDE	CL	CD	L/D
172	.800	329.97	.01	-5.70	-.1260	.00389	.0308	.0004	.0006	-.0022	-.1245	.01956	-6.37
173	.801	330.65	.01	-4.31	-.0682	.01116	.0251	.0003	.0005	-.0023	-.0671	.01445	-4.65
174	.799	329.36	.01	-3.61	-.0375	.01216	.0217	.0001	.0005	-.0020	-.0366	.01269	-2.88
175	.801	330.45	.01	-2.97	-.0163	.01289	.0192	.0000	.0004	-.0020	-.0156	.01192	-1.31
176	.800	330.11	.01	-2.28	.0133	.01379	.0155	-.0000	.0003	-.0019	.0139	.01145	1.21
177	.798	328.91	.01	-1.63	.0367	.01441	.0129	-.0002	.0002	-.0018	.0371	.01156	3.21
178	.799	329.57	.01	-.97	.0610	.01492	.0105	-.0003	.0003	-.0019	.0612	.01199	5.11
179	.801	330.52	.01	-.30	.0846	.01499	.0081	-.0004	.0002	-.0017	.0847	.01274	6.65
180	.801	330.24	.01	.33	.1074	.01503	.0051	-.0004	.0002	-.0017	.1073	.01384	7.75
181	.800	330.18	.01	1.07	.1372	.01492	.0036	-.0005	.0001	-.0014	.1369	.01568	8.74
182	.800	329.70	.01	1.75	.1647	.01499	.0013	-.0005	.0001	-.0015	.1642	.01821	9.02
183	.799	329.36	.01	2.41	.1932	.01528	-.0025	-.0004	.0000	-.0013	.1924	.02160	8.91
184	.799	328.31	.01	3.13	.2280	.01595	-.0065	-.0007	.0001	-.0012	.2263	.02648	8.57
185	.799	329.50	.00	3.89	.2660	.01666	-.0079	-.0008	.0004	-.0015	.2642	.03286	8.04
186	.799	329.63	.00	4.68	.3035	.01760	-.0128	-.0008	.0005	-.0013	.3010	.04048	7.44
187	.799	329.63	.00	5.45	.3404	.01871	-.0148	-.0009	.0004	-.0011	.3371	.04918	6.85
188	.800	329.84	.00	6.28	.3829	.02024	-.0164	-.0009	.0003	-.0011	.3794	.06022	6.29
189	.800	329.70	.01	7.08	.4193	.02149	-.0171	-.0010	.0003	-.0017	.4134	.07119	5.81
190	.799	329.56	.01	7.93	.4610	.02311	-.0178	-.0010	.0003	-.0016	.4534	.08468	5.35
191	.799	329.36	.01	8.78	.5055	.02500	-.0193	-.0011	.0001	-.0013	.4959	.10007	4.95
192	.800	329.70	.01	9.56	.5493	.02715	-.0237	-.0012	.0001	-.0019	.5352	.11590	4.62
193	.799	329.56	.00	-5.72	-.1268	.00390	.0310	.0004	.0006	-.0020	-.1253	.01970	-6.36

POINT	ALPHA	CLSQ	CDC	CDB	CDI	R/FT	TEMP
172	-5.70	.0159	.00032	.00080	.0010	2.00	119.6
173	-4.31	.0045	.00032	.00080	.0010	2.00	119.7
174	-3.61	.0013	.00033	.00080	.0010	2.00	120.0
175	-2.97	.0002	.00033	.00080	.0010	2.00	120.1
176	-2.28	.0002	.00036	.00080	.0010	2.00	120.1
177	-1.63	.0014	.00036	.00080	.0010	2.00	120.1
178	-.97	.0037	.00036	.00080	.0010	2.00	120.0
179	-.30	.0072	.00038	.00080	.0010	2.00	120.0
180	.33	.0115	.00039	.00080	.0010	2.00	120.0
181	1.07	.0188	.00039	.00080	.0010	2.00	120.0
182	1.75	.0270	.00040	.00080	.0010	2.00	120.0
183	2.41	.0370	.00041	.00080	.0010	2.00	120.1
184	3.13	.0515	.00042	.00080	.0010	2.00	120.1
185	3.89	.0698	.00043	.00080	.0010	2.00	120.0
186	4.68	.0906	.00045	.00080	.0010	2.00	120.0
187	5.45	.1136	.00048	.00080	.0010	2.00	120.1
188	6.28	.1432	.00050	.00080	.0010	2.00	120.1
189	7.08	.1709	.00054	.00080	.0010	2.00	120.1
190	7.93	.2056	.00056	.00080	.0010	2.00	120.1
191	8.78	.2458	.00058	.00080	.0010	2.00	120.1
192	9.56	.2865	.00058	.00080	.0010	2.00	120.1
193	-5.72	.0157	.00031	.00080	.0010	2.00	119.9

TEST 503 RUN 3 MACH .900 CONFIG. 1

POINT	MINF	Q	BETA	ALPHA	CN	CA	CM	CROLL	CYAW	CSIDE	CL	CD	L/D
67	.901	358.40	.00	-5.74	-.1301	.00870	.0337	.0004	.0006	-.0016	-.1285	.01987	-6.47
68	.900	357.44	.00	-4.26	-.0675	.01098	.0265	.0004	.0006	-.0018	-.0664	.01417	-4.69
69	.900	357.59	.00	-3.64	-.0416	.01195	.0233	.0003	.0005	-.0018	-.0407	.01276	-3.19
70	.901	358.01	.00	-2.92	-.0102	.01302	.0191	.0002	.0003	-.0013	-.0096	.01173	-.82
71	.900	357.77	.01	-2.24	.0150	.01374	.0154	.0000	.0003	-.0014	.0155	.01134	1.37
72	.901	357.76	.01	-1.51	.0435	.01446	.0119	-.0001	.0002	-.0012	.0439	.01150	3.82
75	.901	357.88	.01	-.90	.0673	.01484	.0092	-.0003	.0002	-.0014	.0676	.01198	5.64
76	.900	357.65	.01	-.20	.0903	.01504	.0067	-.0004	.0002	-.0014	.0903	.01293	6.99
77	.900	357.72	.01	.49	.1181	.01509	.0038	-.0005	.0002	-.0014	.1179	.01429	8.25
78	.900	357.65	.01	1.17	.1448	.01510	.0011	-.0005	.0002	-.0013	.1445	.01626	8.89
79	.900	357.71	.01	1.95	.1803	.01529	-.0030	-.0003	.0001	-.0012	.1797	.01961	9.16
80	.900	357.71	.01	2.65	.2146	.01588	-.0081	-.0005	.0000	-.0008	.2135	.02397	8.91
81	.900	357.47	.01	3.39	.2505	.01659	-.0123	-.0006	.0002	-.0011	.2491	.02950	8.44
82	.900	357.48	.00	4.19	.2912	.01766	-.0167	-.0009	.0004	-.0012	.2891	.03707	7.80
83	.900	357.59	.00	5.01	.3332	.01894	-.0201	-.0008	.0003	-.0009	.3303	.04616	7.16
84	.900	357.48	.00	5.83	.3740	.02046	-.0223	-.0009	.0003	-.0010	.3700	.05657	6.54
85	.900	357.65	.00	6.70	.4171	.02218	-.0238	-.0009	.0003	-.0014	.4116	.06891	5.97
86	.900	357.54	.00	7.54	.4576	.02386	-.0247	-.0011	.0003	-.0012	.4505	.08190	5.50
87	.900	357.72	.01	8.51	.5064	.02616	-.0259	-.0011	.0002	-.0014	.4969	.09902	5.02
88	.900	357.47	.01	9.34	.5518	.02658	-.0297	-.0012	.0003	-.0017	.5398	.11599	4.65
89	.900	357.77	.00	-5.73	-.1312	.00870	.0338	.0005	.0006	-.0019	-.1297	.01995	-6.50

POINT	ALPHA	CLSQ	CDC	CDB	CDI	R/FT	TEMP
67	-5.74	.0165	.00027	.00080	.0010	1.99	123.2
68	-4.26	.0044	.00029	.00080	.0010	1.99	121.6
69	-3.64	.0017	.00027	.00080	.0010	2.00	121.1
70	-2.92	.0001	.00029	.00080	.0010	2.00	120.6
71	-2.24	.0002	.00031	.00080	.0010	2.00	120.4
72	-1.51	.0019	.00032	.00080	.0010	2.00	120.1
75	-.90	.0046	.00033	.00080	.0010	2.00	120.1
76	-.20	.0082	.00034	.00080	.0010	2.00	120.1
77	.49	.0139	.00034	.00080	.0010	2.00	120.1
78	1.17	.0209	.00035	.00080	.0010	2.00	120.1
79	1.95	.0323	.00036	.00080	.0010	2.00	120.0
80	2.65	.0456	.00037	.00080	.0010	2.00	120.0
81	3.38	.0621	.00039	.00080	.0010	2.00	120.0
82	4.19	.0836	.00042	.00080	.0010	2.00	120.1
83	5.01	.1091	.00044	.00080	.0010	2.00	120.1
84	5.83	.1369	.00046	.00080	.0010	2.00	120.1
85	6.70	.1694	.00050	.00080	.0010	2.00	120.0
86	7.54	.2030	.00054	.00080	.0010	2.00	119.9
87	8.51	.2469	.00057	.00080	.0010	2.00	120.1
88	9.34	.2914	.00058	.00080	.0010	2.00	120.1
89	-5.73	.0168	.00027	.00080	.0010	2.00	120.1

NASA LANGLEY RESEARCH CENTER 8-FT TPT SCAT 15-F

TEST 503 RUN 16 MACH 1.200 CONFIG. 3

POINT	MI F	Q	BETA	ALPHA	CN	CA	CM	CROLL	CYAW	CSIDE	CL	CD	L/D
17	1.200	417.21	.00	-5.59	-.1225	.01269	.0382	0003	.0008	-.0022	-.1207	.02146	-5.62
18	1 200	417 26	.01	-4.17	-.0635	.01490	.0293	.0002	.0006	-.0022	-.0623	.01638	-3.80
19	1.200	417.26	.01	-3 41	-.0312	.01598	.0241	0001	.0005	-.0020	-.0302	.01471	-2.05
20	1.200	417.26	.01	-2.73	-.0047	.01675	.0196	.0001	.0005	- 0019	- 0039	01385	-.28
21	1 200	417 26	.01	-2.04	.0236	01744	.0147	-.0000	.0004	- 0018	.0242	01348	1 80
22	1 200	417 26	.01	-1 33	0515	01793	.0098	- 0001	0004	- 0018	.0519	01363	3.81
23	1 201	417.29	.01	-.64	.0794	.01828	.0050	-.0002	.0005	-.0020	.0796	01429	5 57
24	1.201	417 29	.01	.05	.1062	.01857	.0003	-.0003	0004	- 0019	.1062	.01557	6 82
25	1 201	417.29	01	.70	.1324	.01874	- 0041	-.0003	0005	- 0019	.1322	.01726	7 66
26	1.201	417.29	.01	1.41	.1647	.01891	-.0093	-.0005	.0005	- 0018	.1642	01986	8.27
27	1.201	417.29	.01	2.16	.1977	.01934	-.0149	-.0006	.0004	- 0015	.1968	.02367	8.31
28	1.201	417.29	.00	2.92	.2349	.02013	-.0205	-.0007	0003	-.0012	.2336	.02898	8 06
29	1.200	417.24	.00	3.73	.2731	.02108	-.0256	-.0008	.0004	-.0012	.2711	.03569	7.60
30	1 200	417.17	.00	4 57	.3131	.02223	-.0299	-.0008	.0004	-.0012	.3103	.04400	7.05
31	1 199	417.03	.00	5.37	.3490	.02326	-.0329	-.0009	.0003	- 0010	.3453	.05271	6.55
32	1.198	416.93	.00	6.26	.3904	.02453	-.0361	-.0010	.0003	-.0011	.3854	.06387	6.03
33	1 198	416.96	.01	7.13	.4302	.02599	-.0387	-.0011	.0002	- 0013	.4237	.07610	5.57
34	1.198	416.75	.01	8.04	.4731	.02797	-.0420	-.0012	.0003	-.0017	.4646	.09078	5 12
35	1.199	417 04	.01	8 37	.4889	.02877	-.0431	-.0012	.0003	- 0016	.4795	.09650	4.97
36	1 200	417.19	.01	1 43	.1656	.01892	-.0095	-.0005	.0006	- 0019	.1651	.01993	8.28

POINT	ALPHA	CLSQ	CDC	CDB	CDI	R/FT	TEMP
17	-5.59	.0146	00112	.00210	.0010	2.00	120.5
18	-4 17	0039	.00114	.00210	.0010	2 00	120.4
19	-3 41	.0009	.00117	.00210	.0010	2.00	120.4
20	-2 73	.0000	.00118	.00210	.0010	2.00	120.4
21	-2 04	.0006	.00119	.00210	.0010	2.00	120 2
22	-1 33	0027	.00119	.00210	0010	2 00	120.4
23	-.64	.0063	00121	.00210	.0010	2 00	120 2
24	.05	0113	.00121	.00210	.0010	2.00	120 2
25	.70	.0175	.00125	00210	.0010	2 00	120 2
26	1 41	0270	.00129	00210	.0010	2 00	120.2
27	2.16	0397	.00131	.00210	.0010	2.00	120.1
28	2.92	0546	00133	.00210	.0010	2.00	120.1
29	3.73	.0735	00134	.00210	.0010	2 00	120.1
30	4.57	0963	.00135	.00210	.0010	2.00	120.0
31	5 37	.1192	.00137	00210	.0010	2 00	120.0
32	6 26	1485	.00138	.00210	0010	2.00	120.1
33	7.13	.1795	.00139	.00210	.0010	2 00	120.1
34	8.04	2159	.00142	.00210	.0010	2.00	120.1
35	8.37	.2279	00146	.00210	.0010	2.00	120.2
36	1.43	.0272	.00128	.00210	.0010	2.00	120 0

NASA LANGLEY RESEARCH CENTER UPWT TS2 PRELIMINARY 01/14/69

BODY AXIS PRJ 927 RUN 93 MACH 2.30 CONFIG. CODE 0 BATCH 17

POINT	MACH	DYN PRS	BETA	ALPHA	CN	CA	CM	CLB	CNB	CY	CAC	CABN	CAI
1757	2 30	454 66	02	-7 18	-.1322	.0054	.0207	.0001	0000	-.C005	.0011	.0009	.0008
1758	2 30	454 90	.02	-5.88	-.0943	.0072	.0168	.0001	.0000	-.0002	0011	.0009	.0009
1759	2 30	454.96	.02	-4.62	-.0549	0089	.0125	.0001	-.0000	- C002	.0011	0009	.0008
1760	2 30	454.81	.02	-3.36	- 0149	.0104	.0083	.0001	.0000	-.C003	.0010	.0010	.0003
1761	2.30	455.14	.02	-2.11	.0257	.0117	.0039	.0000	0001	- C006	.0010	.0010	.0008
1762	2 30	454 84	.02	-.84	.0577	.0130	- 0007	.0000	.0001	-.C006	.0011	.0010	.0008
1763	2 30	455.14	.02	.49	.1102	.0139	- 0053	-.0001	.0001	-.0006	.0011	.0010	.0008
1764	2 30	455.08	.03	1.82	.1552	0151	- 0094	-.0001	.0001	-.C010	.0011	.0010	.0008
1765	2 30	454.90	.03	3.15	.1949	.0166	-.0125	-.0002	-.0000	-.C005	.0011	0010	.0008
1766	2 30	454 78	.03	4.49	.2346	.0184	-.0155	- 0004	.0001	- C011	.0011	.0010	0008
1767	2 30	455 34	.03	5 84	2750	.0202	- 0179	- 0004	.0001	-.0011	0011	0009	.0008
1768	2.30	455 11	.03	7 22	.3137	.0221	-.0199	-.0003	0001	-.C013	.0011	0009	.0008
1769	2 30	455.14	.02	-2.10	0262	.0117	.0039	.0000	.0001	-.C006	.0011	.0010	.0008

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NASA LANGLEY RESEARCH CENTER UPWT TS2 PRELIMINARY 01/14/69

STABILITY AXIS PRJ 927 RUN 93 MACH 2.30 CONFIG. CODE 0 BATCH 17

POINT	M CH	L/D	BETA	ALPHA	CL	CD	CM	CLS	CNS	CY	CEC	CDBN	CDI
1757	2 30	-6 0648	02	-7.18	- 1300	.0214	.0207	0001	0001	-.C005	.0010	0009	.0013
1758	2 30	-5.6222	.02	-5 88	-.0927	.0165	0168	.0001	0000	- C002	.0010	.0009	.0011
1759	2 30	-4 1125	.02	-4.62	- 0537	.0131	0125	.0001	- 0000	- C002	0010	.0009	0010
1760	2.30	-1 2630	02	-3 36	-.0140	0111	.0083	0001	0000	- C003	0010	.0010	.0009
1761	2 30	2 4552	.02	-2.11	.0263	.0107	.C039	0000	.0001	- C006	0010	.0010	.0008
1762	2 30	5 6689	.02	- 84	.0530	0120	-.0007	.0000	.0001	- C006	.0011	0010	.0008
1763	2 30	7 4359	.02	.49	.1100	.0148	-.0053	-.C001	.0001	-.C006	0011	0010	.C008
1764	2 30	7 7038	.03	1.82	.1545	.0200	-.0094	- 0001	.0001	-.C010	.0011	0010	.0008
1765	2 30	7.0237	C3	3 15	.1935	0273	-.0125	-.0002	-.0000	- C005	0011	0010	.0008
1766	2 30	6 3470	.03	4 49	2322	.0366	-.0155	-.000-	.0001	- C011	.0011	.0010	.0008
1767	2 30	5 6502	03	5 84	.2712	.0480	-.0179	-.C004	0001	- C011	.0011	0009	0009
1768	2 30	5 0407	03	7 22	3080	.0611	- 0199	- 0003	.0002	- CC13	.0011	.0009	0010
1769	2 30	2.5037	.C2	-2.10	.0268	.0107	.0039	0000	.0001	- C006	.0011	.0010	.0008

NASA LANGLEY RESEARCH CENTER UPWT TS2 PRELIMINARY 01/14/69

BODY AXIS PRJ 827 RUN 95 MACH 2.70 CONFIG. CODE 0 BATCH 17

POINT	MACH	DYN PRS	BETA	ALPHA	CN	CA	CM	CLB	CNB	CY	CAC	CABN	CAI
1784	2.70	413.65	.02	-7 25	-.1131	.0058	.0166	.0002	.0002	-.0002	.0008	.0007	.0007
1785	2.70	413 98	.02	-6.04	- 0818	.0071	.0135	.0001	.0002	-.0002	.0008	.0007	.0007
1786	2.70	413 49	.02	-4.80	-.0496	.0084	.0102	.0002	.0002	-.0002	0008	.0007	.0007
1787	2.70	413 41	.02	-3 62	- 0157	0097	.0064	.0001	.0002	- 0003	.0008	.0007	0007
1788	2.70	413.67	.02	-2.43	.0181	.0109	.0029	.0001	0002	-.0005	0008	.0007	.0007
1789	2.70	413.63	.02	-1.23	.0531	.0121	-.0009	.0001	.0002	- 0005	.0008	.0007	.0007
1790	2.70	413.47	.02	.04	.0899	.0132	-.0047	-.0000	.0002	-.0004	.0008	.0007	.0007
1791	2.70	413.49	.02	1.27	.1248	.0144	-.0079	-.0000	.0002	-.0005	.0008	.0007	.0007
1792	2.70	413.78	.02	2.56	.1592	.0157	-.0106	-.0001	.0002	-.0007	.0008	.0007	.0007
1793	2.70	413.71	.02	3 81	.1925	.0173	-.0128	-.0002	.0001	-.0004	.0009	.0007	.0007
1794	2.70	413.47	.02	5.05	.2248	.0187	-.0149	-.0003	.0001	-.0005	.0009	.0007	.0007
1795	2.70	413.54	.03	6 33	.2563	.0202	-.0166	-.0003	.0001	- 0010	.0009	.0007	.0007
1796	2.70	413.65	.03	7.62	.2899	.0218	-.0181	-.0003	.0002	-.0009	.0009	.0006	.0007
1797	2.70	413.58	.03	8 94	.3241	.0234	-.0194	-.0003	.0002	- 0013	0009	.0006	.0007
1798	2.70	413.71	.02	-2.42	.0190	.0109	.0027	.0001	.0002	-.0005	.0008	.0007	.0007

NASA LANGLEY RESEARCH CENTER UPWT TS2 PRELIMINARY 01/14/69

STABILITY AXIS PRJ 827 RUN 95 MACH 2.70 CONFIG. CODE 0 BATCH 17

POINT	MACH	L/D	BETA	ALPHA	CL	CD	CM	CLS	CNS	CY	CDC	CDBN	CDI
1784	2.70	-5.6944	.02	-7 25	-.1111	.0195	.0166	.0002	.0002	-.0002	.0008	.0007	.0012
1785	2.70	-5.2340	.02	-6 04	-.0802	.0153	.0135	.0001	.0002	-.0002	.0008	.0007	.0011
1786	2.70	-3 9475	.02	-4.80	-.0485	.0123	.0102	.0001	.0002	-.0002	.0008	.0007	.0010
1787	2.70	-1 4038	.02	-3 62	-.0148	0106	0064	.0001	.0002	-.0003	.0008	.0007	.0009
1788	2.70	1.8640	.02	-2.43	.0187	.0100	.0029	.0001	.0002	-.0005	.0008	0007	.0008
1789	2.70	4.9055	.02	-1.23	.0534	.0109	-.0009	.0001	.0002	-.0005	.0008	.0007	.0008
1790	2.70	6.7699	.02	.04	.0899	.0133	-.0047	-.0000	.0002	- 0004	.0008	.0007	.0007
1791	2.70	7.2606	.02	1.27	1244	.0171	-.0079	.0000	.0002	-.0005	.0008	.0007	.0007
1792	2.70	6 9313	.02	2.56	.1582	.0228	-.0106	-.0001	.0002	- 0007	.0008	.0007	0007
1793	2.70	6 3591	02	3 81	.1907	.0300	-.0128	-.0002	.0001	-.0008	0009	.0007	.0008
1794	2.70	5 7519	.02	5 05	.2220	.0383	-.0149	- 0003	.0001	-.0005	.0009	0007	.0008
1795	2 70	5 2388	03	6 33	.2522	.0481	-.0166	-.0003	.0001	-.0010	.0009	0007	0009
1796	2.70	4 7577	03	7.62	.2841	.0597	- 0181	- 0003	.0002	-.0009	.0009	.0006	0010
1797	2 70	4.3268	.03	8.94	.3161	.0731	-.0194	-.0003	.0002	-.0013	.0009	0006	0011
1798	2.70	1.9488	.02	-2 42	.0195	.0101	.0027	.0001	.0002	-.0005	.0008	.0007	.0008

NASA LANGLEY RESEARCH CENTER UPWT TS2 PRELIMINARY 01/14/69

BODY AXIS PRJ 827 RUN 146 MACH 2.30 CONFIG. CODE 0 BATCH 33

POINT	MACH	DYN PRS	BETA	ALPHA	CN	CA	CM	CLB	CNB	CY	CAC	CABN	CAI
2969	2 30	452.74	-.00	-7 35	-.1295	0041	0158	0002	.0001	- 0001	0010	.0009	.0008
2970	2 30	453 21	-.00	-6 02	- 0913	.0062	.0129	0002	.0001	-.0000	.0010	0009	.0008
2971	2 30	453 30	00	-4.71	- 0519	0080	0095	0001	.0001	-.0001	.0010	.0009	0008
2972	2 30	453 89	-.00	-3 44	-.0136	0095	0060	0001	.0000	0000	.0010	.0010	0008
2973	2.30	453 60	00	-2.18	.0255	0109	0025	0000	0000	-.0001	0010	0010	.0008
2974	2 30	453 92	.00	- 87	0680	.0121	- 0015	- 0000	.0000	-.0001	.0010	.0010	.0000
2975	2 30	453.98	.00	.50	.1108	.0130	- 0052	-.0001	0000	-.0000	0010	.0010	.0008
2976	2.30	454 45	.00	1.87	1556	.0145	-.0084	- 0000	- 0000	-.0002	.0011	.0010	0008
2977	2 30	454 81	.01	3 21	.1934	0158	-.0106	-.0002	- 0001	-.0003	.0011	.0010	.0008
2978	2 30	454 01	.01	4 65	2354	0177	- 0121	- 0003	-.0001	-.0003	0011	.0010	0008
2979	2 30	454 54	.01	6.01	.2723	.0194	- 0132	- 0003	- 0001	-.0007	0011	0009	.0008
2980	2.30	454 93	02	7.43	.3119	.0213	-.0145	- 0001	-.0002	-.0011	.0011	.0009	.0008
2981	2.30	454.19	.02	8 80	.3504	.0232	-.0159	0000	-.0003	-.0015	.0011	0009	.0008
2982	2 30	455 28	.00	-2 17	.0257	0108	.0025	0000	0000	- 0001	.0010	.0010	0008

