Aerodynamic Analysis of Supersonic Aircraft with Subsonic Leading Edges

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Methods for the preliminary analysis and design of supersonic aircraft intended for operation during the 1990 time period pose a problem for computational methods presently in use. Many of these aircraft exhibit a high degree of configuration blending to achieve the required performance. Also, many of these vehicles have wings whose leading edges are subsonic. A modeling technique, which uses three different representations of an aircraft configuration, was used to estimate the aerodynamic forces. A method of calculating leading-edge thrust for highly swept wings is presented and applied. For the aircraft configurations considered the test-theory comparisons were favorable. These comparisons indicate that this technique is bounded in its application by both low and high supersonic speeds.

Nomenclature

b	= airplane span
c	= local chord length
C	= $\lim_{X-X_{l.e.}} u\sqrt{X-X_{l.e.}}$
C_D	= drag coefficient
C_{D_f}	= skin friction drag coefficient
C_{D_L}	= drag due to lift (wave + vortex) coefficient
C_{D_W}	= zero lift wave drag coefficient
C_l	= see Eq. (10)
C_L	= lift coefficient
C_M	= moment coefficient
C_s	= suction force per unit span length (directed normal to wing leading edge)
C_t	= thrust force per unit span length (measured in streamwise direction)
C_T	= leading-edge thrust
H,I,J	= see Eq. (13)
i	= $\sqrt{-1}$
I.P.	= imaginary part
K,L,M	= see Eq. (15)
k	= $\sqrt{1-\beta^2/\tan^2\Lambda}$
m	= $\beta\cot\Lambda$
M	= freestream Mach number
p_∞	= freestream pressure
Re	= Reynolds number
S	= reference wing area
T	= thrust per unit streamwise length of wing leading edge
u	= x component of perturbation velocity
V	= freestream velocity
w	= z component of perturbation velocity
x	= longitudinal distance, see Fig. 1
$X,X_{l.e.}$	= see Fig. 1
z	= vertical distance, positive upwards
α	= angle of attack, deg
β	= $\sqrt{M^2-1}$
Δp	= $(p-p_\infty)$
Λ_{loc}	= local leading-edge wing sweep (90 deg = sweep)
η	= semispan function $y/(b/2)$
ρ_∞	= freestream density

I. Introduction

THERE has been considerable Air Force interest[1] shown recently in the development of supersonic cruise vehicle technology. At the present time the technology base available for preliminary analysis and design of supersonic aircraft intended for operation during the 1985 to 1990 time frame requires considerable extension and improvement. One computational system recently developed by NASA and the Boeing Company for the analysis and design of supersonic aircraft is based on the Carlson-Middleton "Mach-box" method.[2-5] Originally, this program was developed for use on SST-type aircraft, in which the fuselage is small in proportion to the wing and, hence, the individual aircraft components were easily distinguishable from one another.[6,7]

Most of these programs, such as the Carlson-Middleton code, are based on linear theory equations and boundary conditions. Such computer programs assume a slender fuselage, a thin wing, and engine(s) that are easily distinguishable from the rest of the aircraft. A highly blended aircraft configuration does not necessarily fit within these assumptions. This can lead to considerable computational difficulty in the analysis of such an aircraft.

Many of these aircraft have highly swept wings. At low supersonic speeds this necessitates consideration of an aerodynamic phenomenon known as leading-edge thrust.[8-12] Test data for several aircraft configurations indicate that theoretical drag estimates based on linearized potential theory, without considering leading-edge thrust, can be off by an appreciable amount at low supersonic speeds. This paper presents a modeling technique used to overcome the described difficulties.

II. Scope and Methods of Approach

Through the use of a suitable modeling technique it was possible to extend linear theory to estimate the aerodynamic forces on a highly blended aircraft configuration.[13-15] The Carlson-Middleton program has three modules which can be used to calculate zero lift wave drag, skin friction, and drag due to lift (wave + vortex). The basic structure of this program is described in Ref. 2. In the present study it was found that by using a different representation of the aircraft in each of the program modules a favorable test-theory comparison can be obtained.

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Index categories: Aerodynamics; Supersonic Flow; Computational Methods.

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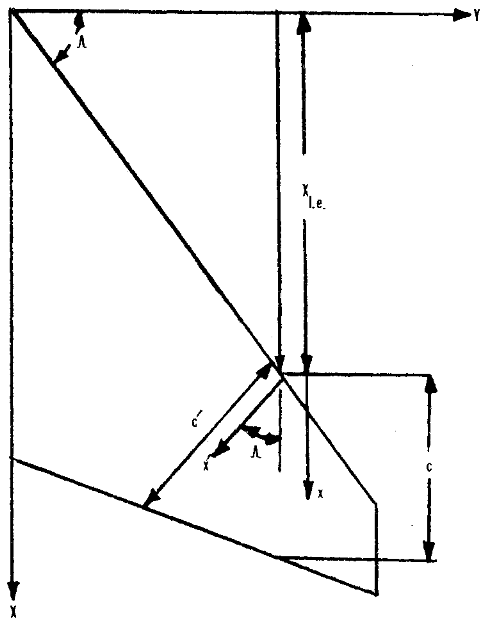


Fig. 1 Coordinate system defining $F(C_p)$.

Many of the aircraft configurations currently being considered have a highly cambered wing and fuselage. When calculating the zero lift wave drag directly from the given geometric input data, an incorrect area distribution can result. This, in turn, usually leads to drag overestimation. It was found that by decambering the fuselage, wing, and carrying the inlet streamtube forward along the fuselage, the estimated value for zero lift wave drag, when combined