NASA LANGLEY RESEARCH CENTER UPWT TS2 PRELIMINARY 01/14/69

STABILITY AXIS PRJ 827 RUN 146 MACH 2 30 CONFIG. CODE 0 BATCH 33

POINT	MACH	L/D	BETA	ALPHA	CL	CD	CM	CLS	CNS	CY	CDC	CDBN	CDI
2969	2 30	-6 3288	- 00	-7 35	-.1275	.0201	.0158	0001	0001	- 0001	.0010	0009	0013
2970	2 30	-5 8193	- 00	-6.02	- 0898	0154	0129	0002	0001	- 0000	0010	0009	0011
2971	2 30	-4 2109	00	-4.71	- 0508	.0121	0095	0001	0001	- 0001	0010	0009	0010
2972	2 30	-1 2492	- 00	-3 44	-.0127	0102	0060	0001	0000	.0000	.0010	0010	0009
2973	2 30	2 6437	.00	-2.18	0260	.0099	0025	0000	0000	- 0001	0010	.0010	.0009
2974	2 30	6 1893	.00	-.87	0683	0110	- 0015	- 0000	.0000	- 0001	0010	.0010	.0008
2975	2.30	7.9039	.00	.50	.1107	.0140	-.0052	-.0001	.0000	-.0000	.0010	0010	.0008
2976	2 30	7 9265	.00	1.87	1549	.0195	- 0084	- 0000	- 0000	-.0002	.0011	.0010	.0008
2977	2.30	7 2119	.01	3 21	.1920	.0266	- 0106	- 0002	-.0001	- 0003	.0011	0010	0008
2978	2 30	6 3646	01	4 65	2329	.0366	- 0121	- 0003	- 0001	-.0003	.0011	.0010	0009
2979	2.30	5 6347	.01	6.01	.2685	0476	-.0132	- 0003	-.0001	-.0007	.0011	0009	.0009
2980	2 30	5.0016	.02	7 43	3061	.0612	- 0145	- 0002	- 0002	-.0011	0011	0009	0010
2981	2 30	4 4929	02	8 80	3422	.0762	- 0159	- 0000	- 0003	- 0015	.0011	0009	.0012
2982	2.30	2.6759	.00	-2 17	0262	.0098	0025	0000	.0000	- 0001	.0010	.0010	.0009

NASA LANGLEY RESEARCH CENTER UPWT TS2 PRELIMINARY 01/14/69

BODY AXIS PRJ 827 RUN 148 MACH 2 70 CONFIG CODE 0 BATCH 33

POINT	MACH	DYN PRS	BETA	ALPHA	CN	CA	CM	CLB	CNB	CY	CAC	CAB4	CAI
2998	2 70	414 23	-.01	-7 43	-.1113	0044	.0119	.0003	0003	.0001	.0008	.0007	.0007
2999	2.70	415.07	- 01	-6.18	-.0898	0059	.0098	.0003	.0003	.0001	0008	.0007	0007
3001	2 70	413 87	-.01	-4 91	-.0494	.0073	.0073	.0002	.0003	.0001	.0008	.0007	0007
3002	2 70	413 76	-.01	-3 70	-.0159	.0088	0045	0002	0002	0002	.0008	.0007	0007
3003	2 70	413 89	-.00	-2.49	0172	.0100	0016	0002	.0002	.0000	0008	.0007	0007
3004	2 70	413 84	- 00	-1 27	0518	.0 12	- 0014	.0001	0002	-.0001	0008	0007	0007
3005	2 70	414 09	-.00	0	0881	0123	- 0046	0001	0002	0001	0008	0007	0007
3006	2 70	413 95	-.00	1 32	1230	0135	-.0070	0001	.0001	-.0000	.0008	.0007	0007
3007	2 70	414.15	-.00	2 60	.1561	0149	- 0086	.0000	.0000	0001	.0009	0007	.0007
3008	2.70	414.11	00	3 86	.1836	.0163	- 0101	- 0000	- 0001	-.0001	.0009	.0007	.0007
3009	2 70	413.60	.00	5 16	2219	.0178	-.0113	-.0001	-.0001	-.0000	.0009	.0007	.0007
3110	2 70	414 09	.01	6 43	2530	.0193	- 0121	-.0001	-.0001	-.0003	0009	.0007	.0007
3111	2.70	413 67	01	7.73	.2852	.0209	- 0130	- 0002	-.0001	- 0005	.0009	.0006	0007
3112	2.70	413 84	.01	9 11	.3184	0225	- 0137	-.0001	- 0001	- 0010	.00 9	.0006	0007
3113	2.70	413 98	01	10.45	.3521	.0241	-.0144	-.0002	.0000	- 0008	.0009	0006	.0007
3114	2 70	414.02	.00	11.78	.3879	0259	- 0154	-.0003	0001	- 0004	.0009	0005	.0007
3116	2 70	414.19	- 00	-2.49	.0181	.0100	.0015	.0002	0002	.0001	.0008	.0007	.0007

NASA LANGLEY RESEARCH CENTER UPWT TS2 PRELIMINARY 01/14/69

STABILITY AXIS PRJ 827 RUN 148 MACH 2.70 CONFIG CODE 0 BATCH 33

POINT	MACH	L/D	BETA	ALPHA	CL	CD	CM	CLS	CNS	CY	CDC	CDBN	CDI
2998	2.70	-6.0039	- 01	-7 43	- 1099	.0183	.0119	0002	.0003	0001	.0008	.0007	.0013
2999	2.70	-5 6129	-.01	-6.18	- 0794	0141	.0098	.0002	.0003	.0001	.0008	0007	.0011
3001	2 70	-4 2904	-.01	-4.91	-.0483	0113	.0073	.0002	.0003	.0001	.0008	0007	0010
3002	2.70	-1.5681	-.01	-3 70	-.0151	0096	.0045	0001	.0003	.0002	0008	0007	.0009
3003	2 70	1 9427	-.00	-2 49	.0173	0092	.0016	.0002	.0002	.0000	.0008	.0007	.0008
3004	2 70	5 2283	-.00	-1 27	0521	.0100	-.0014	.0001	.0002	-.0001	.0008	.0007	.0008
3005	2.70	7.1282	-.00	04	0881	.0124	- 0046	.0001	0002	.0001	.0008	.0007	.0007
3006	2.70	7.4959	-.00	1.32	.1226	.0164	-.0070	0001	.0001	-.0000	.0008	0007	.0007
3007	2.70	7 0789	-.00	2 60	1551	.0219	-.0086	.0000	.0000	.0001	.0009	0007	.0007
3008	2 70	6 4557	.00	3.56	1869	0290	-.0101	- 0000	- 0001	-.0001	0009	.0007	0008
3009	2 70	5 8232	.00	5 16	.2191	.0376	- 0113	- 0001	- 0001	-.0000	.0009	.0007	.0008
3110	2 70	5 2347	.01	6 48	2499	0476	- 0121	- 0001	-.0001	- 0003	.0009	.0006	.0009
3111	2 70	4 7369	.01	7.78	.2804	.0592	- 0130	- 0002	- 0001	-.0005	0009	.0006	.0010
3112	2.70	4 3029	.01	9 11	3104	0721	-.0137	-.0001	- 0001	-.0010	.0009	.0006	.0012
3113	2.70	3 9262	.01	10 45	.3414	.0870	-.0144	-.0002	.0001	-.0008	.0009	.0006	.0014
3114	2.70	3.6062	.00	11.78	3740	.1037	-.0154	-.0003	.0002	-.0004	.0009	0005	0016
3116	2.70	2 0418	- 00	-2.49	.0187	.0091	.0015	.0002	.0002	.0001	.0008	.0007	.0008

NASA LANGLEY RESEARCH CENTER UPWT TS2 PRELIMINARY 01/14/69

BODY AXIS PRJ 827 RUN 150 MACH 2 30 CONFIG. CODE 0 BATCH 34

POINT	MACH	DYN PRS	BETA	ALPHA	CN	CA	CM	CLB	CNB	CY	CAC	CAB4	CAI
3150	2 30	453 68	.00	-7.27	- 1366	.0038	0178	.0092	-.0001	.0004	.0010	0 0000	0 0000
3151	2.30	453.62	.00	-5 93	-.0990	.0053	.0155	.0002	-.0001	.0005	.0010	0 0000	0 0000
3152	2 30	453.60	.00	-4.59	-.0507	.0068	.0127	0001	- 0001	.0003	0010	0 0000	0 0000
3153	2.30	454.39	.00	-3.35	- 0239	.0081	.0097	.0001	- 0001	.0001	.0010	0 0000	0 0000
3154	2.30	454.87	.00	-2 04	.0165	.0094	.0061	.0001	-.0000	.0003	0010	0 0000	0 0000
3155	2 30	453 75	00	-.75	.0573	.0105	.0025	0001	-.0001	.0000	0010	0 0000	0 0000
3156	2 30	454 07	00	.67	.1026	0114	-.0012	-.0001	-.0001	0004	0010	0 0000	0 0000
3157	2 30	454.31	.00	1 99	1431	0124	-.0039	- 0001	-.0001	.0001	.0011	0 0000	0 0000
3158	2 30	454 51	01	3 35	.1817	0136	- 0058	- 0001	- 0002	0000	0011	0 0000	0 0000
3159	2.30	454 75	.01	4.76	2202	.0151	-.0066	-.0003	- 0002	- 0001	.0011	0 0000	0.0000
3160	2 30	454 90	01	6 20	2588	.0166	-.0074	-.0003	-.0002	- 0004	0010	0 0000	0.0000
3161	2 30	453.95	.02	7.59	.2956	0182	- 0080	-.0001	- 0002	-.0007	0010	0.0000	0 0000
3162	2 30	453.95	02	8.50	.3212	.0194	-.0085	-.0001	-.0003	-.0010	.0010	0 0000	0 0000
3163	2 30	454 87	.00	-2.04	.0162	.0093	.0061	.0001	-.0001	.0003	.0010	0.0000	0 0000

NASA LANGLEY RESEARCH CENTER UPWT TS2 PRELIMINARY 01/14/69

STABILITY AXIS PRJ 827 RUN 150 MACH 2.30 CONFIG. CODE 0 BATCH 34

POINT	MACH	L/D	BETA	ALPHA	CL	CD	CM	CLS	CNS	CY	CDC	CDBN	CDI
3150	2.30	-6 4213	.00	-7.27	-.1350	.0210	.0178	.0002	-.0001	.0004	.0010	0 0000	0 0000
3151	2 30	-6 3273	.00	-5 93	- 0980	0155	0155	.0002	-.0001	0005	.0010	0 0000	0 0000
3152	2 30	-5 1453	.00	-4.59	- 0600	.0117	0127	0001	- 0001	0003	.0010	0 0000	0 0000
3153	2 30	-2 4607	.00	-3 35	- 0234	0095	.0097	0001	- 0001	0001	.0010	0.0000	0 0000
3154	2 30	1 7203	00	-2 04	0168	0088	0061	000	-.0000	0003	.0010	0 0000	0 0000
3155	2 30	5 9096	.00	-.75	.0574	.0097	0025	0001	-.0001	0000	.0010	0 0000	0 0000
3156	2 30	8 1500	.00	.67	.1025	0126	- 0012	-.0001	- 0001	.0004	0010	0 0000	0 0000
3157	2 30	8.2145	.00	1.99	.1426	.0174	- 0039	- 0001	-.0001	.0001	.0011	0 0000	0 0000
3158	2.30	7.4539	01	3 35	.1806	.0242	- 0059	- 0001	-.0001	0000	.0011	0 0000	0 0000
3159	2.30	6 5470	.01	4.76	.2182	.0333	-.0066	-.0003	-.0001	- 0001	.0011	0 0000	0 0000
3160	2.30	5 7439	.01	6.20	.2555	0445	- 0074	- 0003	-.0002	-.0004	.0010	0 0000	0 0000
3161	2 30	5 0865	.02	7 59	2906	0571	- 0080	- 0002	- 0002	- 0009	0010	0 0000	0 0000
3162	2.30	4 7219	.02	8.50	.3148	.0667	-.0085	-.0001	-.0003	- 0010	.0010	0 0000	0.0000
3163	2 30	1.9018	.00	-2.04	0165	.0087	.0061	.0001	-.0001	0003	.0010	0.0000	0 0000

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NASA LANGLEY RESEARCH CENTER UPWT TS2 PRELIMINARY 01/14/69

BODY AXIS PRJ 827 RUN 152 MACH 2 70 CONFIG. CCOE 0 BATCH 34

POINT	MACH	DYN PRS	BETA	ALPHA	CN	CA	CM	CLB	CNB	CY	CAC	CABN	CAI
3179	2.70	414.46	-.00	-7 34	- 1178	.0040	0140	.0003	.0001	.C002	.CCC8	C CC00	0 0000
3180	2.70	413 60	-.00	-6 07	-.0865	.0052	0122	0002	.00C1	C005	.00C8	0 0000	0 0000
3181	2.70	414 02	- 00	-4 79	- 0547	.0064	.0101	.0003	00C1	CC05	.00C9	0 CCCC	C 0000
3182	2.70	414 19	- 00	-3 60	-.0235	.0076	.0079	.0001	.0000	.C003	.00C8	C 0000	0 CC00
3183	2.70	414.15	-.00	-2.35	.0097	.0026	.0052	.0001	.0001	.CC03	.0CC8	C CCC0	0 0000
3184	2.70	414.02	.00	-1.11	0.41	C098	.0025	0001	.0000	.0002	.CCC8	0 0000	C 0000
3185	2.70	414.13	00	.16	0793	.0107	-.0004	CC00	.0000	.C001	.0CC8	0 0000	C 0000
3186	2.70	414 44	.C0	1 45	.1138	.0118	-.0025	.0000	- 0000	-.0000	.0CC8	C C000	C CC00
3187	2.70	414 26	.00	2 76	.1465	0128	- 0039	- 0000	-.0CC0	.CCC0	0CC9	0 0000	0 0000
3188	2 70	414 24	C0	4 04	1787	.0140	-.0050	0000	- 0001	CC01	CCC9	0 0000	0 0000
3189	2 70	414 04	.01	5 34	.2109	0153	- 0059	- 0001	- 0002	- C004	0CC9	0 0000	0 0000
3190	2.70	414 37	.01	6 65	2406	.0165	- 0056	- 0002	- 0001	- C002	.CCC9	C C000	0 0000
3191	2.70	414.19	.01	7.98	2728	.0179	-.0070	- 0002	- 0002	- C007	.0CC9	C 0000	0 0000
3192	2.70	414 46	.01	9.29	.3041	0192	-.0072	-.0001	- 0002	- CC06	.CCC9	C CC00	0 0000
3193	2.70	414.11	.02	10 63	.3375	.0207	-.0075	-.0002	-.0002	-.CC08	.0CC9	0 C000	0 0000
3194	2.70	414 44	.01	11.18	.3521	.0213	- 0080	-.0002	-.0001	- C0C8	.0CC9	0 0000	0 0000
3195	2.70	414.48	.00	-2.35	.0099	.0087	.0052	.0001	.0001	.0001	.00C8	C.CC00	C C000

NASA LANGLEY RESEARCH CENTER UPWT TS2 PRELIMINARY 01/14/69

STABILITY AXIS PRJ 827 RUN 152 MACH 2.70 CONFIG. CCOE 0 BATCH 34

POINT	MACH	L/D	BETA	ALPHA	CL	CD	CM	CLS	CNS	CY	CCC	CC8N	CDI
3179	2.70	-6 1015	- 00	-7 34	- 1163	.0191	0140	.0003	.0001	.CCC2	.0CC8	C C090	0 0000
3180	2.70	-5.9510	- 00	-6 07	- 0855	.0144	.0122	0002	.0001	C005	00C9	C 0000	0 0000
3181	2 70	-4 9123	-.00	-4.79	-.0539	.0110	0101	.0003	.00C1	.C0C5	.0C08	C CCC0	0 0000
3182	2 70	-2.5462	-.00	-3 60	- 0230	.0090	.0079	0001	.0001	.0003	.0CC8	0 0CC0	0 C000
3183	2.70	1.2203	- 00	-2 35	0100	.0082	.00 2	CC01	.00C1	.CCC3	.0CC8	C 0000	0 0000
3184	2.70	4 9636	.00	-1 11	.C442	0089	.0025	.0001	.0000	.0002	.00C8	C C000	C C000
3185	2.70	7 2371	00	16	.0793	0110	- 00C4	.0000	.0000	C001	0C08	0.0000	0 0000
3186	2 70	7.7521	C0	1 45	.1134	0146	-.0025	0000	-.0000	-.C000	0CC8	C CC00	C 0000
3187	2 70	7 3351	.00	2 76	.1458	.0199	- 0039	- 0C00	- 0000	0000	.0CC9	0 0000	0 C000
3188	2 70	6.6625	.00	4 04	.1773	.0266	- 0050	.0C00	- 0001	.CCC1	.0CC9	C C000	0 0000
3189	2 70	5.9790	.01	5.34	.2086	.0349	-.0059	-.0001	- 0001	-.C004	.0CC9	0.000C	0 0000
3190	2.70	5.3569	.01	6.65	.2370	.0442	- 0056	-.0002	- 00C1	- CCC2	.0CC9	0 0000	0 0000
3191	2.70	4 8173	.C1	7.98	.2677	0556	- 0070	- 0002	- 0001	-.C007	.0CC9	C 0000	0 C000
3192	2.70	4 3621	01	9.29	.2970	0691	- 0072	- 0C02	-.0002	-.C006	.0CC9	0.0000	0 0000
3193	2 70	3 9703	.02	10 63	.3279	.0326	-.0075	- 0002	-.0002	-.CCC8	CCC9	0.C000	0 0000
3194	2 70	3 8285	.01	11.18	3413	0891	- 0080	- 0003	- 0001	-.C008	.0009	0 0000	0 0000
3195	2.70	1.2455	.00	-2 35	.0103	.0083	.0052	.0001	.0001	.C001	.0CC8	C.C000	C 0000

TABLE I. - EXTRAPOLATION OF SCAT 15F-9898
MODEL DATA TO FULL SCALE

	Drag Increment
Skin Friction ($C_{D_{F\ Mod}}$ = 00765, $C_{D_{F\ A/P}}$ = .00379)	-.00386
Roughness Drag	+.00023
Wave Drag ($C_{D_{W\ Mod}}$ = .00110, $C_{D_{W\ A/P}}$ = .00122)	+.00012
Grit Drag	- 00040
TOTAL	-.00391

TABLE II. - COMPONENT WETTED AREAS AND CONFIGURATION SKIN FRICTION COEFFICIENTS.

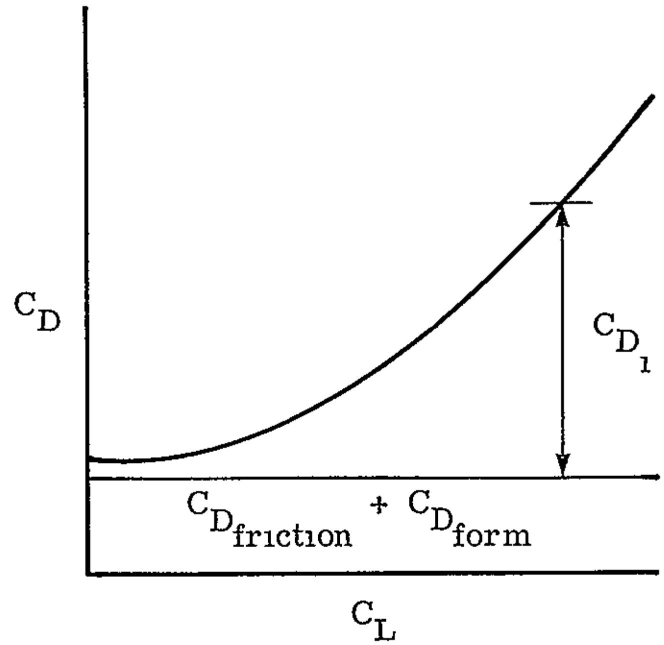
S_{ref} = 781.71 m^2 (8415 ft^2)

Component	Wetted Area, m^2 (ft^2)
Wing	1414.72 (15227.88)
Fuselage	734.06 (7901.34)
Nacelles (4)	250.44 (2695.70)
Wing Fins (2)	73.58 (791.98)
Vertical Tail	73.96 (796.11)
Horizontal Tail	93.30 (1004.30)
TOTAL	2460.06 (28417.31)

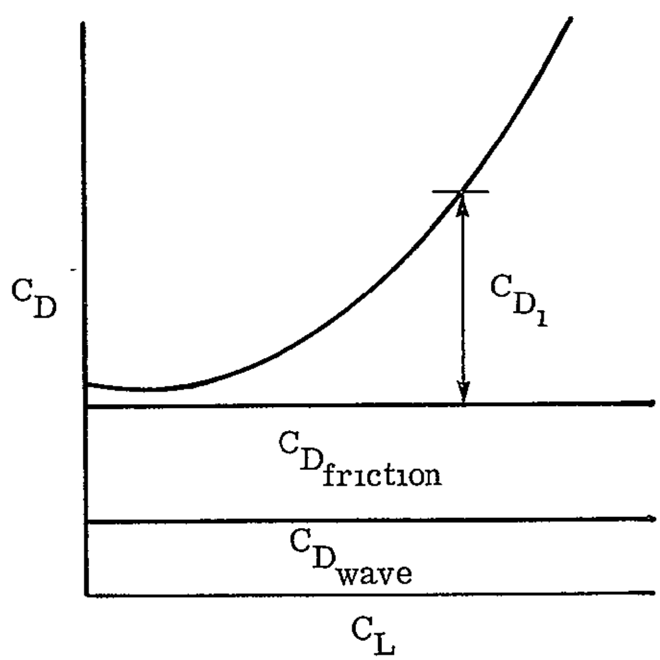
Mach Number	Altitude, m (ft)	$C_{D_{friction}}$
1.2	10375.4 (34040.)	.005480
2.0	14703.6 (48240.)	.004750
2.62	17952.7 (58900.)	.004270

TABLE III. - HORIZONTAL TAIL INCIDENCE ANGLES REQUIRED FOR MAXIMUM CONFIGURATION PERFORMANCE

M_∞	i_{tail} ~ DEG
1.1	5.
1.2	4.
1.3	4.
1.4	4.
1.5	5.
1.6	5.
1.8	5.
2.0	4.
2.2	4.
2.4	5.
2.62	4.

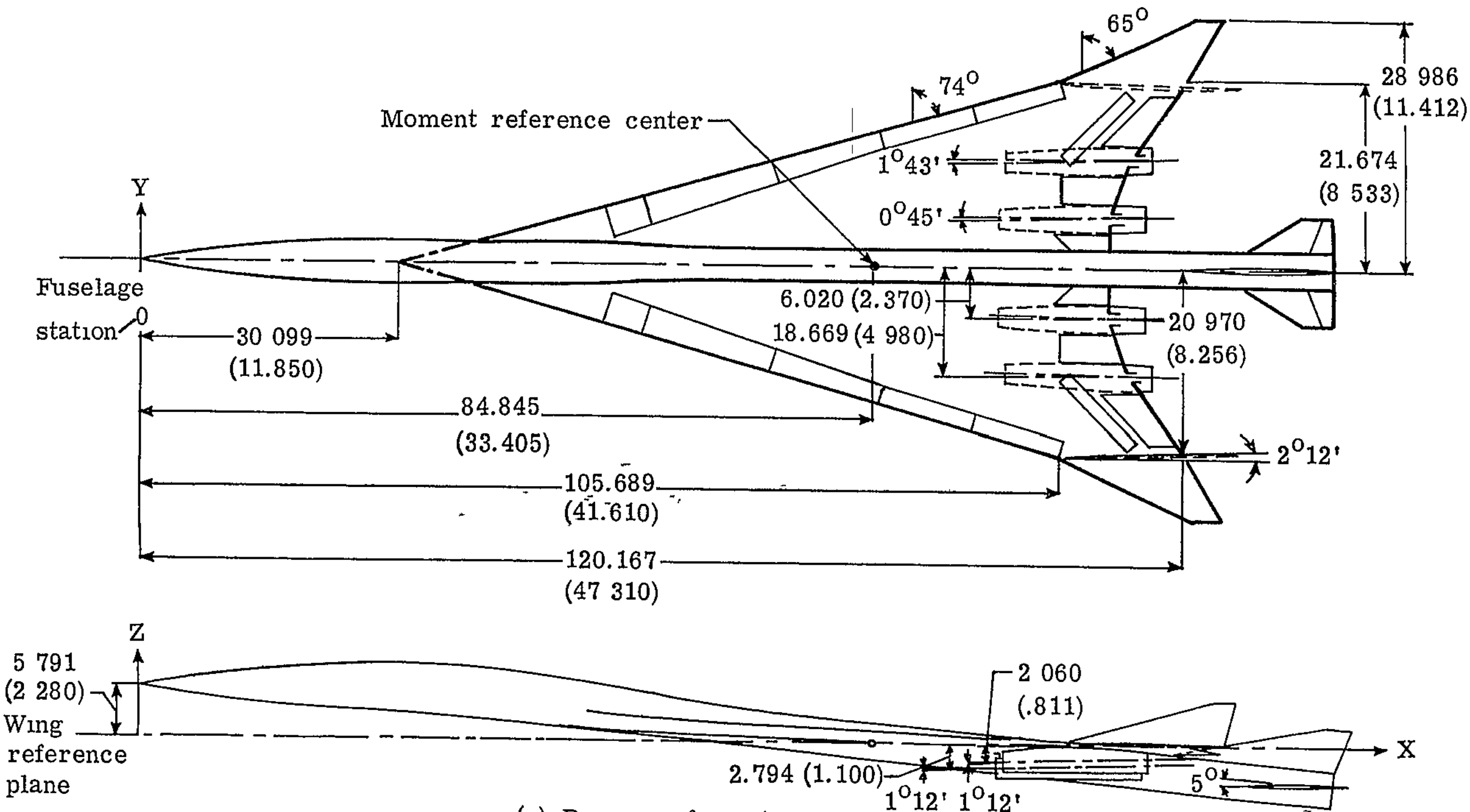


(a) Subsonic



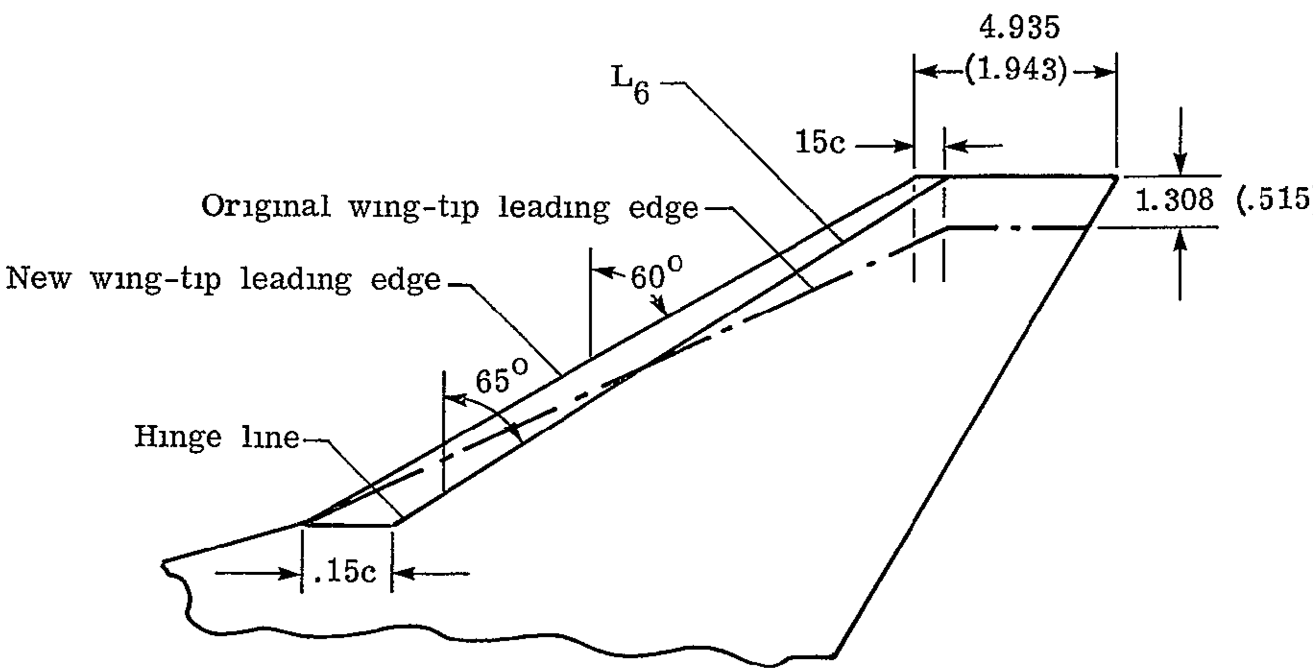
(b) Supersonic.

Figure 1.- Drag buildup procedure.

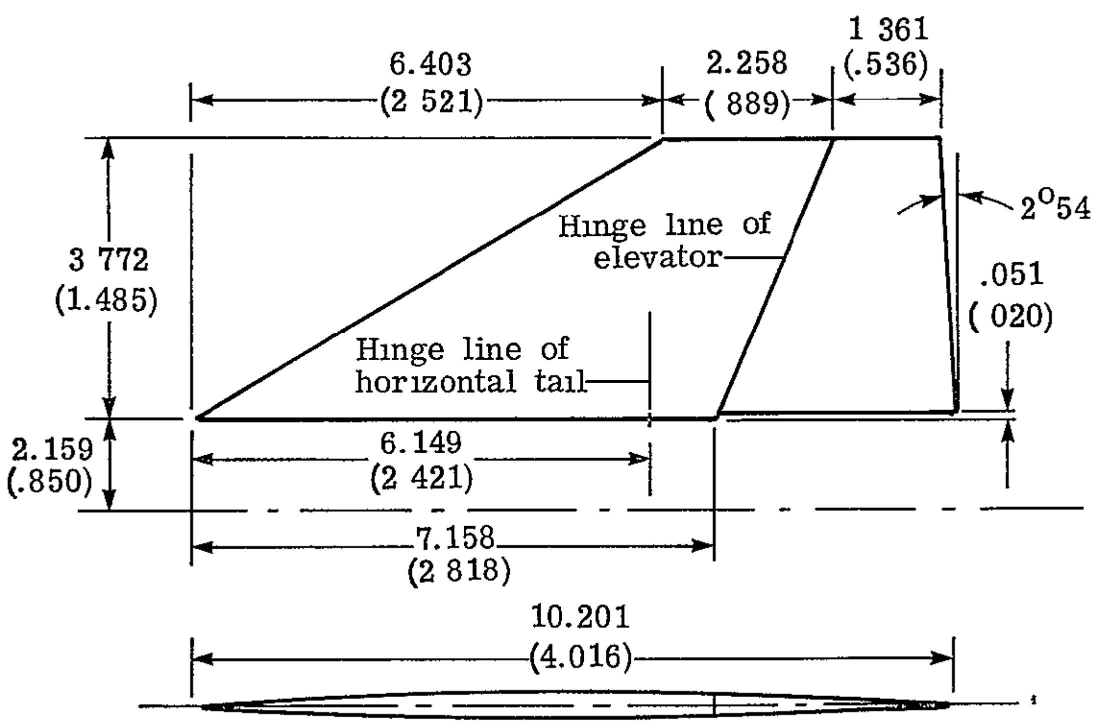


(a) Basic configuration.

Figure 2 - Details of model.

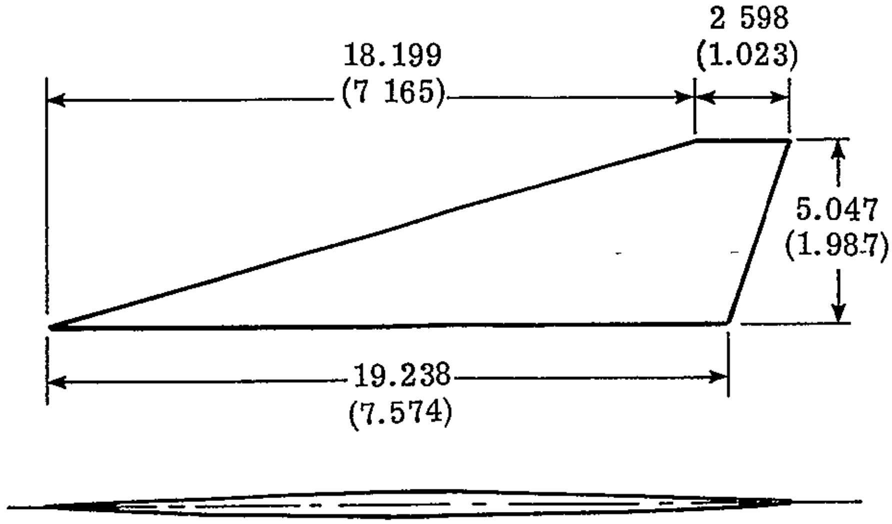


(b) Modifications to wing tip

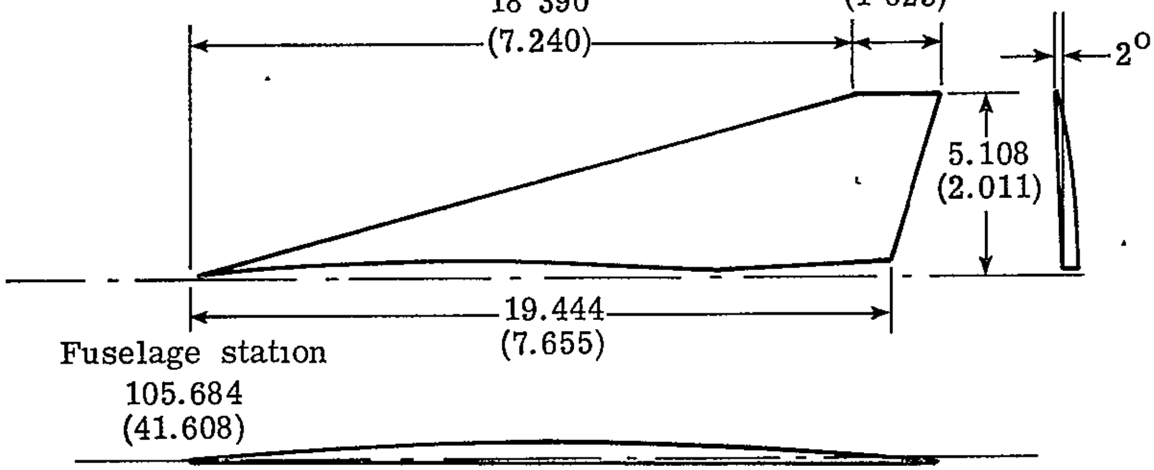


(c) Horizontal tail and elevator

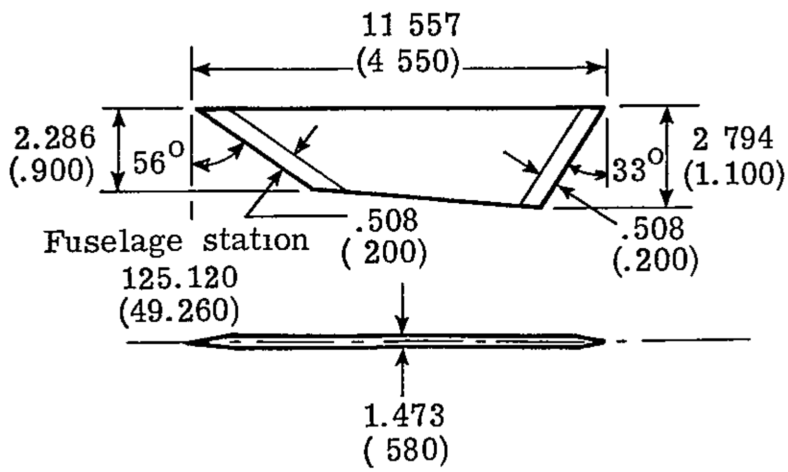
Figure 2. - Continued



(d) Center line vertical tail

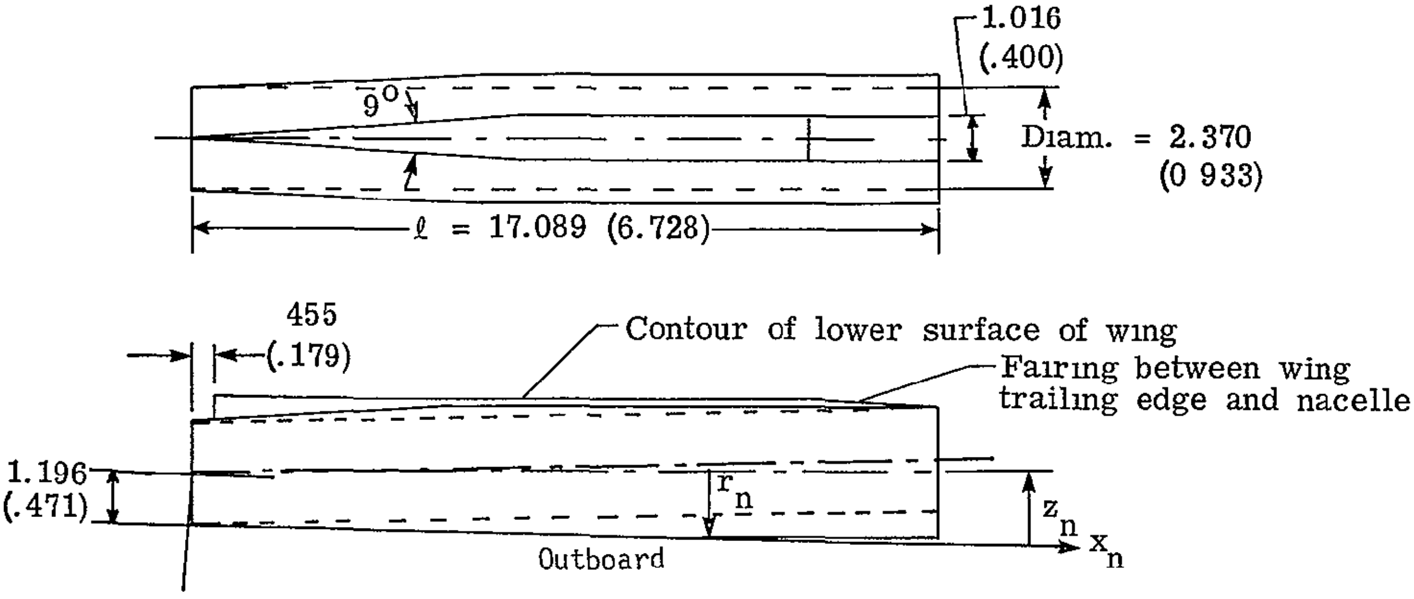


(e) Outboard vertical tails

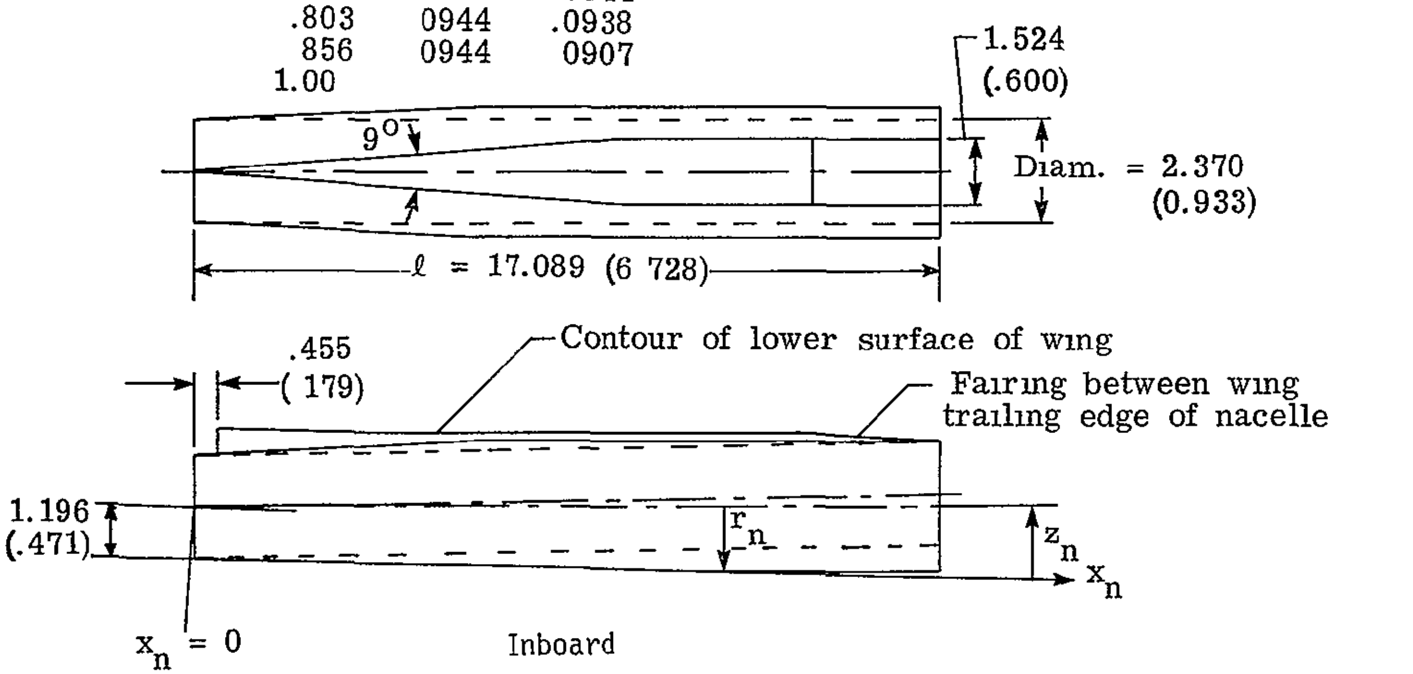


(f) Center line ventral fin

Figure 2. - Continued

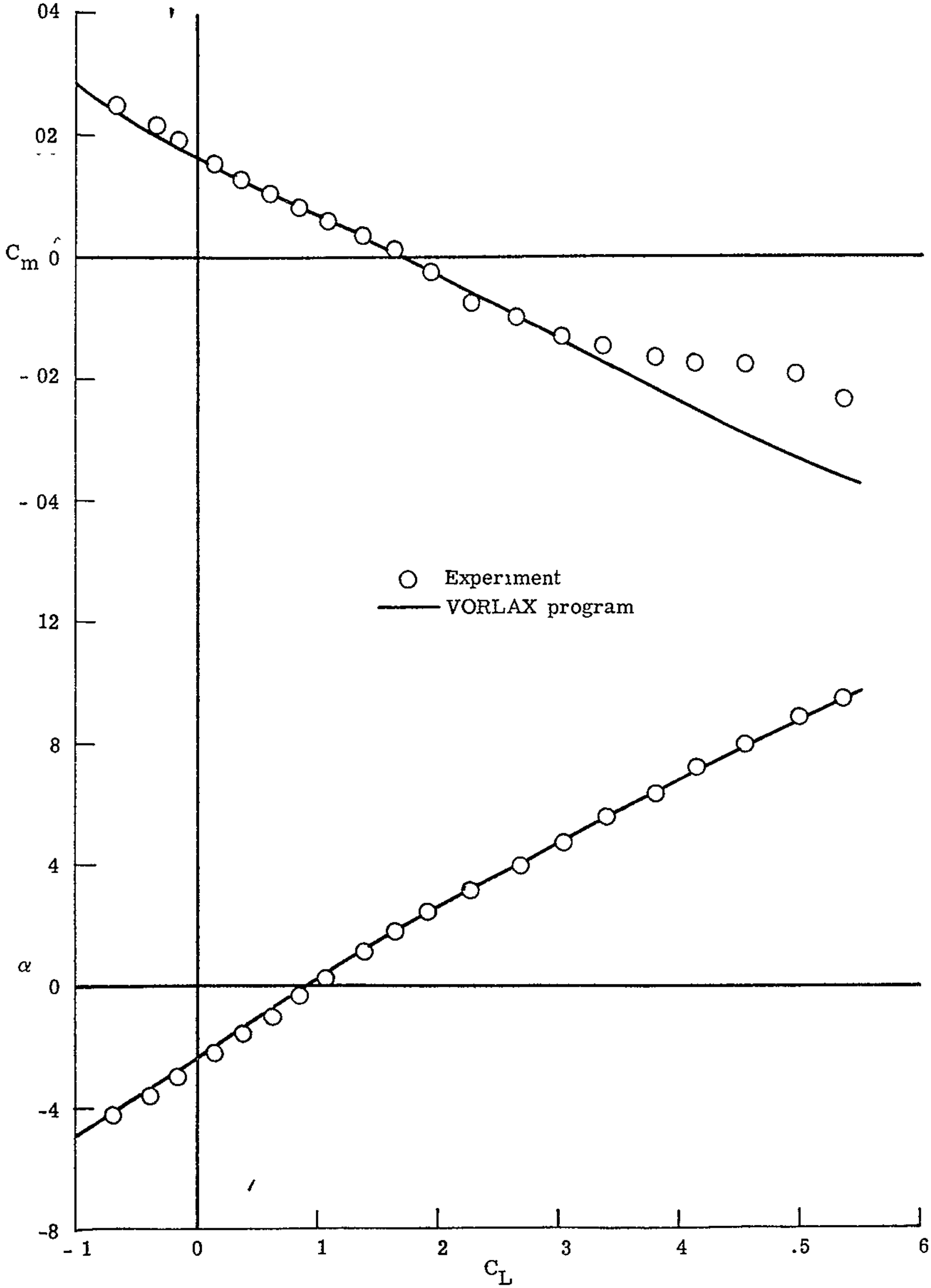


x_n/ℓ	z_n/ℓ	r_n/ℓ
0	0702	.0702
321	.0841	0841
428	.0883	.0883
.535	0914	.0914
.642	.0945	.0945
.749	0944	0944
.803	0944	.0938
856	0944	0907
1.00		



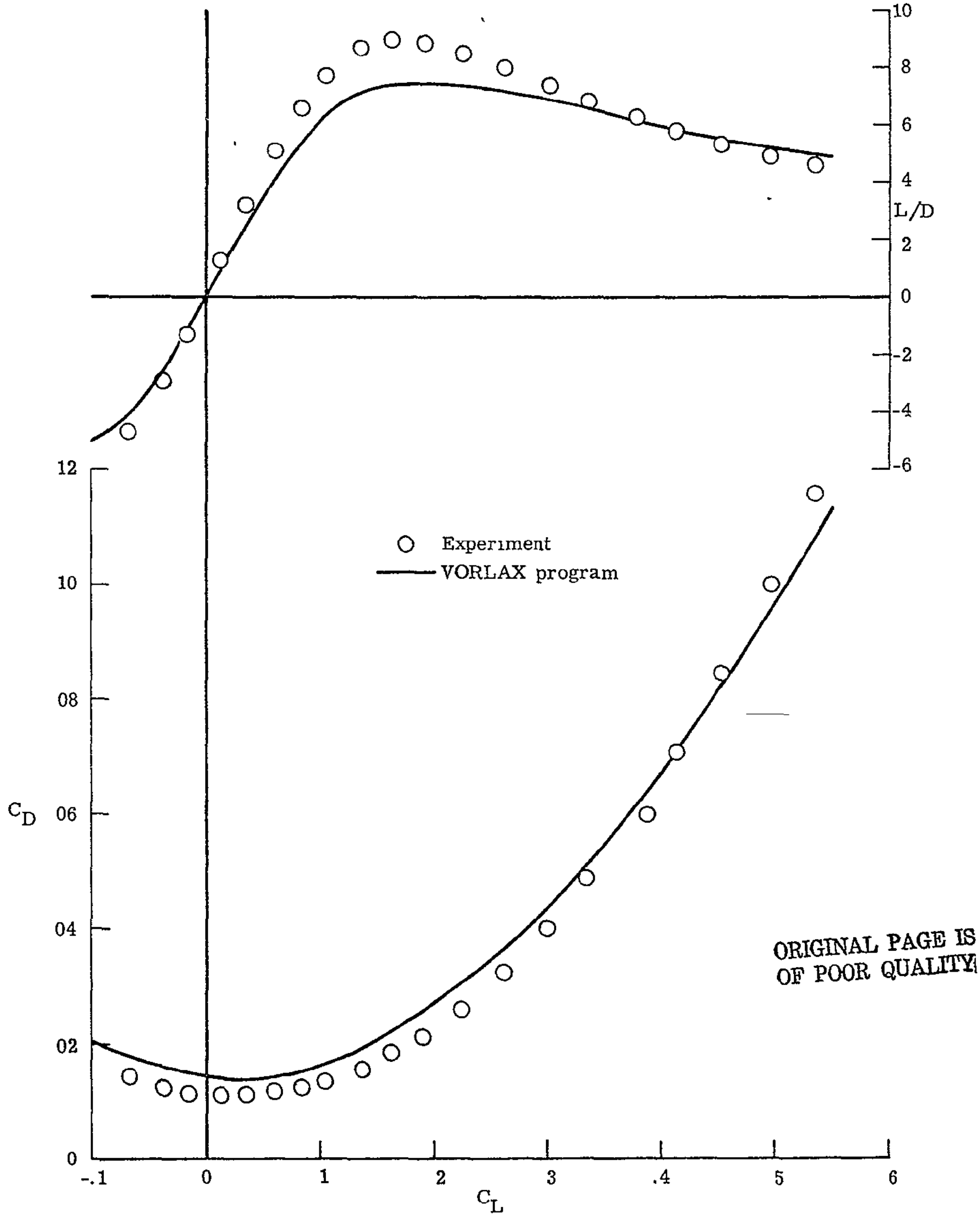
(g) Engine nacelles

Figure 2. - Concluded



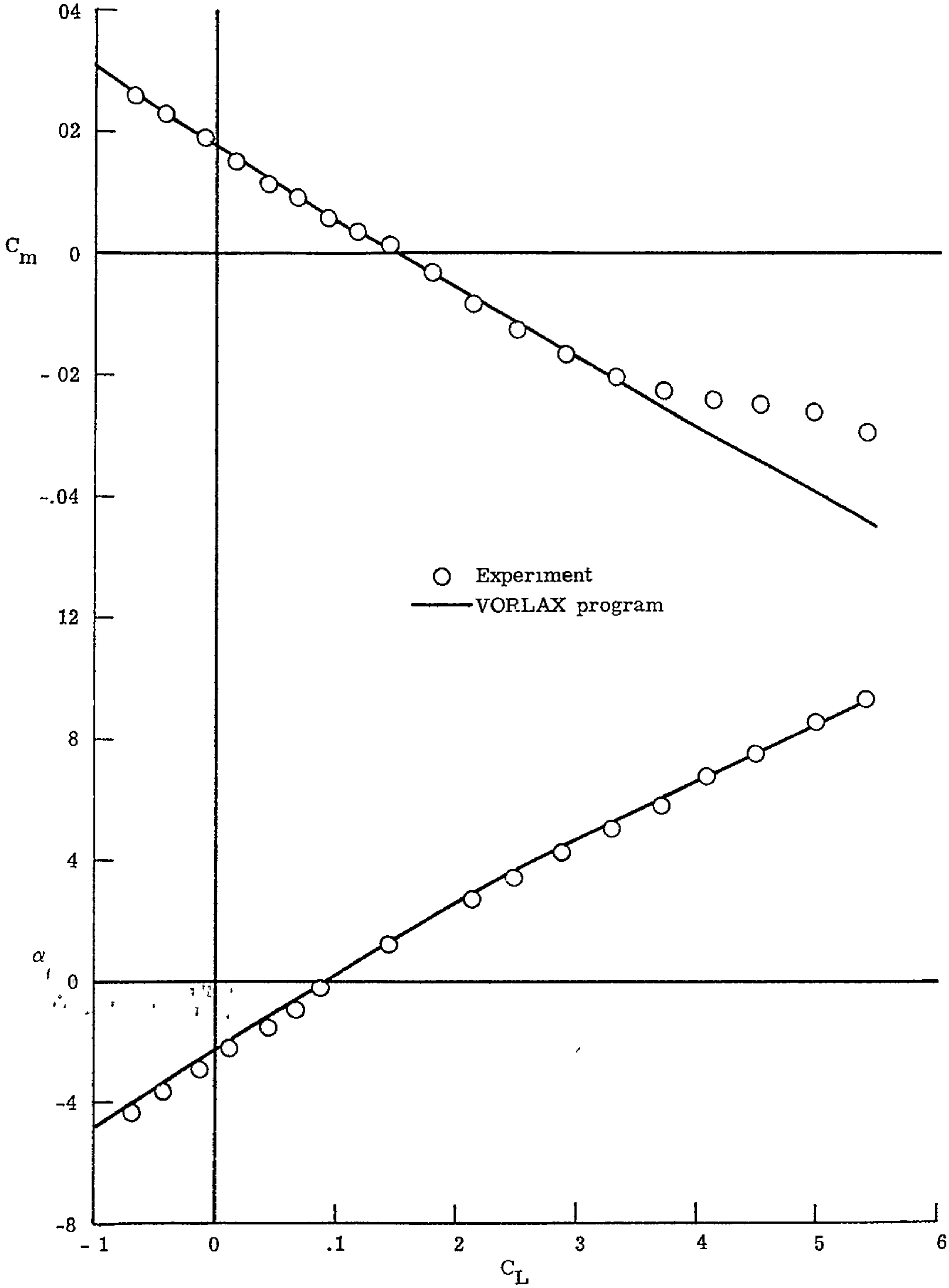
(a) M = 0 80

Figure 3 - Comparison of thoretical and experimental results for complete model at subsonic speeds



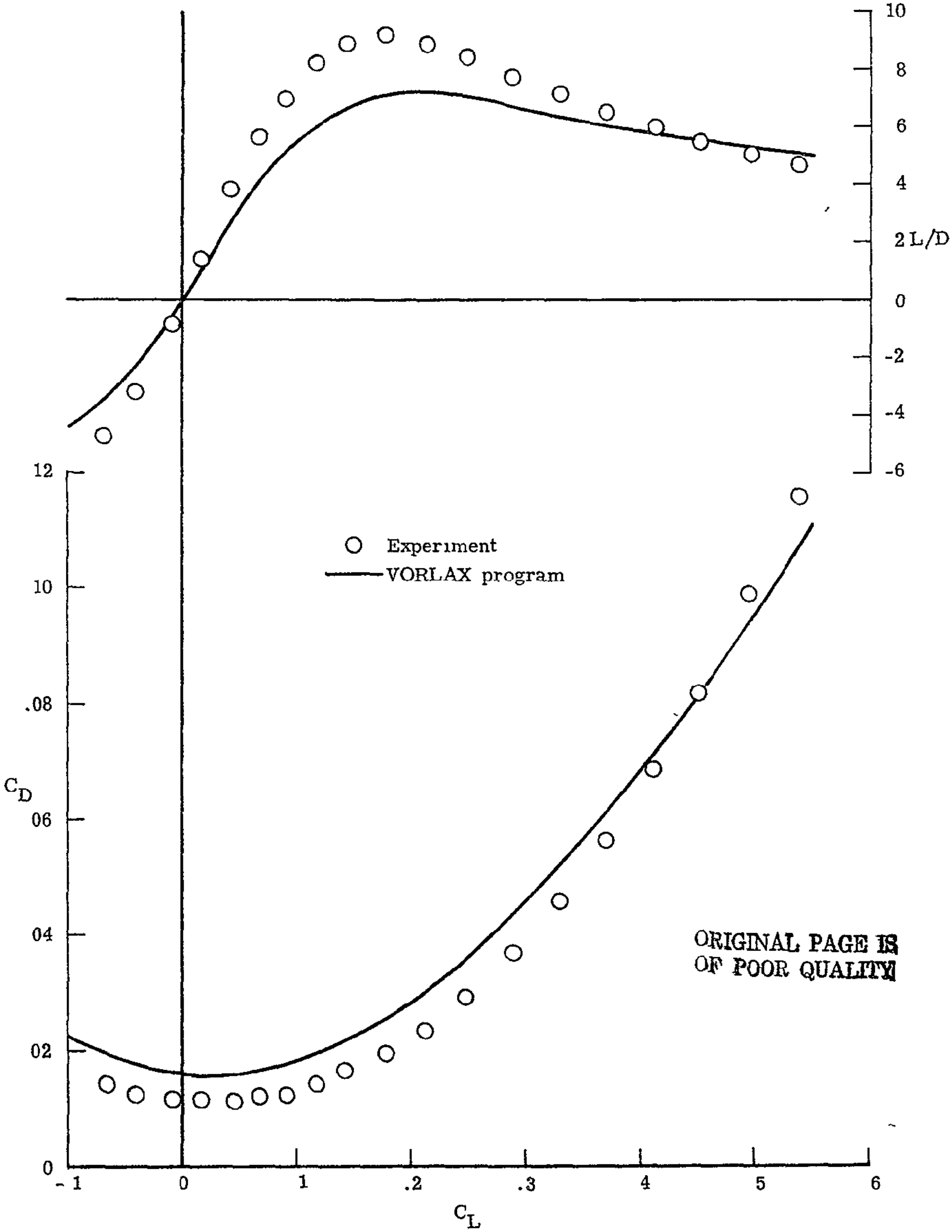
(a) Concluded

Figure 3 - Continued.



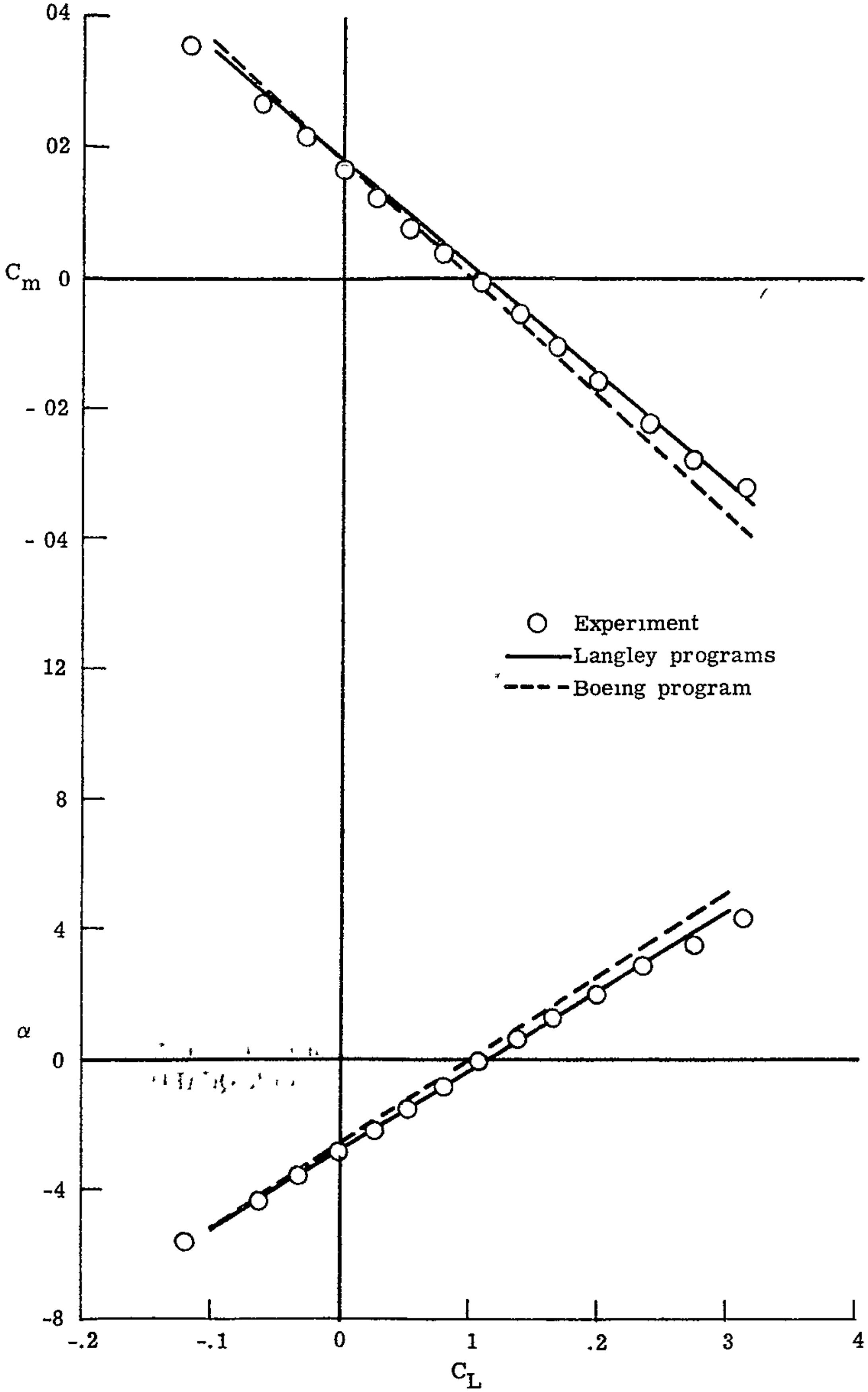
(b) M = 0 90

Figure 3 - Continued.



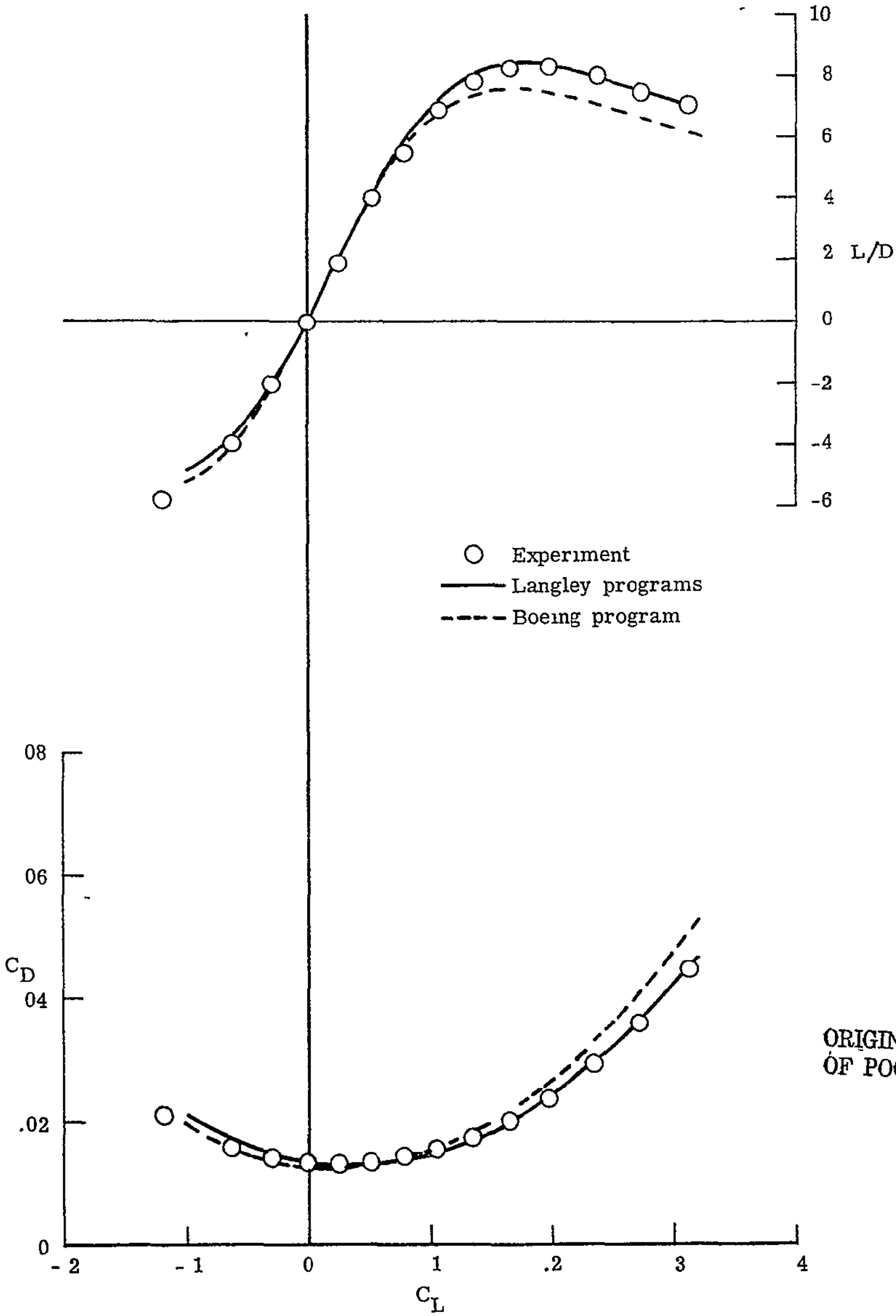
(b) Concluded

Figure 3 - Concluded.



(a) $\delta L_6 = 0^{\circ}$

Figure 4 - Comparison of theoretical and experimental results for complete model at Mach 1 20.



(a) Concluded.

Figure 4 - Continued

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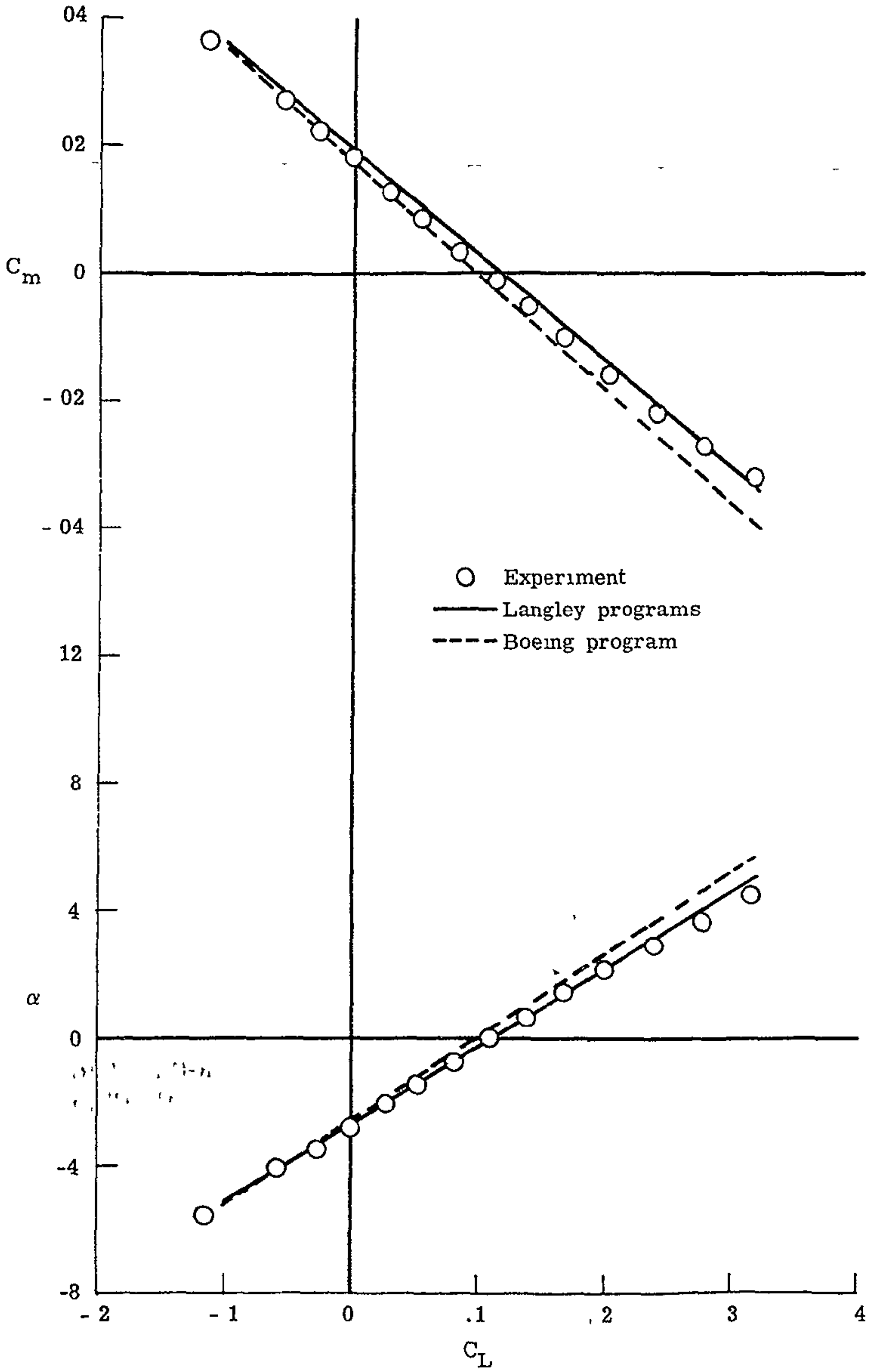
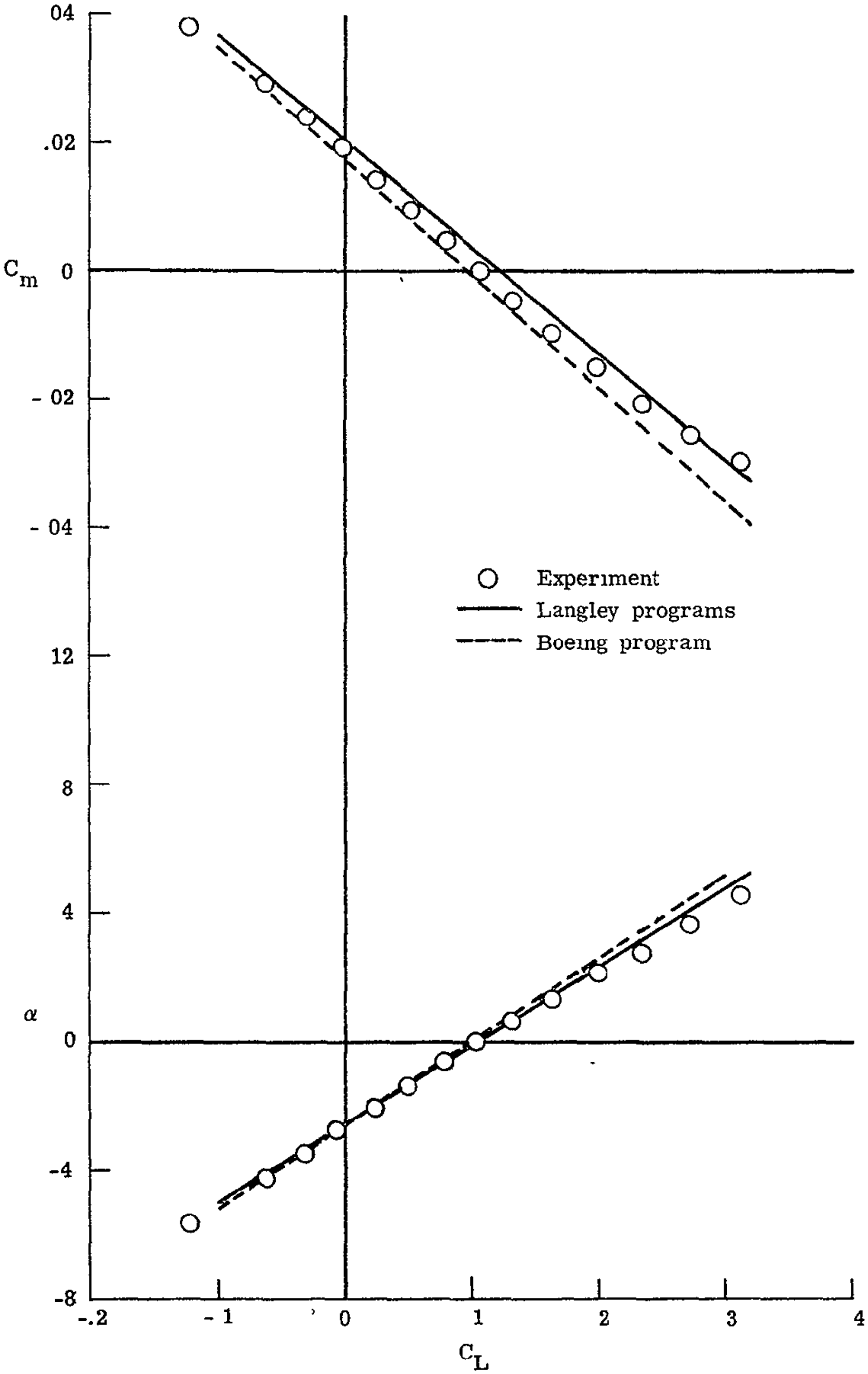


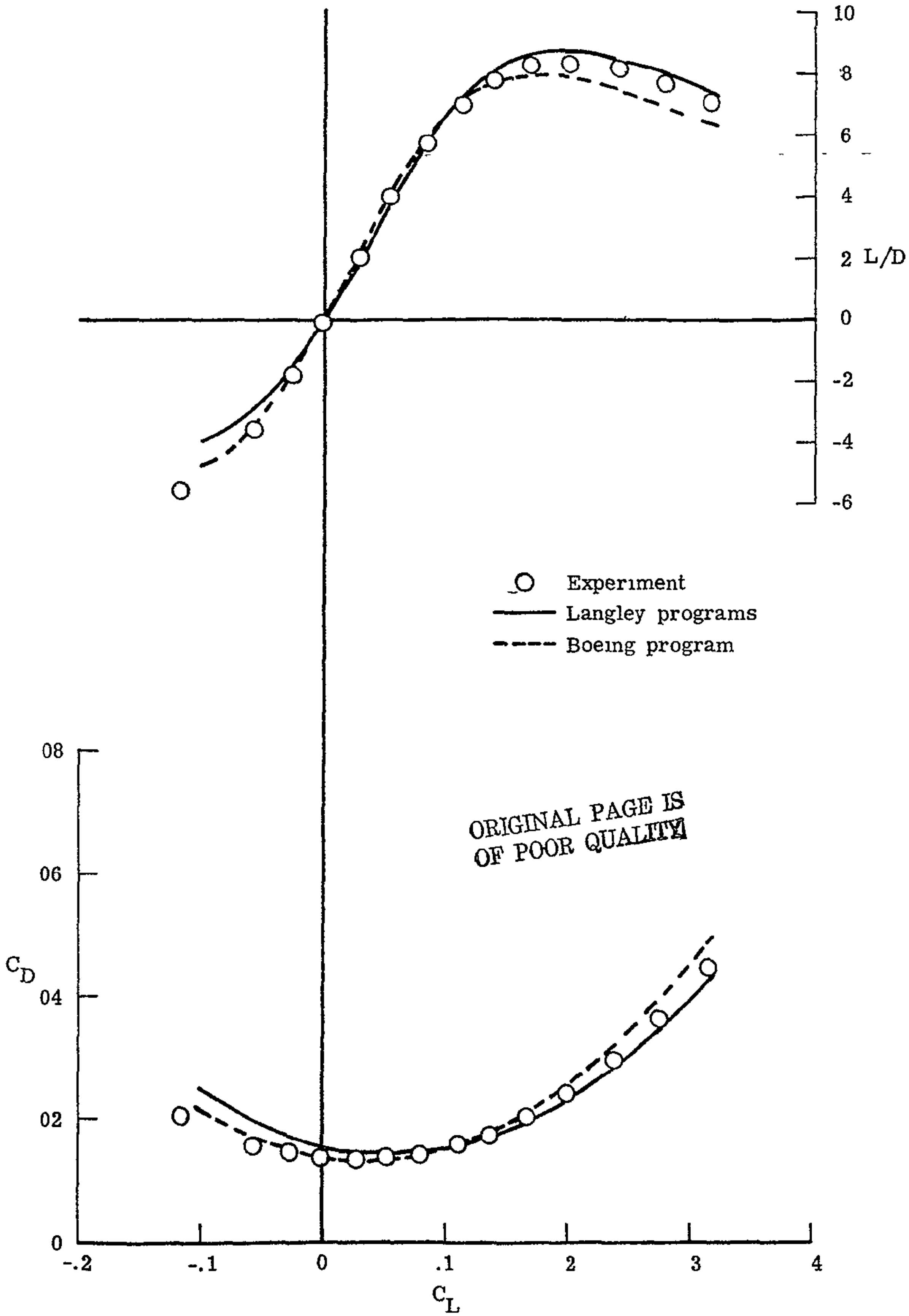
(b) $\delta L_6 = 10^o$

Figure 4.- Continued



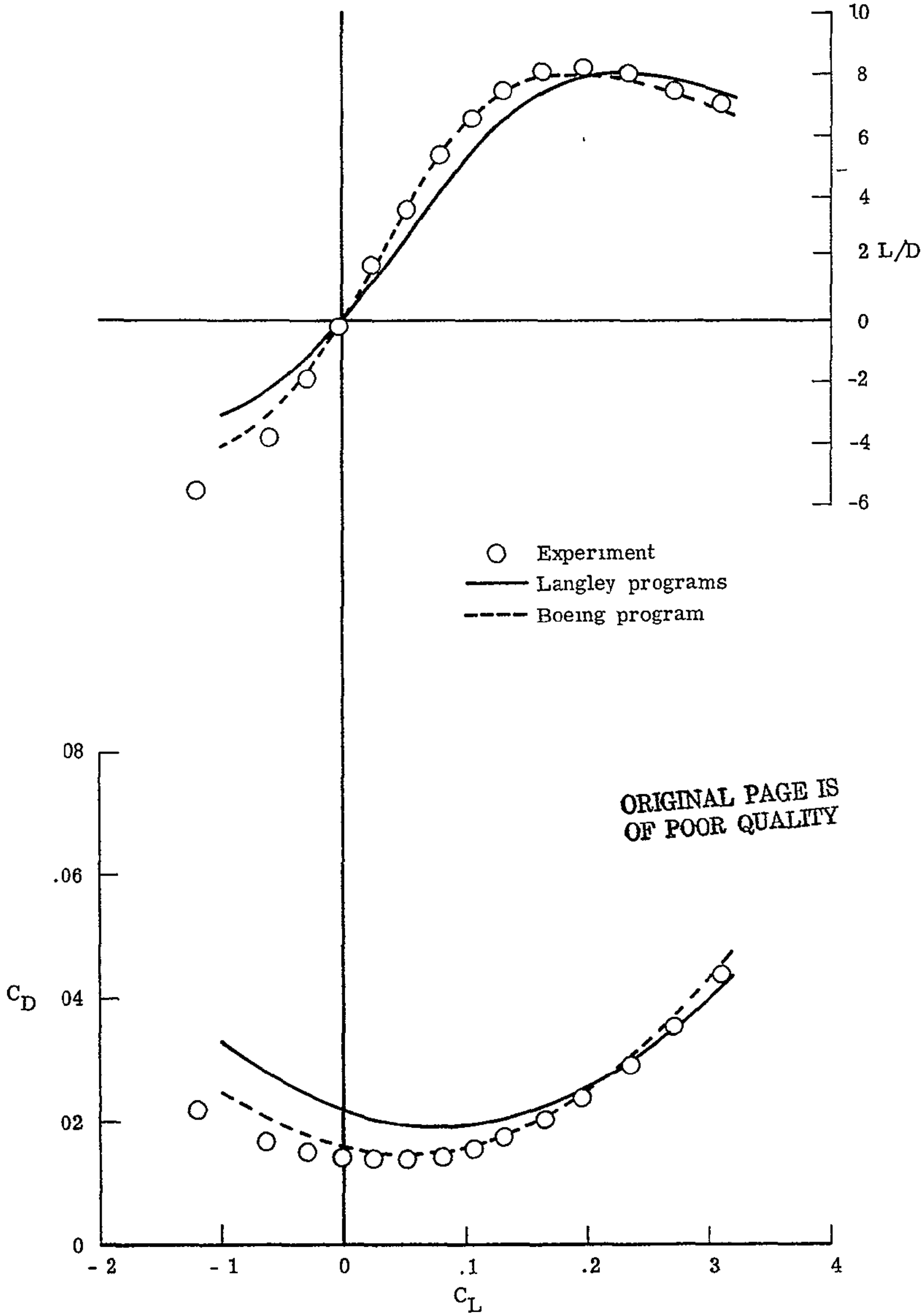
(c) $\delta L_6 = 20^0$

Figure 4 - Continued.



(b) Concluded

Figure 4.- Continued.



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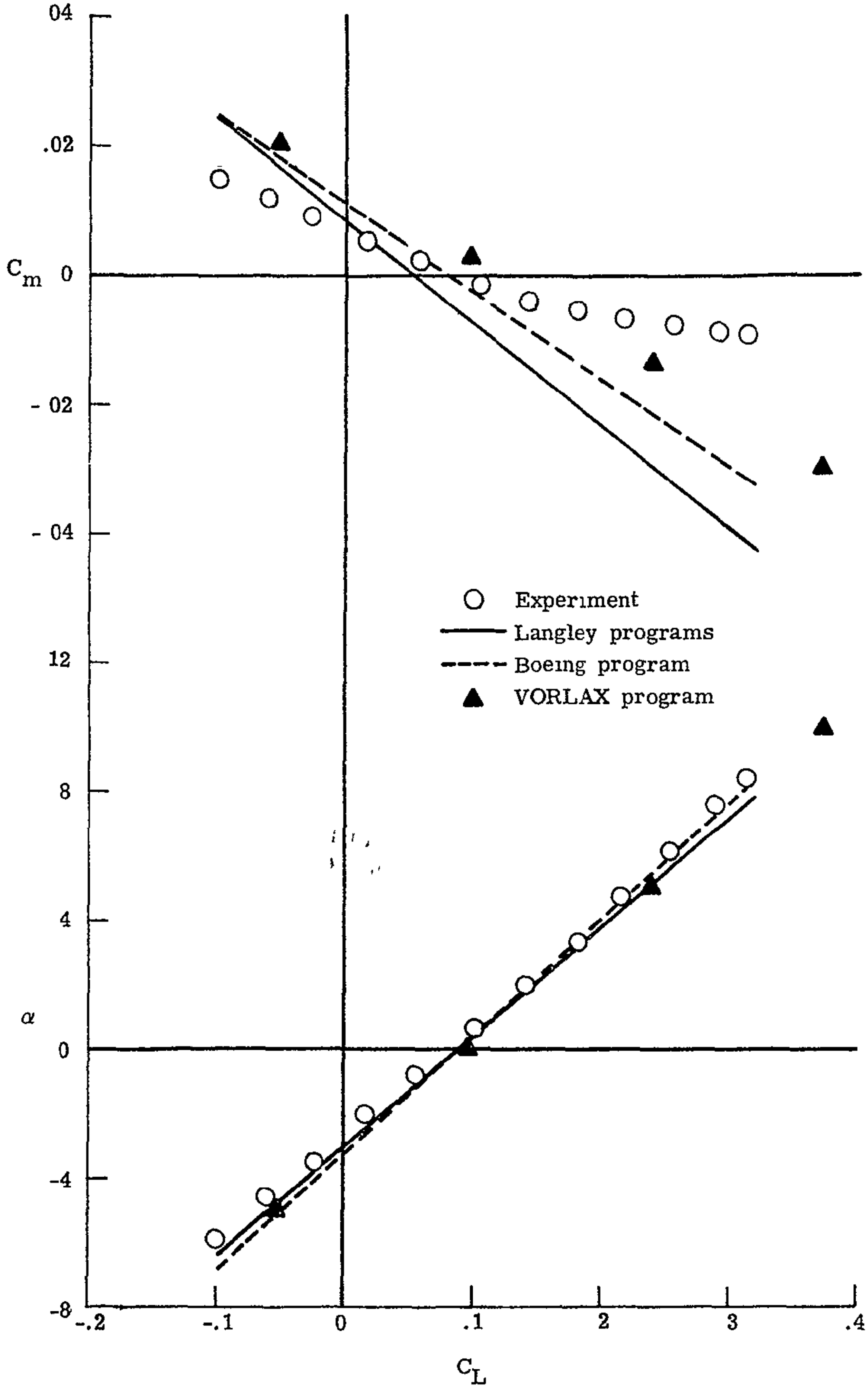
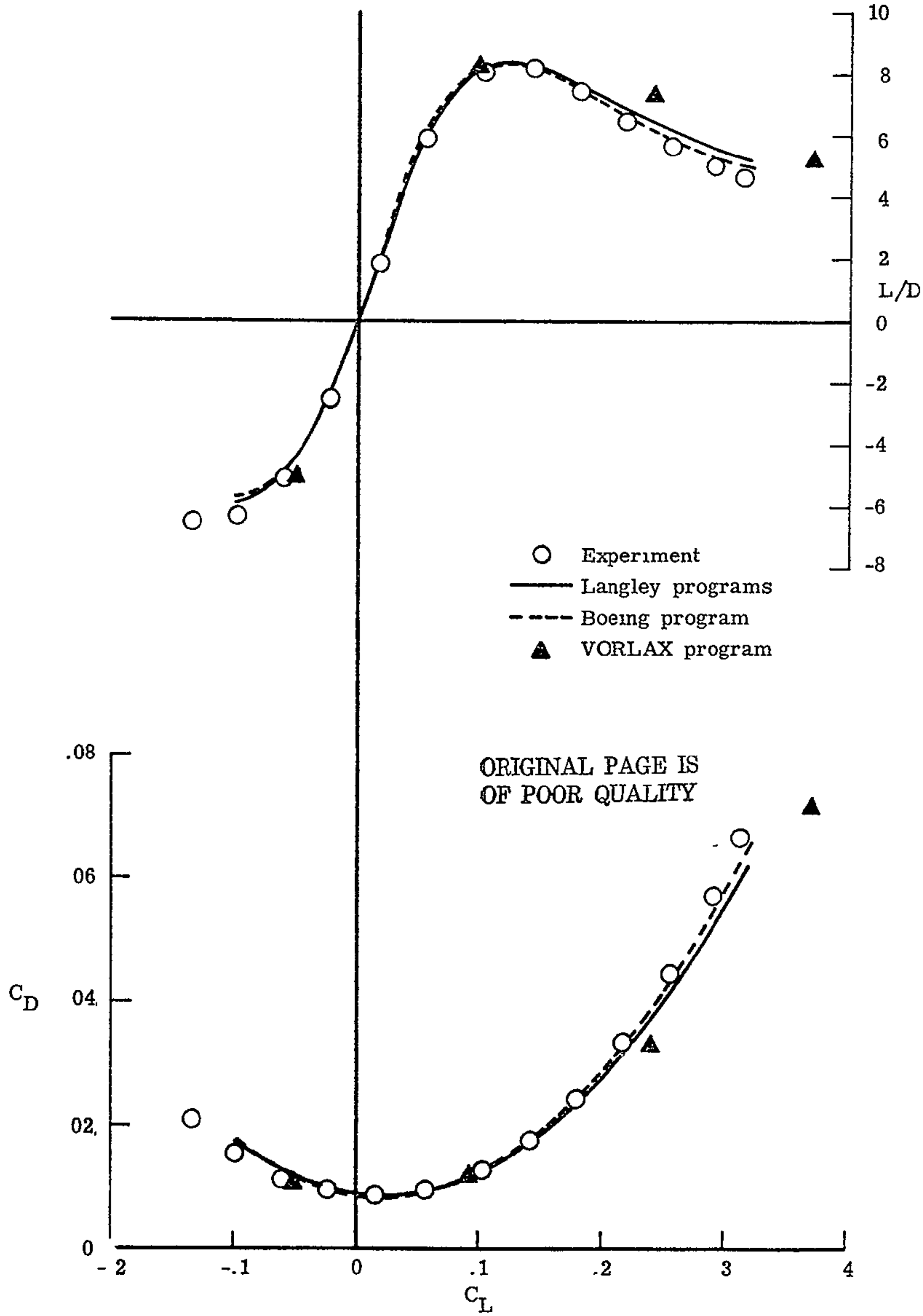
(c) Concluded

Figure 4 - Concluded



(a) M = 2.30

Figure 5 - Comparison of theoretical and experimental results for wing-body at supersonic speeds



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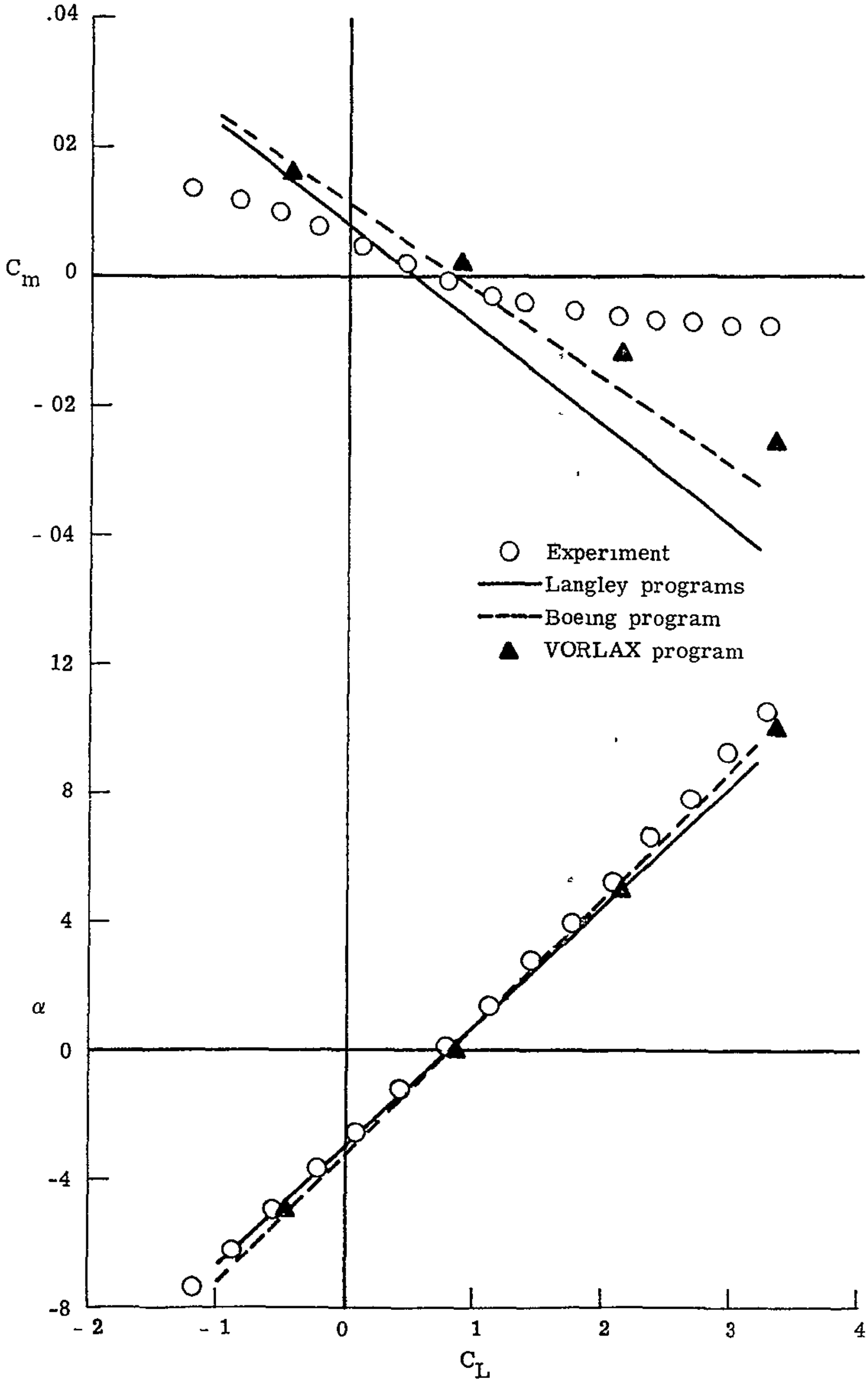
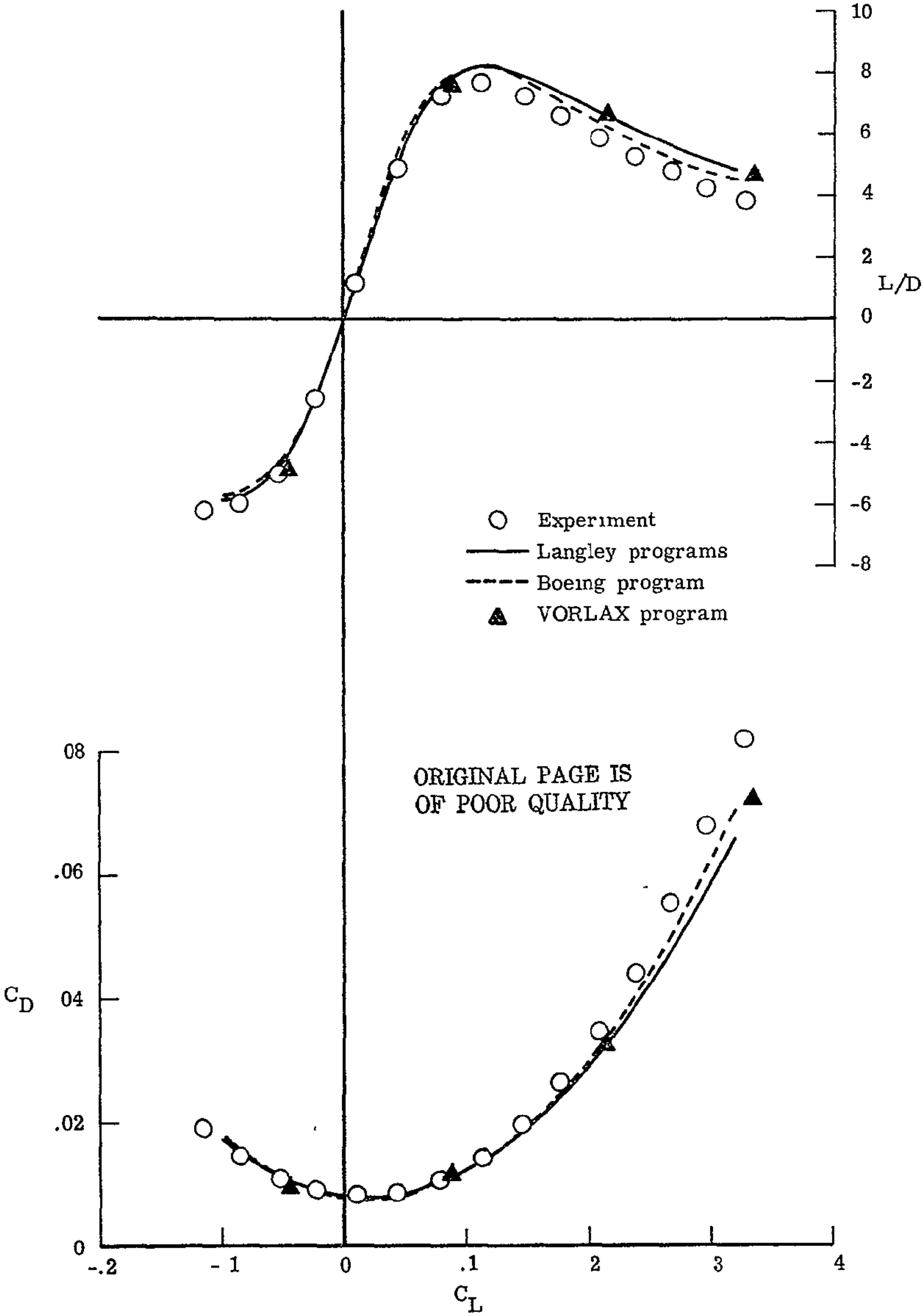
(a) Concluded.

Figure 5.- Continued.



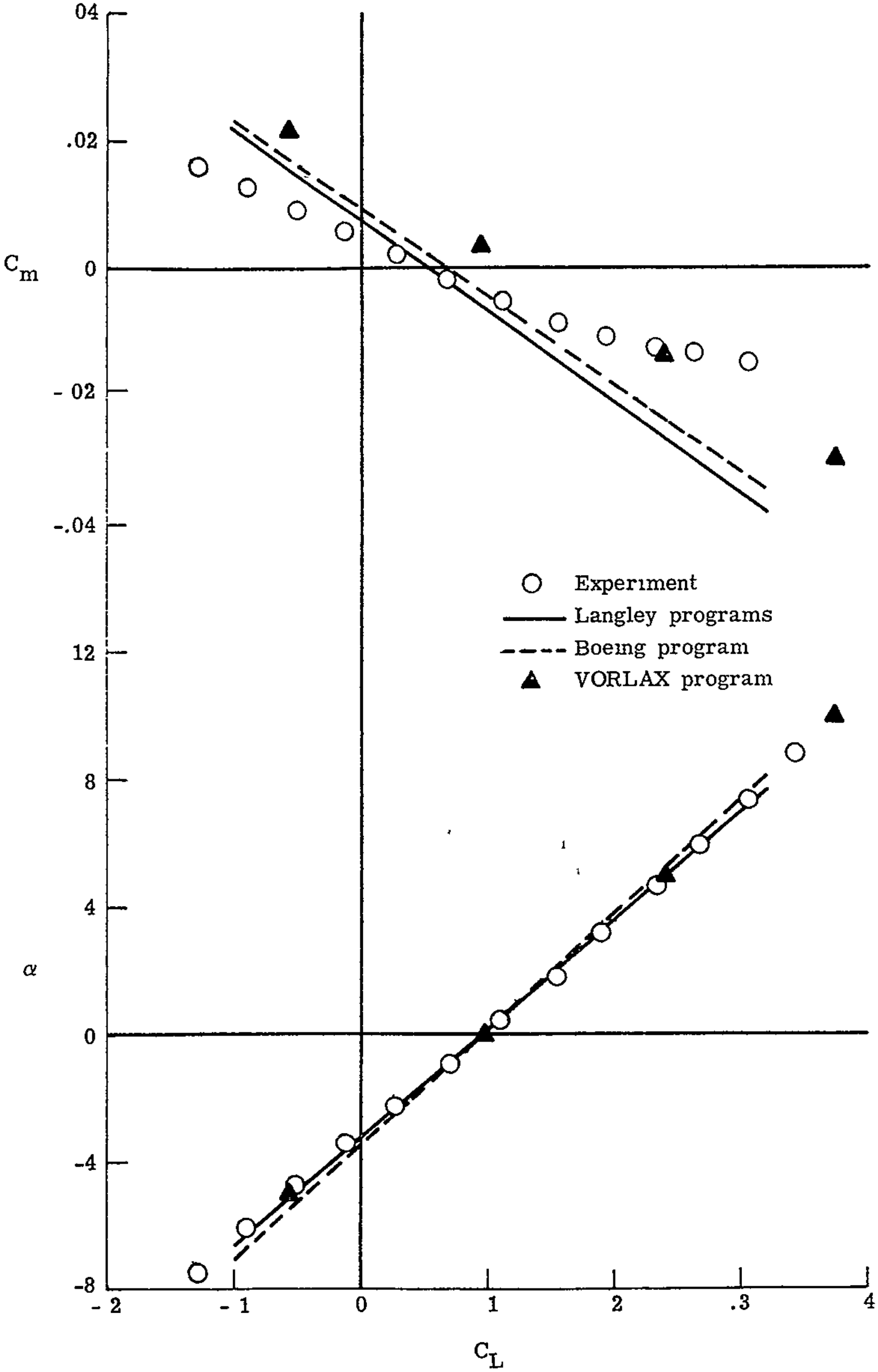
(b) M = 2 70.

Figure 5 - Continued.



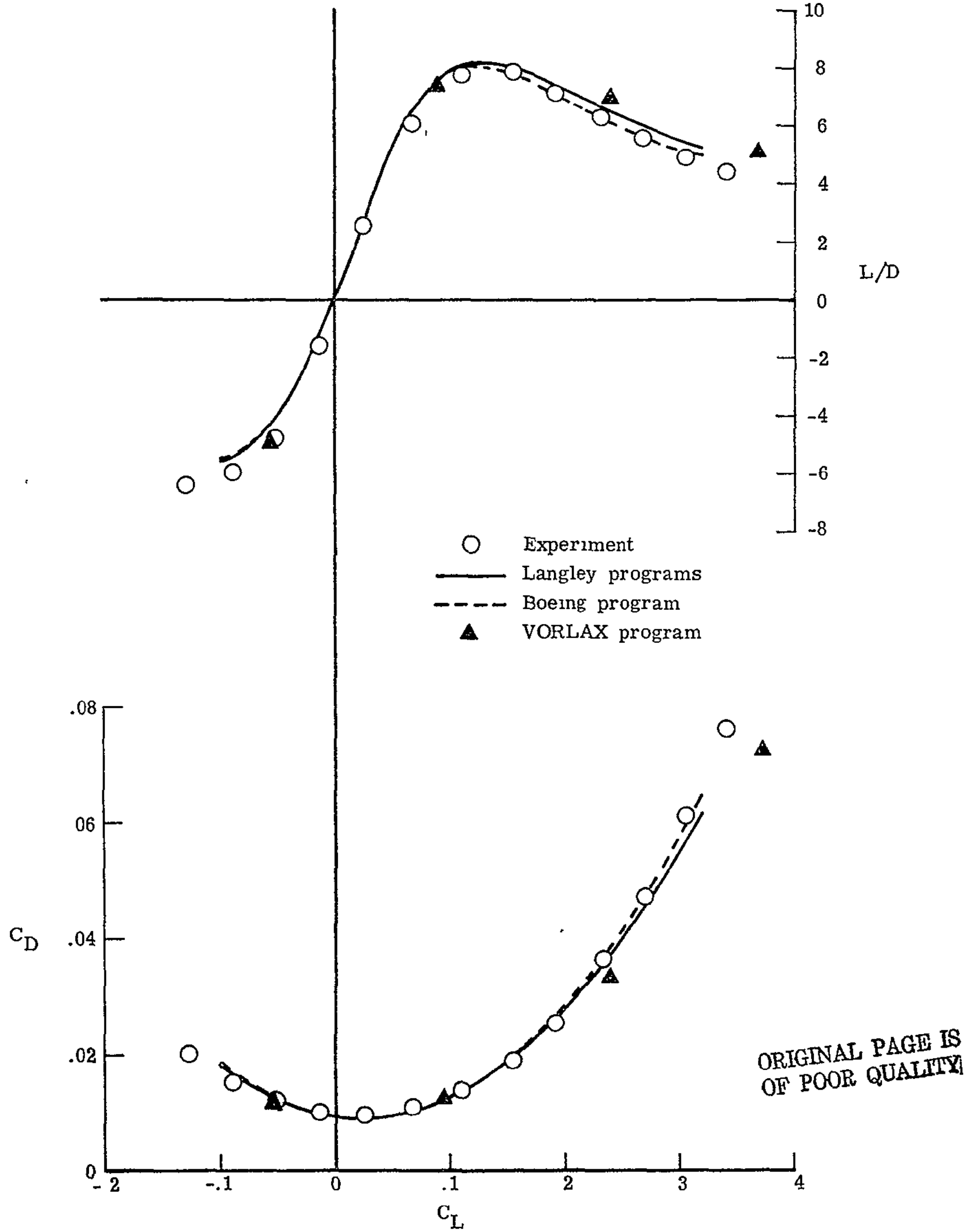
(b) Concluded

Figure 5.- Concluded.



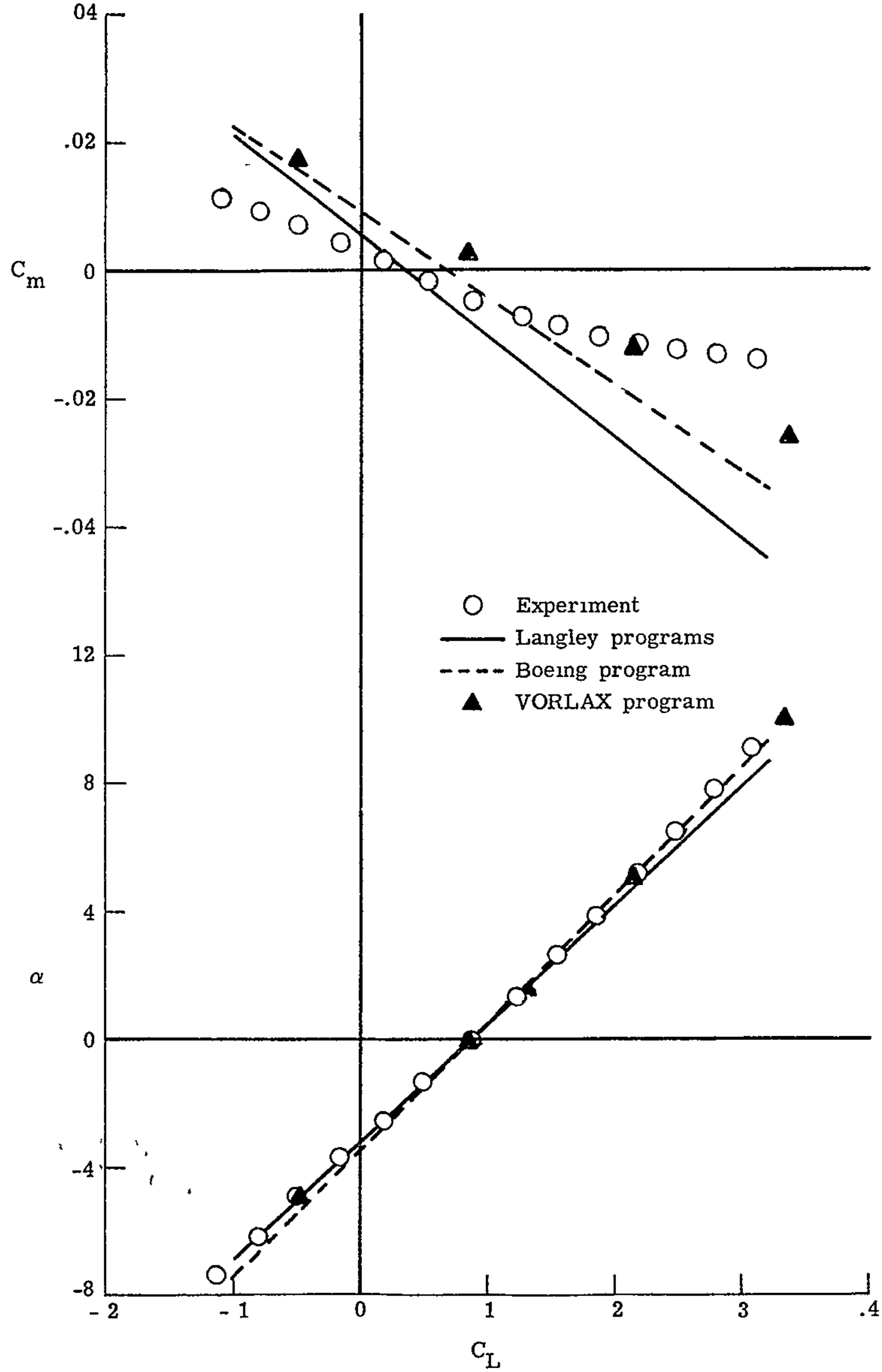
(a) M = 2.30

Figure 6 - Comparison of theortical and experimental resuts for wing-body-nacelles at supersonic speeds.



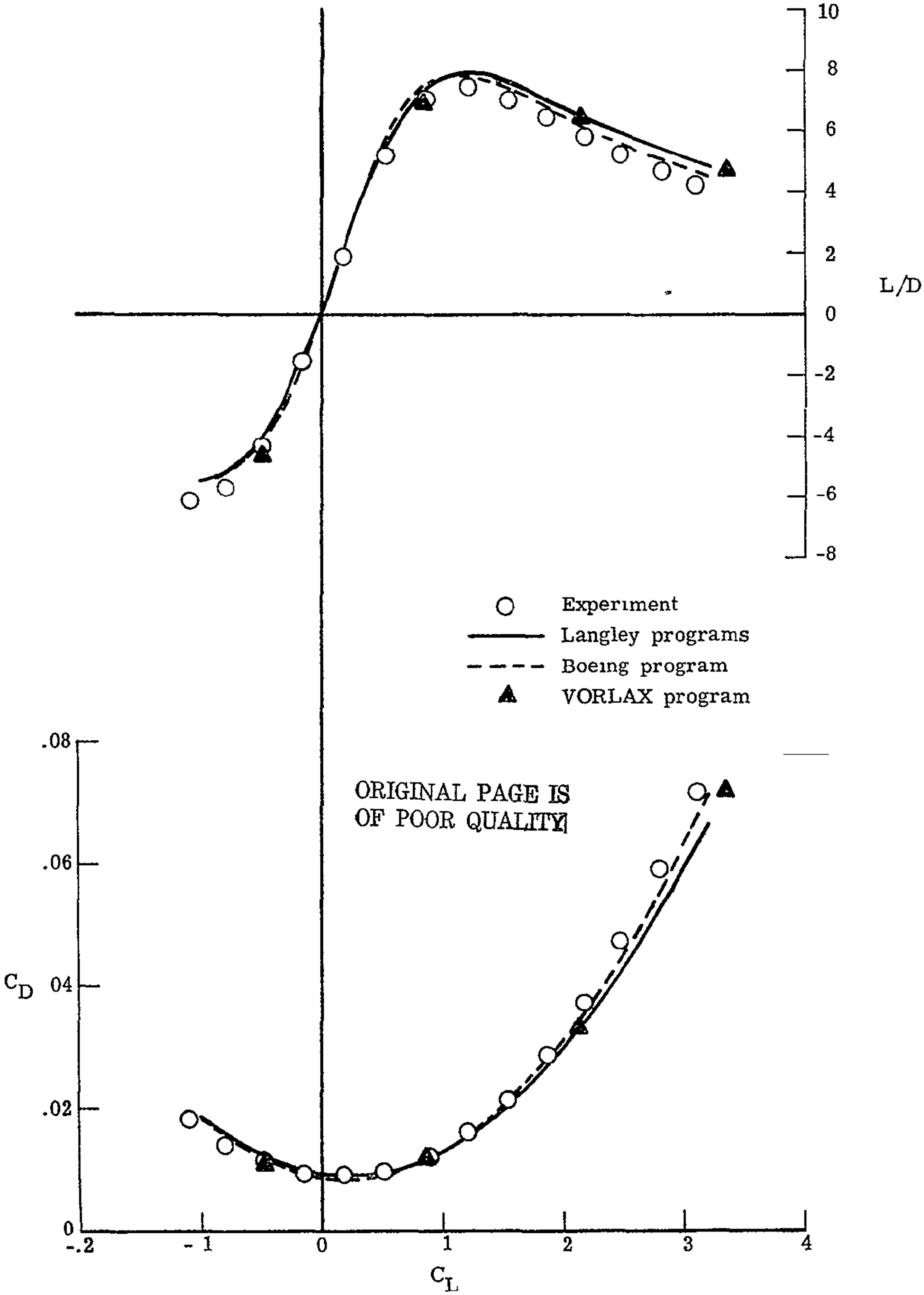
(a) Concluded

Figure 6 - Continued



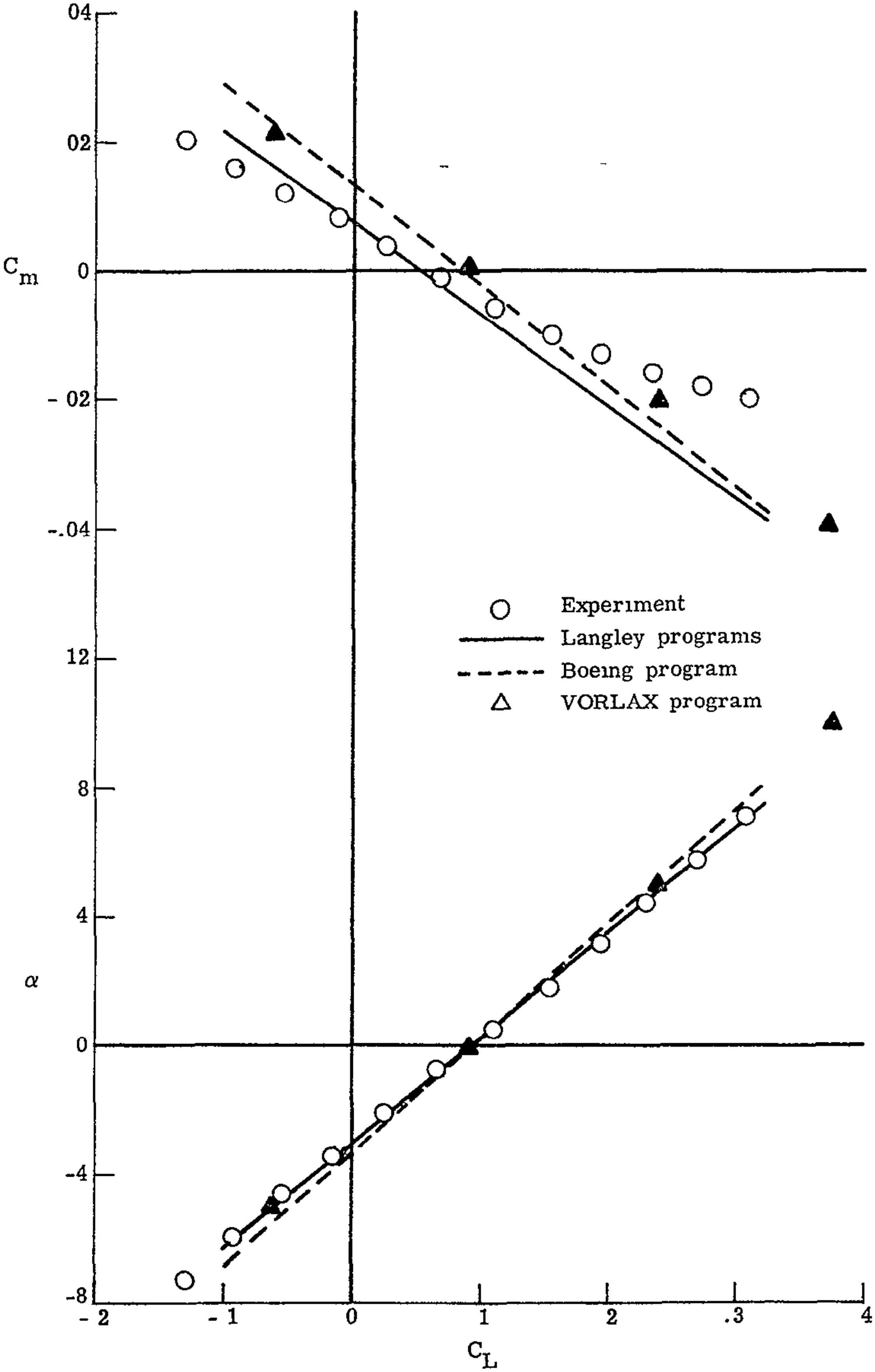
(b) M = 2 70

Figure 6 - Continued



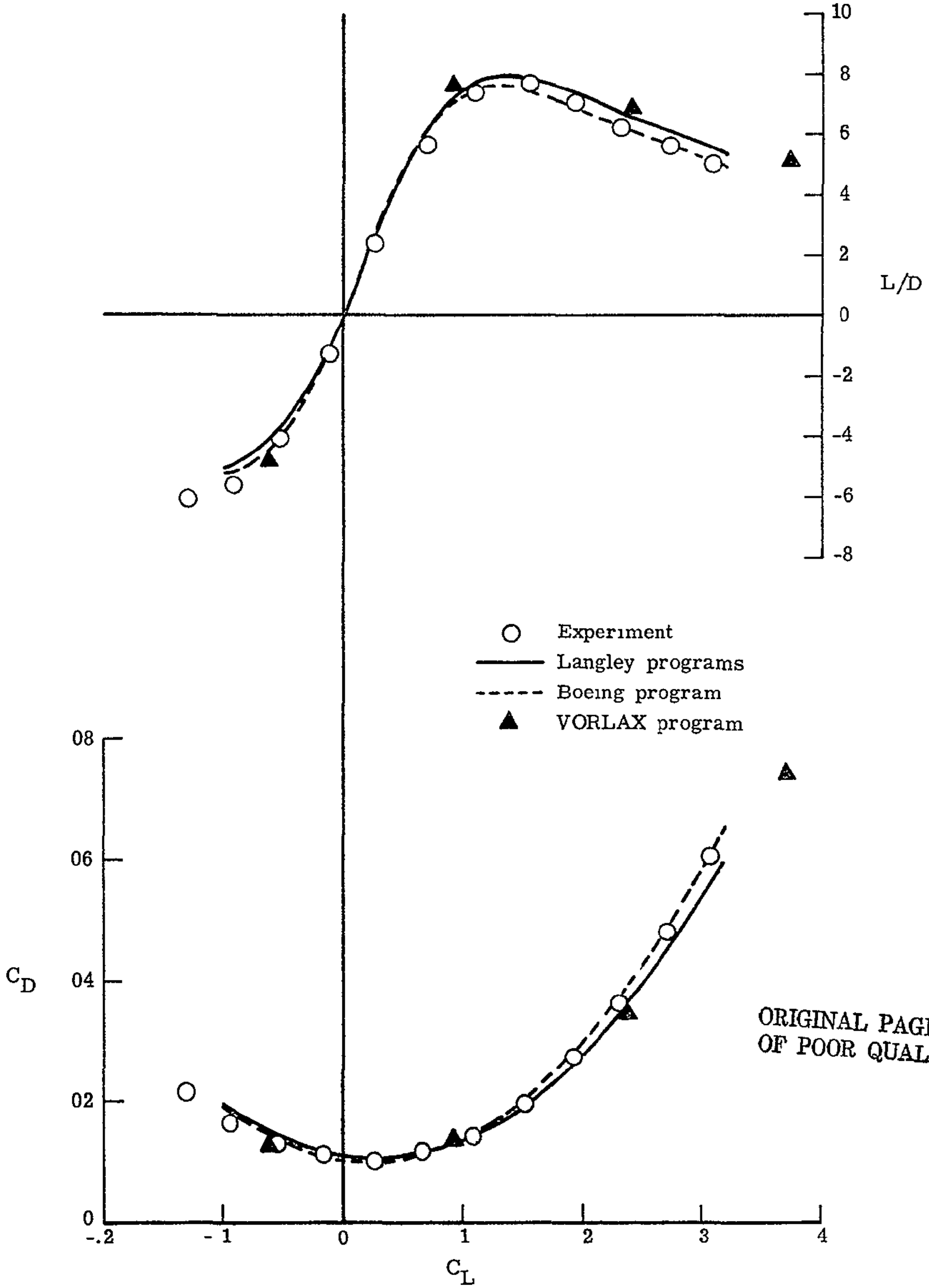
(b) Concluded

Figure 6 - Concluded



(a) M = 2.30

Figure 7.- Comparison of theortical and experimental results for complete model at supersonic speeds



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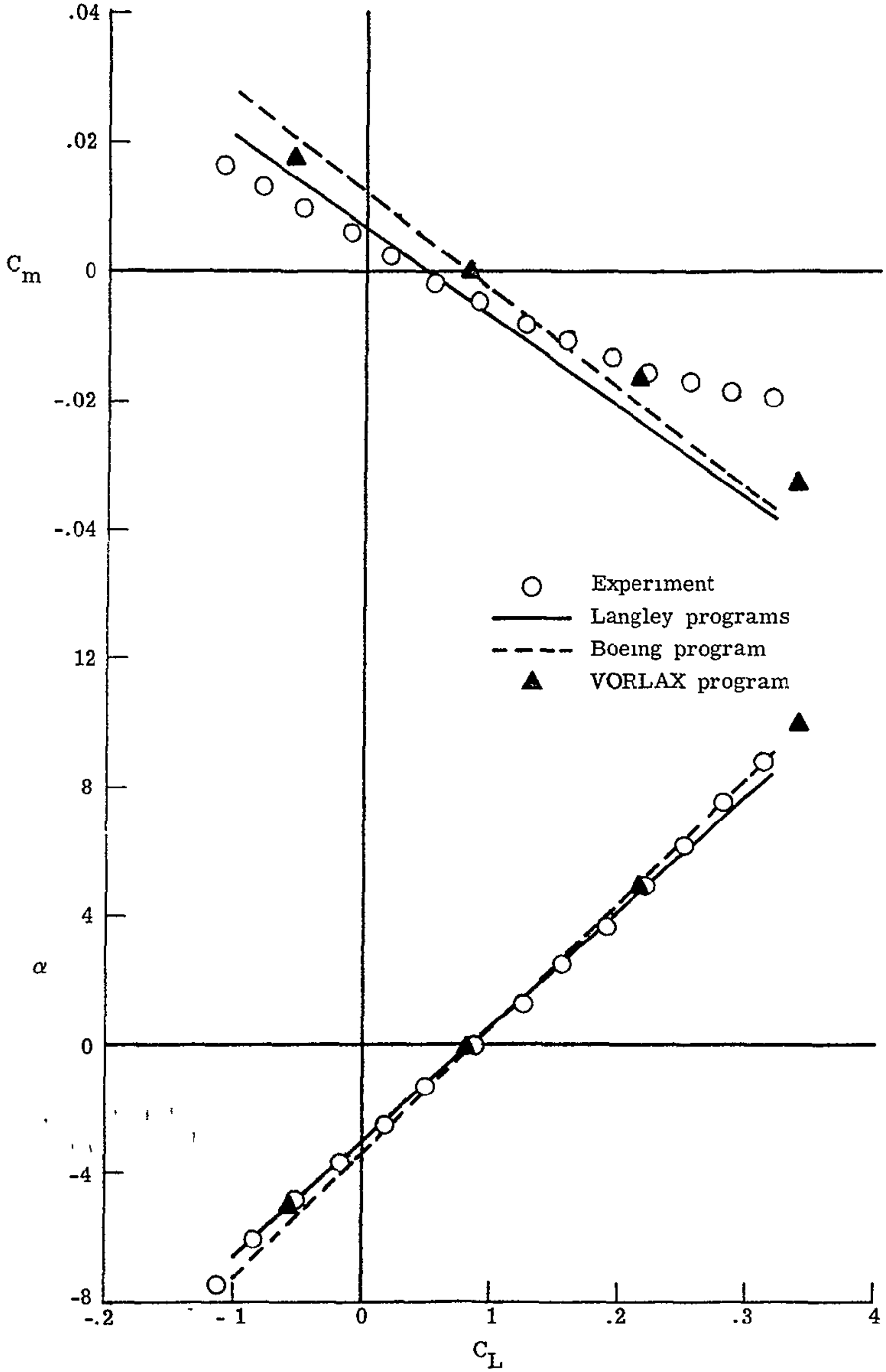
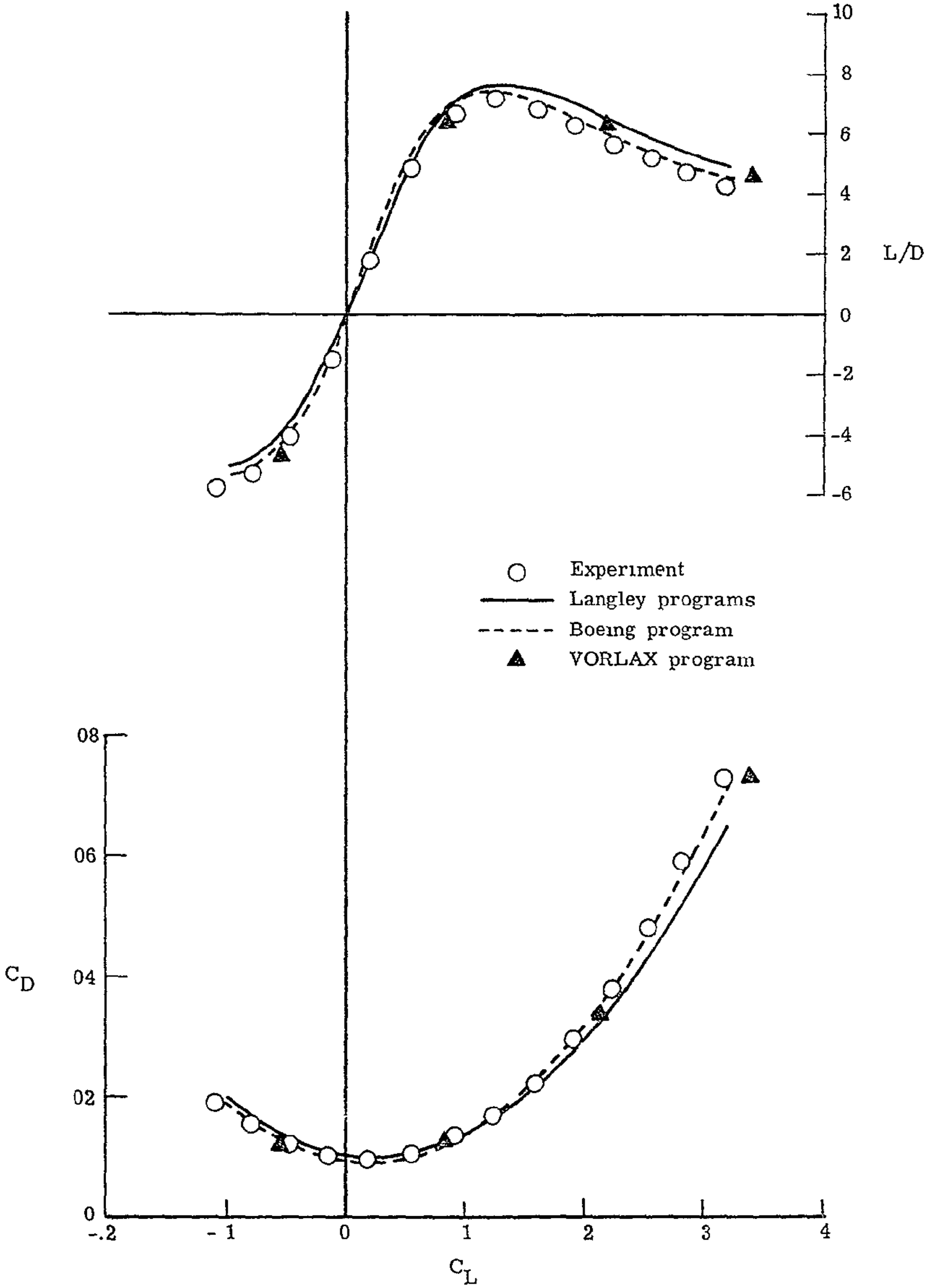
(a) Concluded

Figure 7 - Continued



(b) M = 2.70

Figure 7 - Continued



(b) Concluded

Figure 7 - Concluded

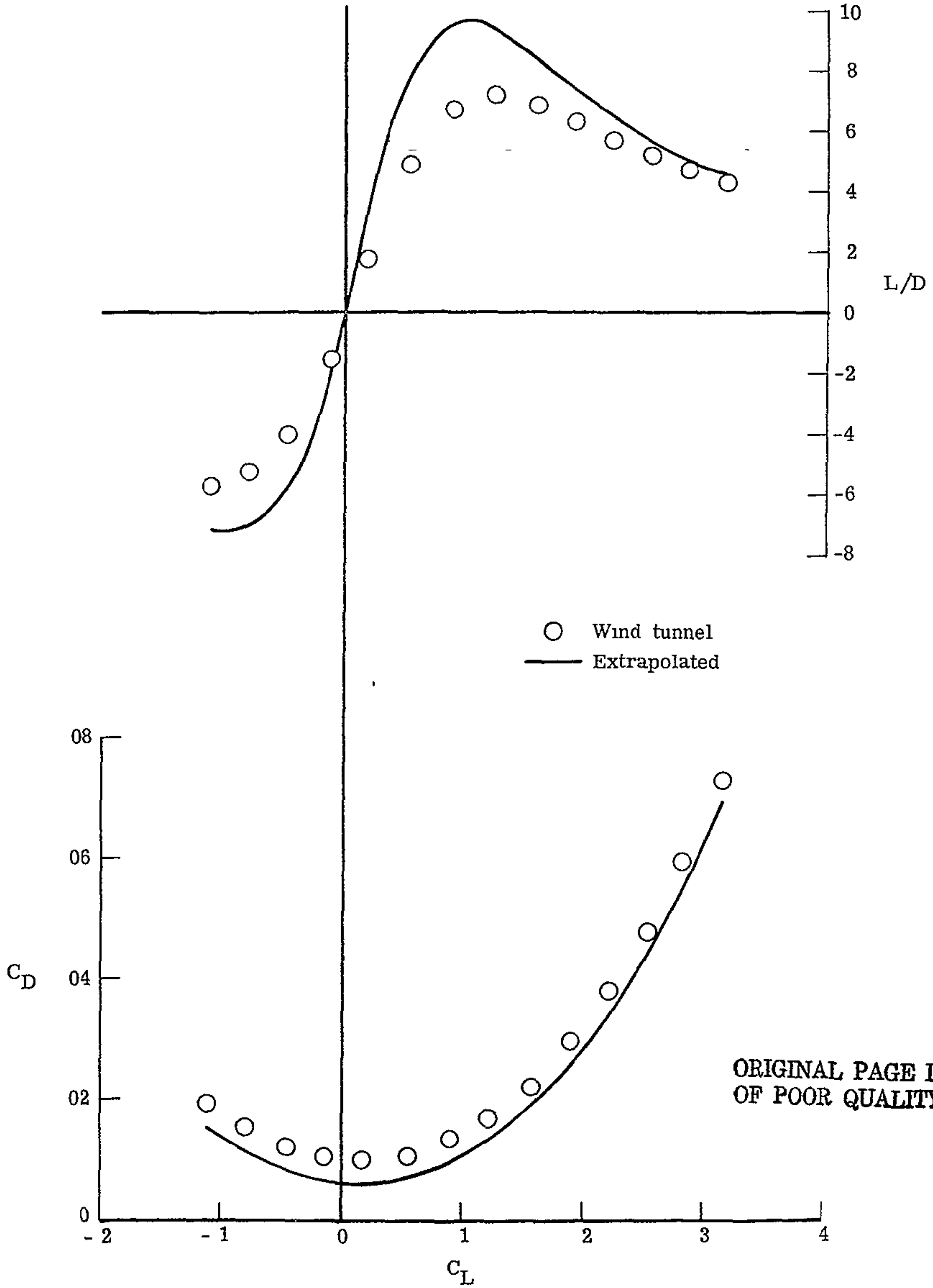


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Figure 8.- Extrapolation of wing-tunnel data to full scale flight conditions at Mach 2 70 and an altitude of 18288 m (60000 ft.).

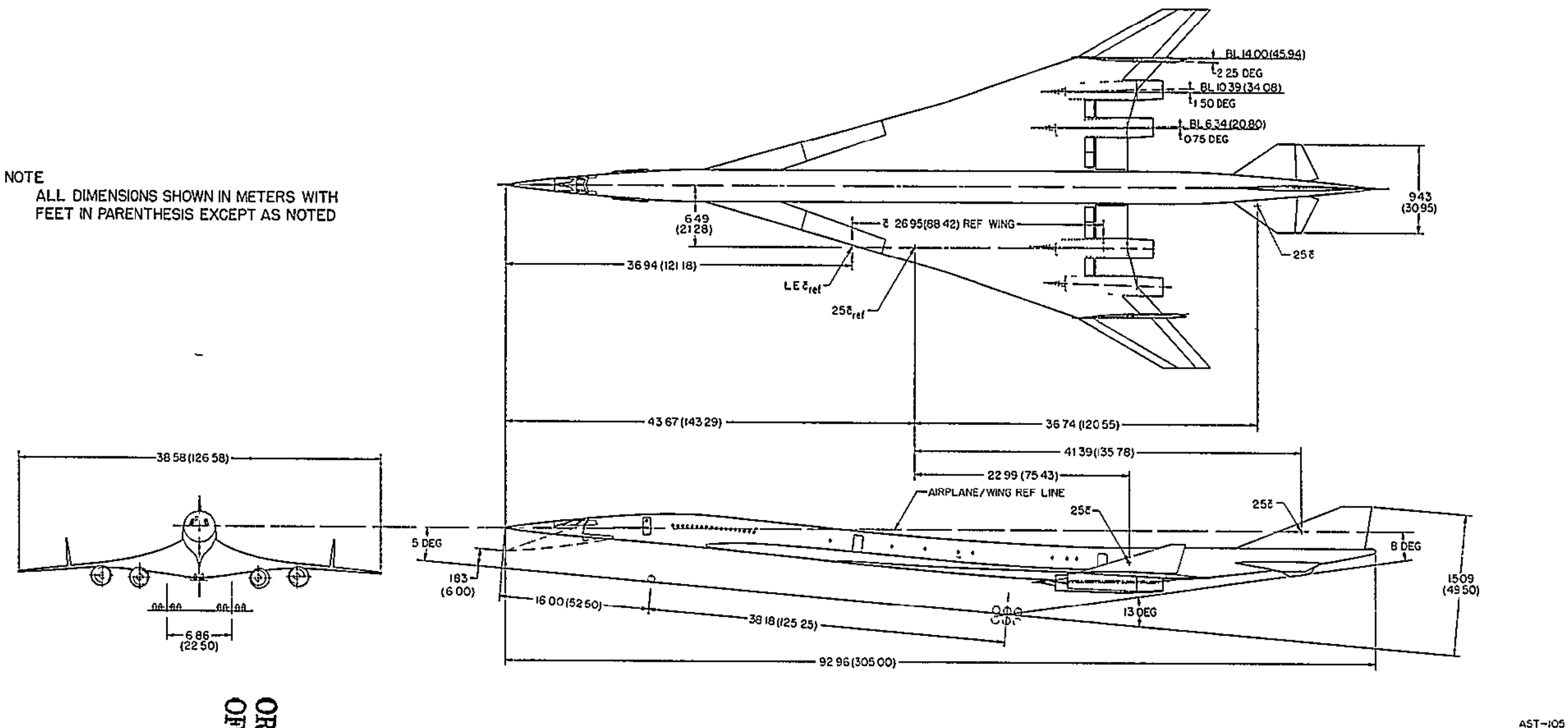

Figure 9. AST-105 Configuration

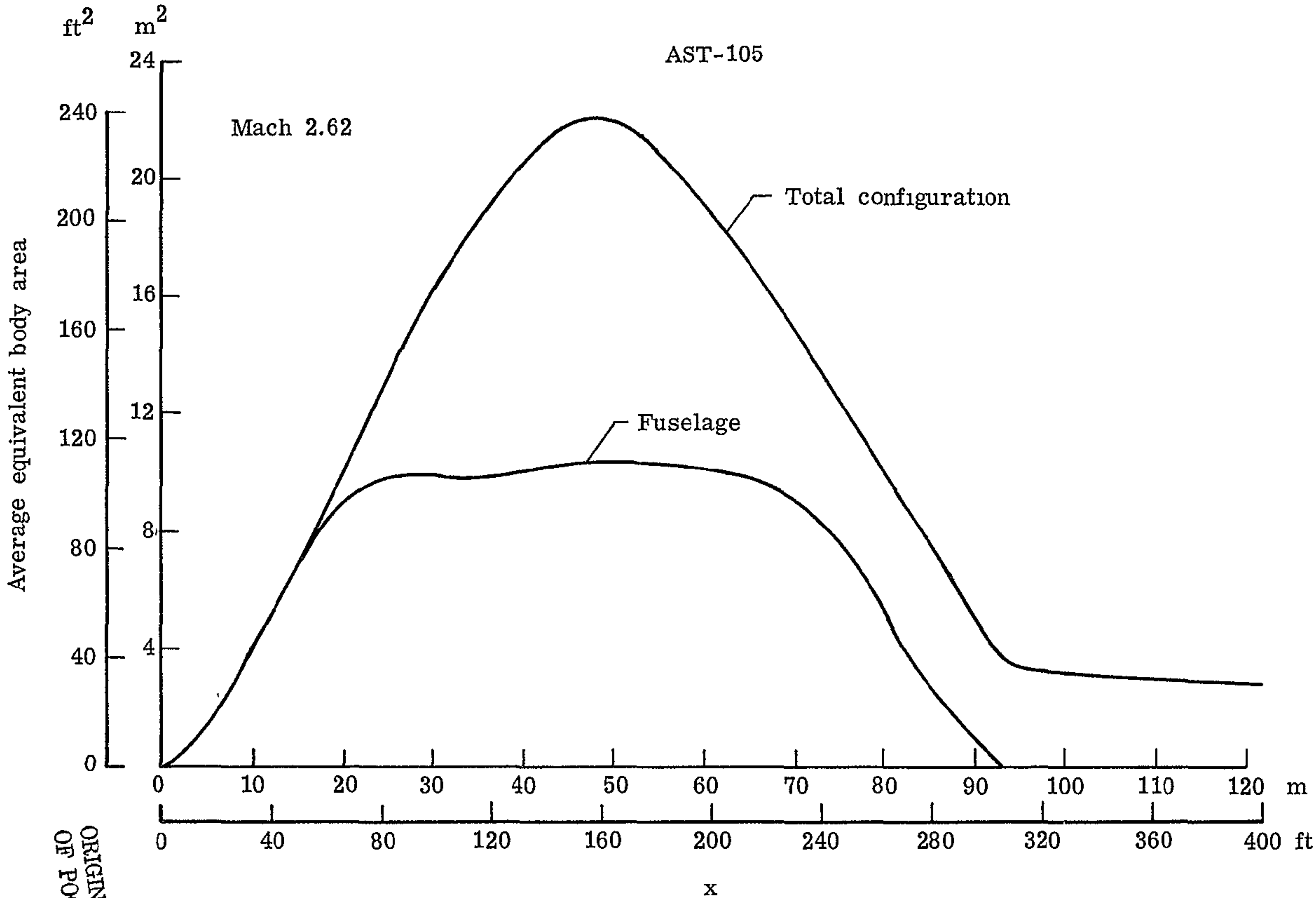

Figure 10.- Equivalent area distributions.

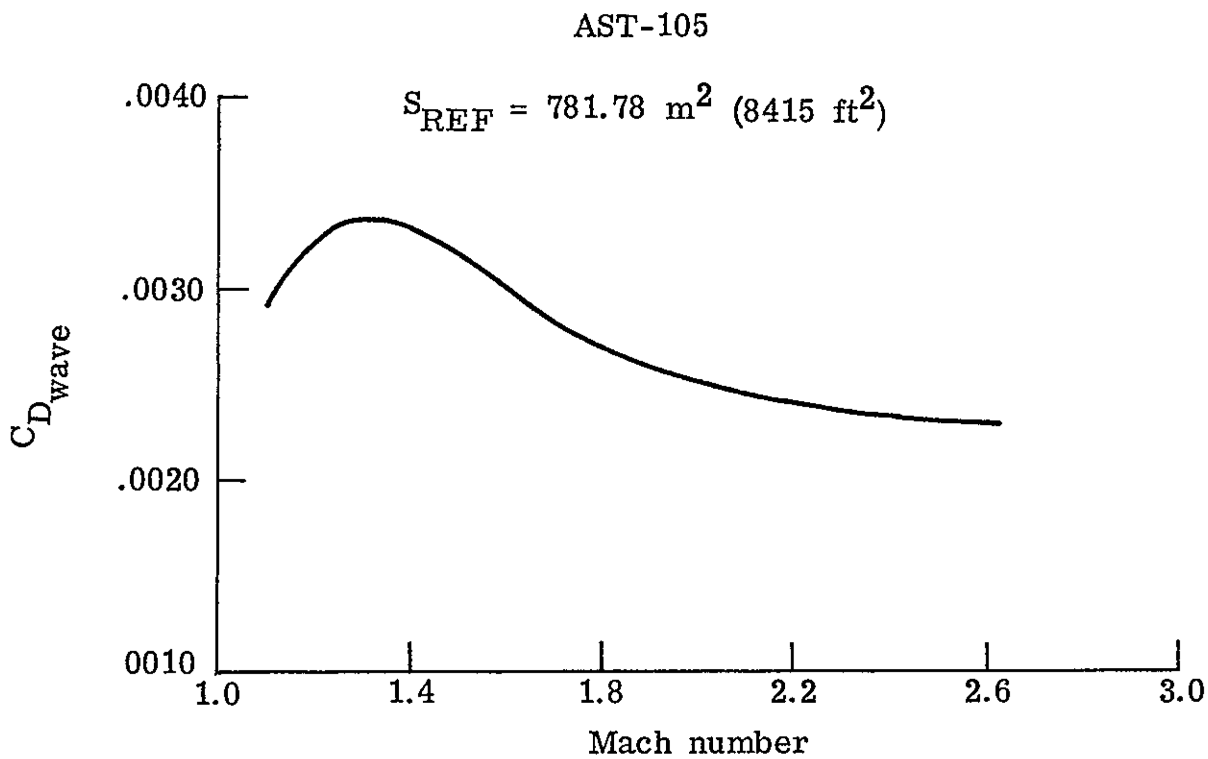

Figure 11.- Wave drag characteristics.

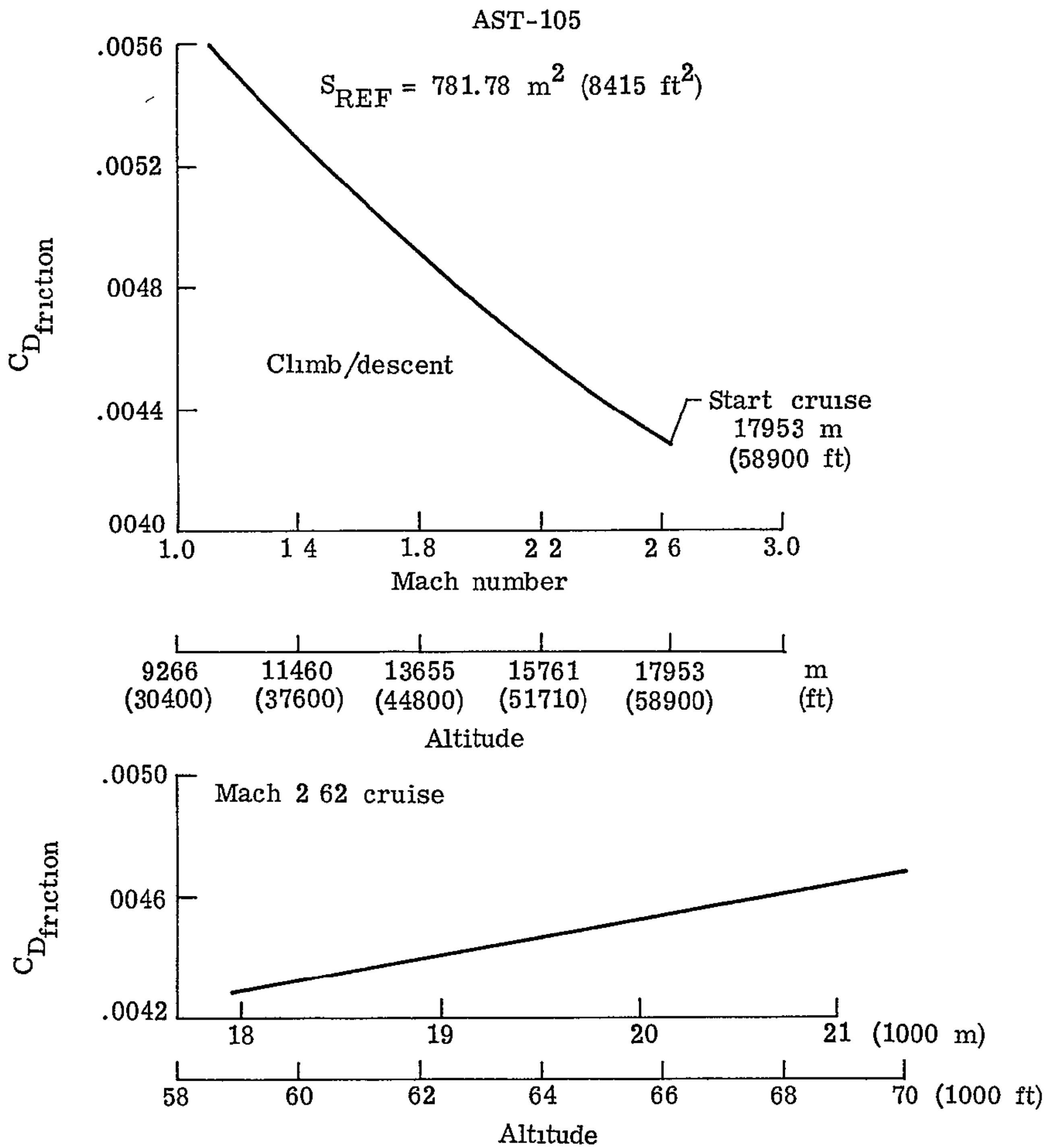

Figure 12.- Skin friction.

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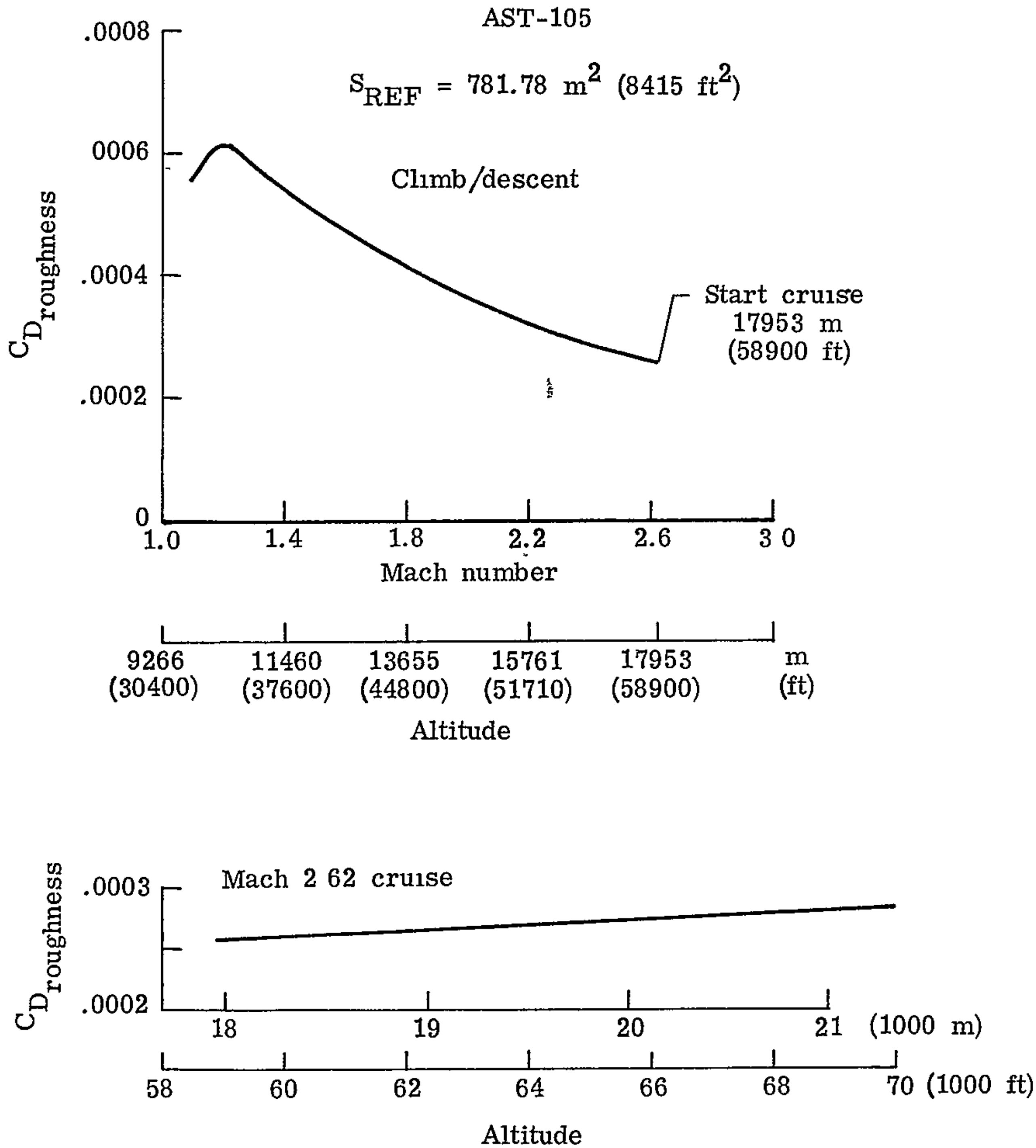

Figure 13.- Roughness drag.

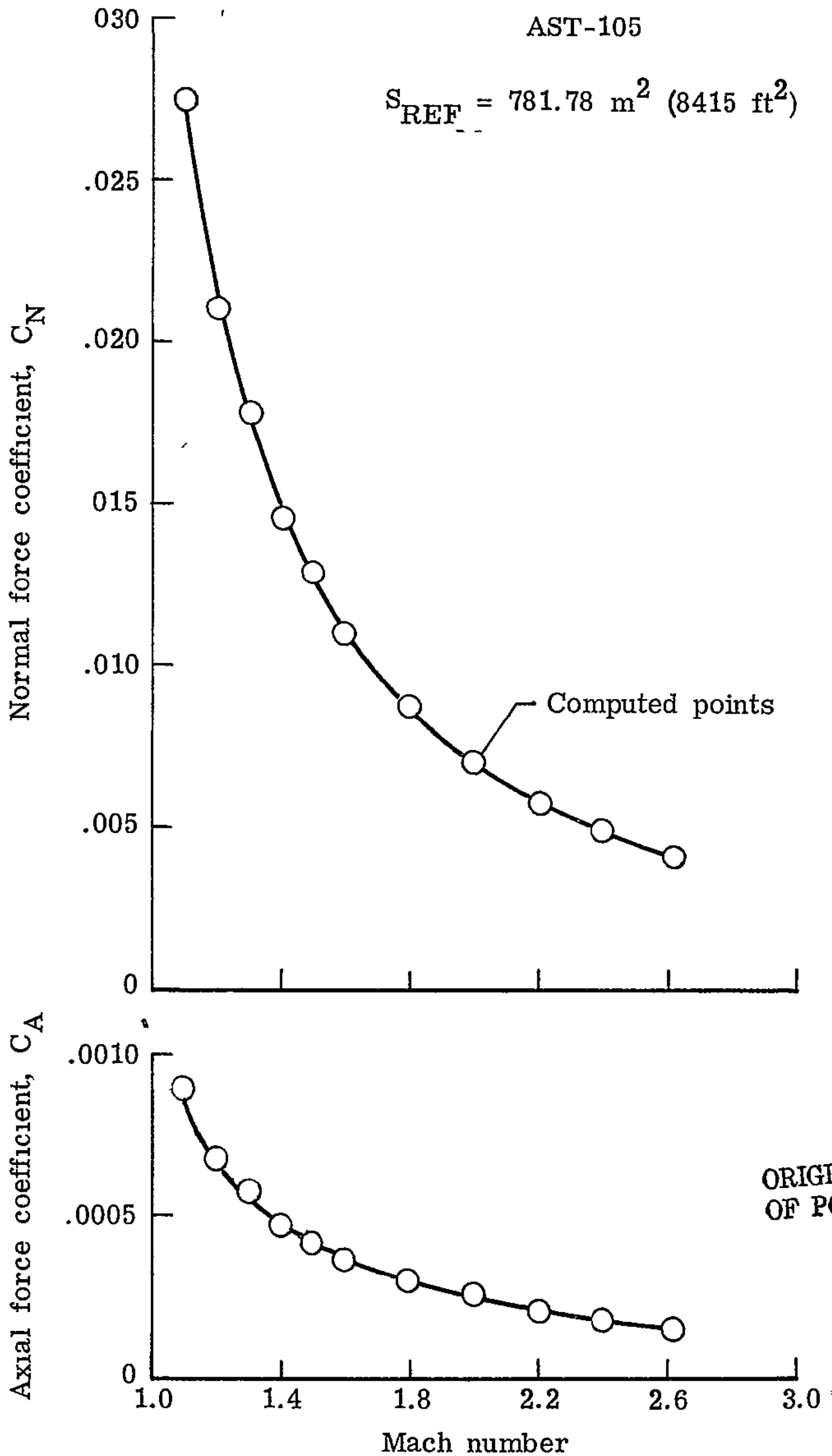

Figure 14.- Nacelle-on-wing interference

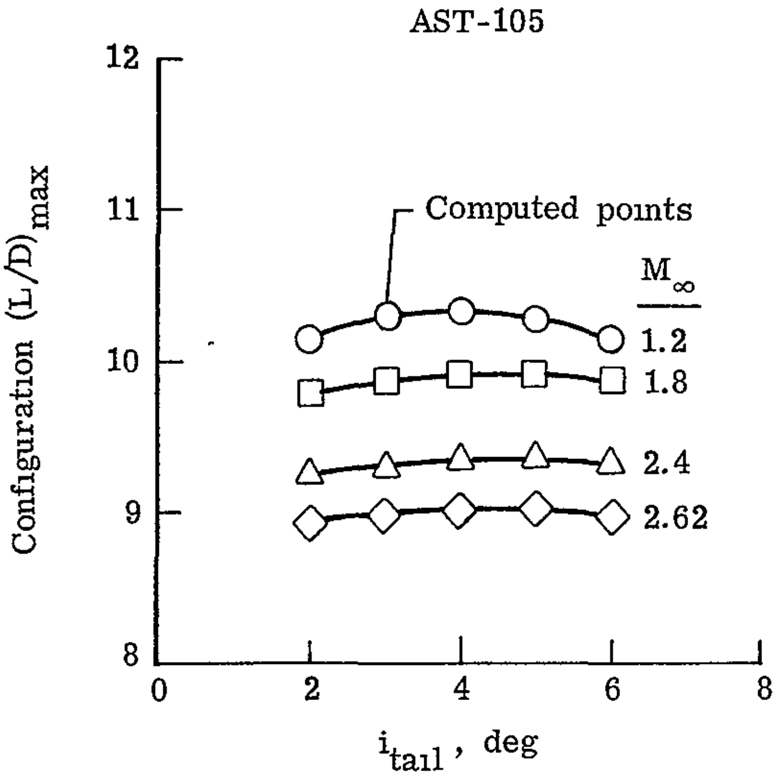

Figure 15.- Typical examples of configuration performance dependence on horizontal tail incidence.

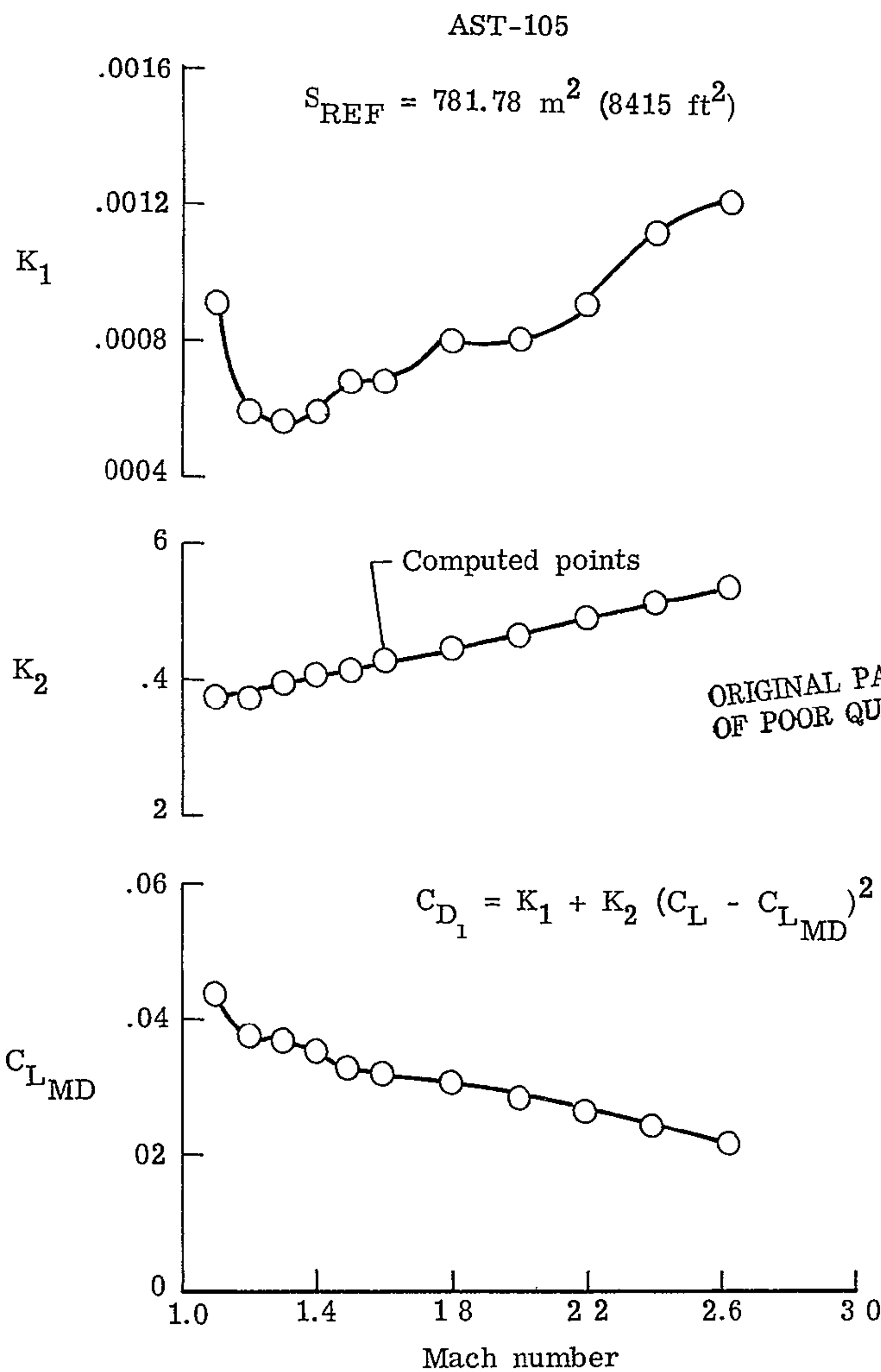

Figure 16.- Drag-due-to-lift parameters

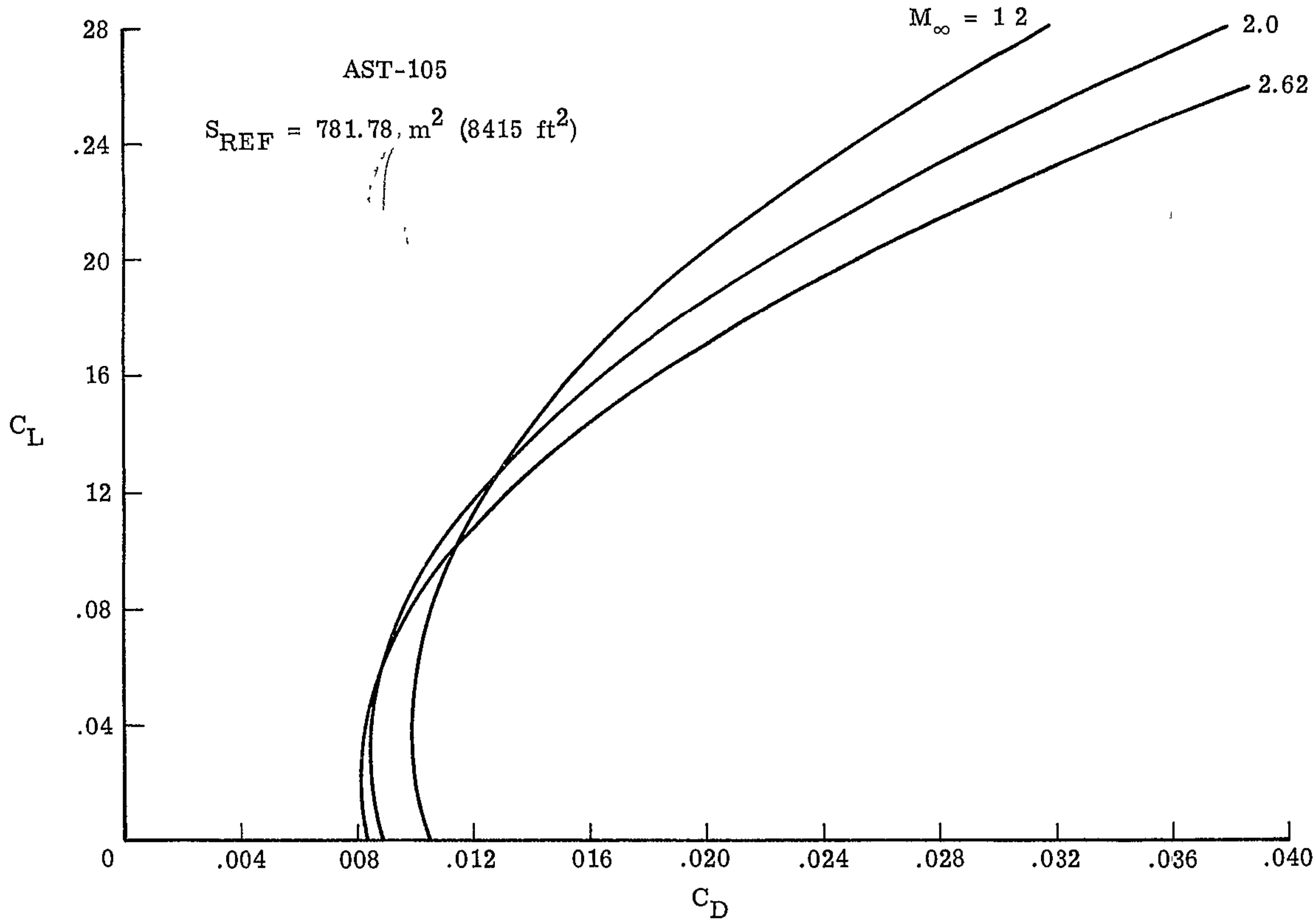

Figure 17.- Typical drag polars.

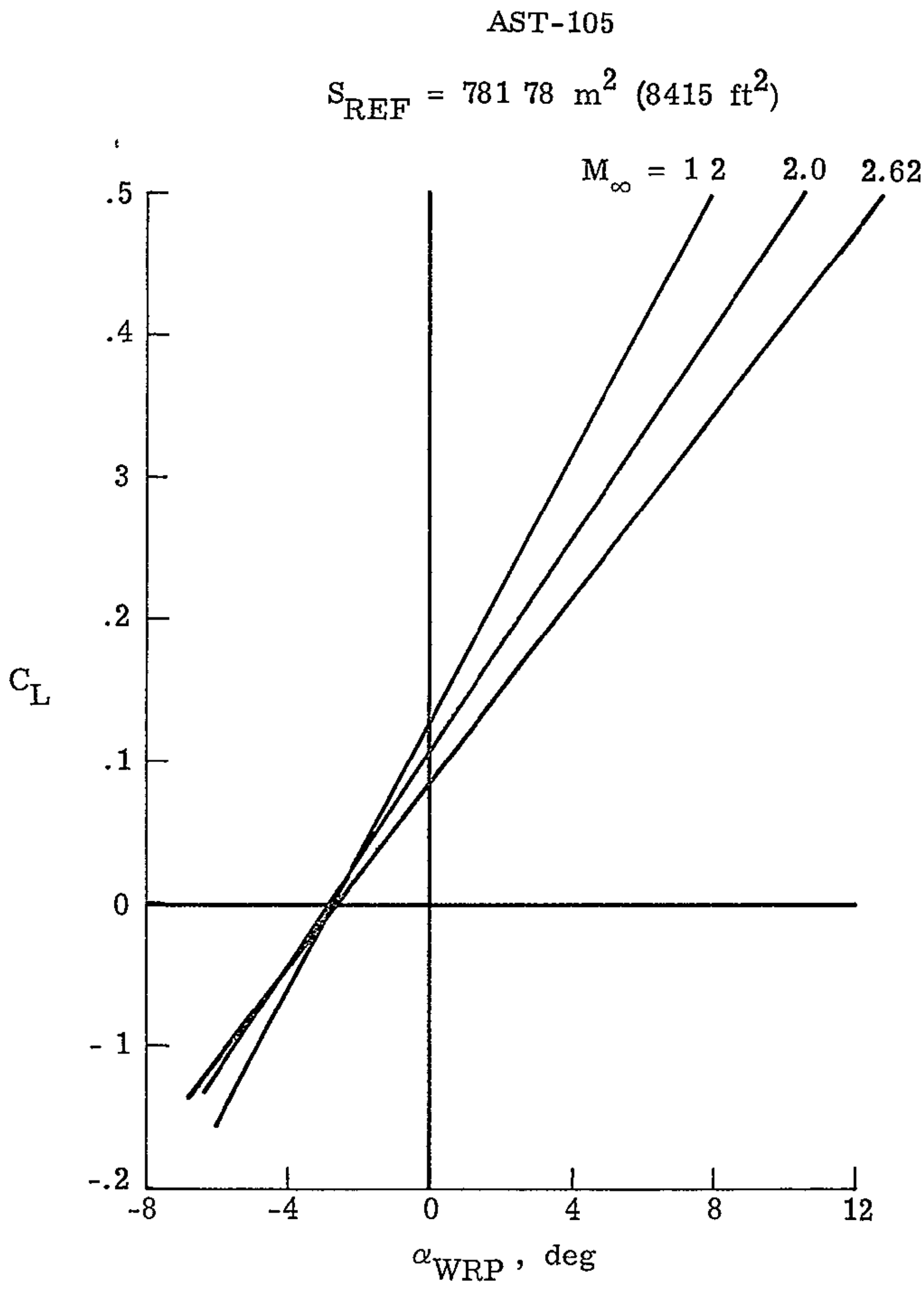

Figure 18.- Typical lift curves.

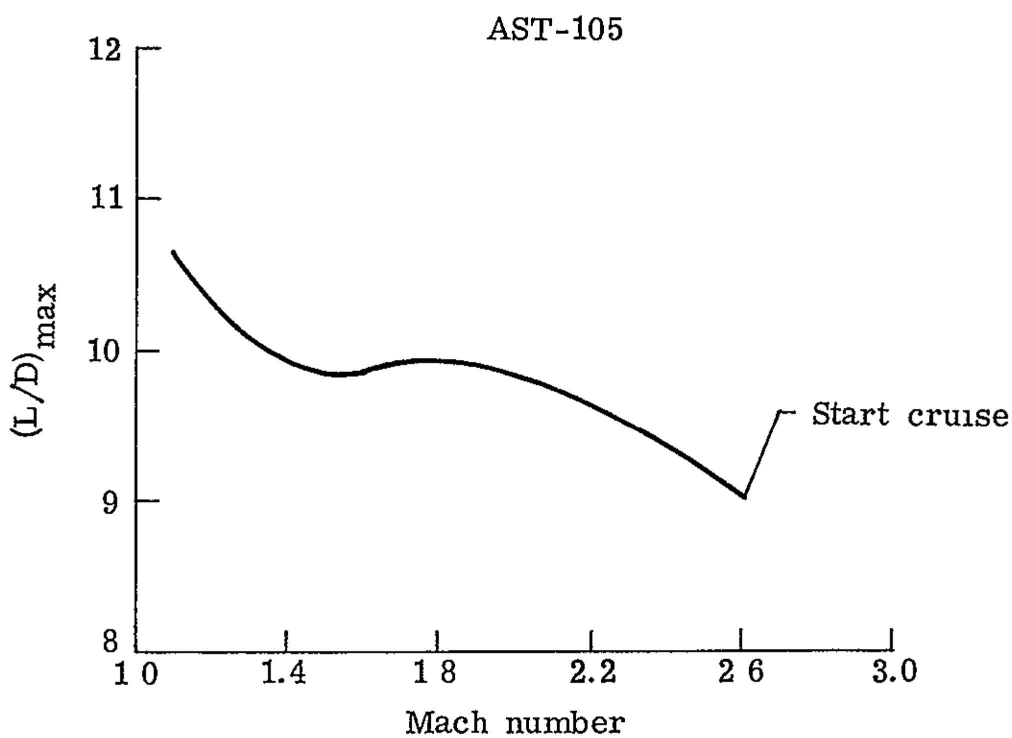

Figure 19.- $(L/D)_{max}$ performance.

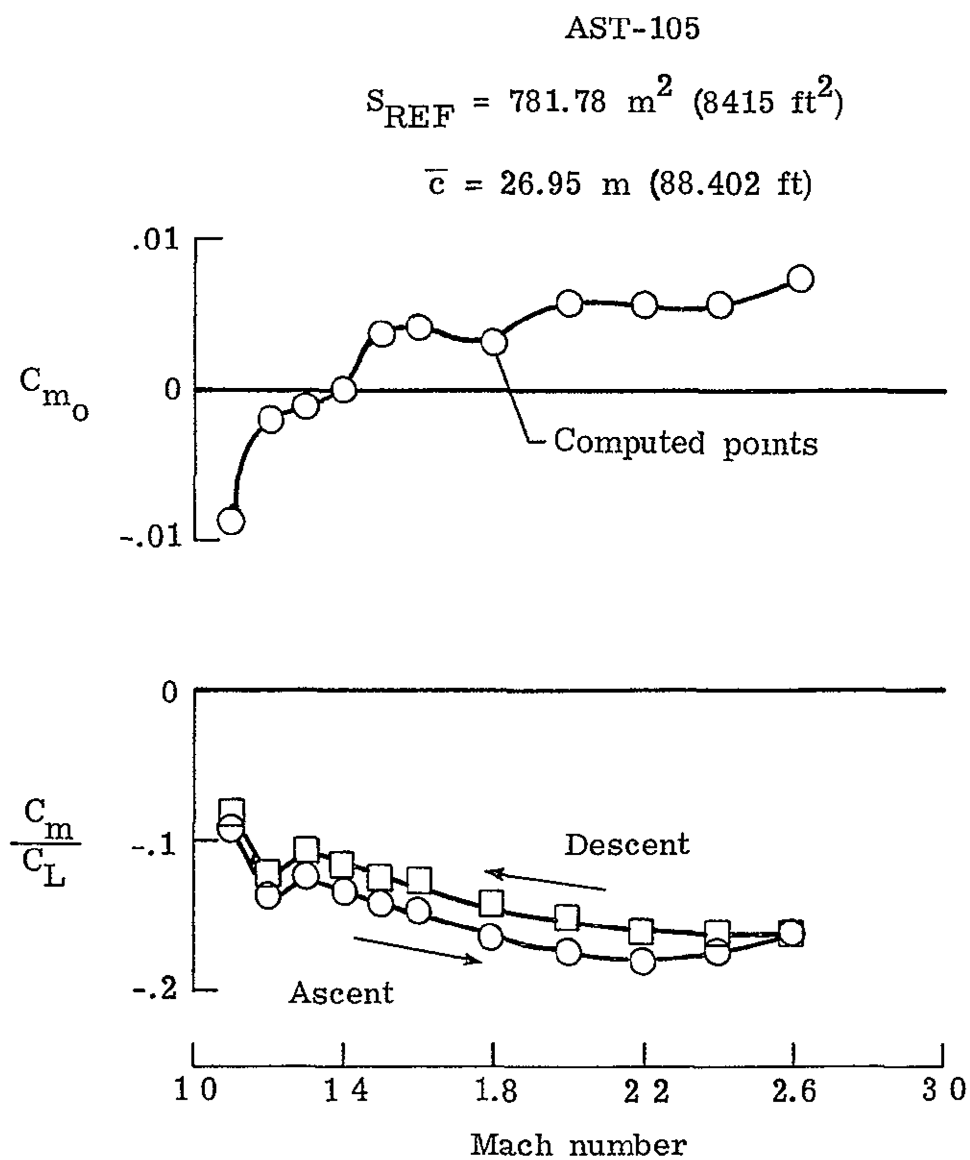

Figure 20.- Pitching moment characteristics

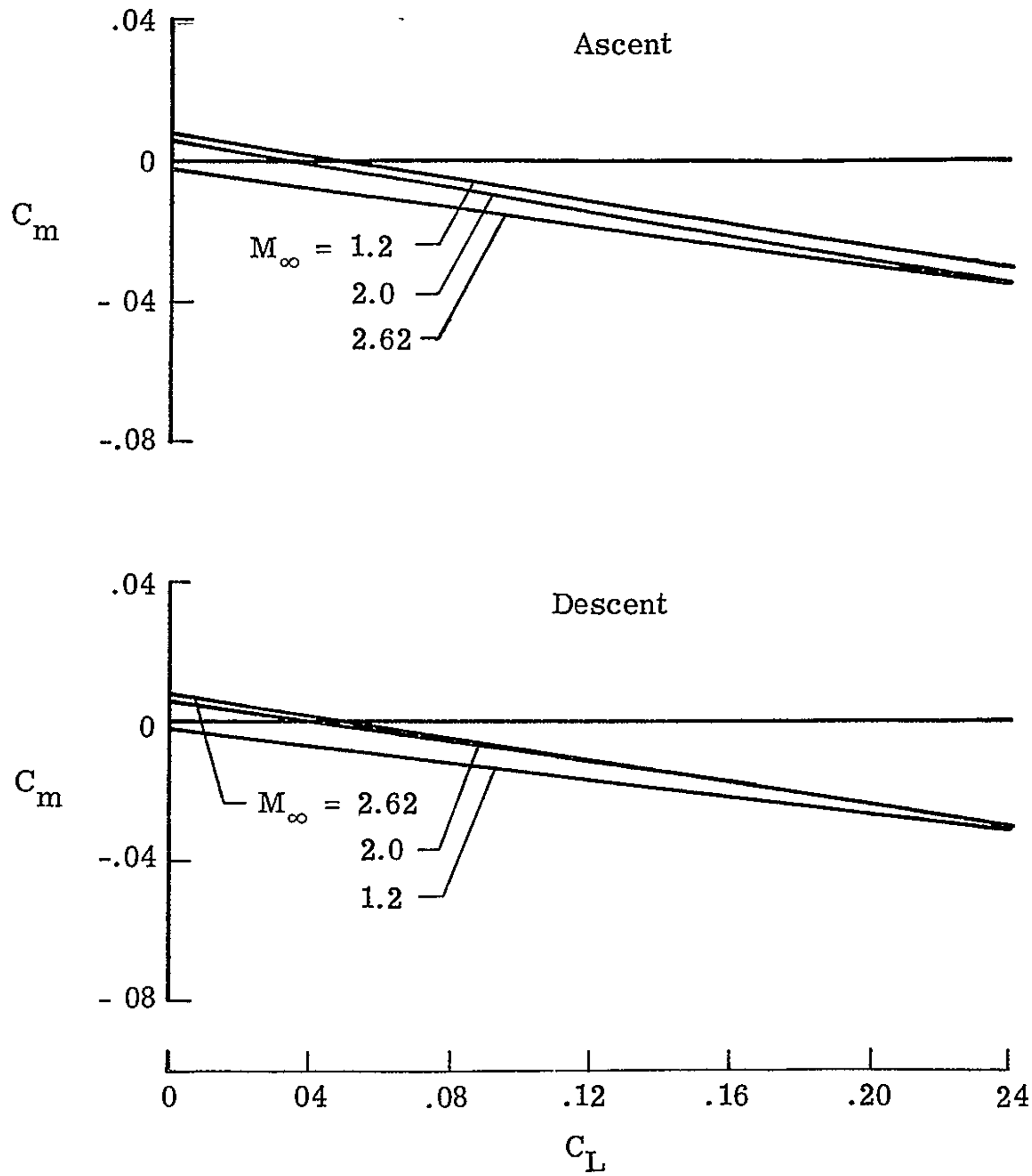

Figure 21 - Typical pitching moment curves.

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15 Supplementary Notes

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16 Abstract

A limited study in the use of theoretical methods to calculate the high speed aerodynamics of arrow wing supersonic cruise configurations has been conducted. The study consisted of correlations with existing wind-tunnel data at Mach numbers from 0.8 to 2.7, using theoretical methods to extrapolate the wind-tunnel data to full-scale flight conditions, and presentation of a typical supersonic data package for an advanced supersonic transport application prepared using the theoretical methods. A brief description of the methods and their use is also given.

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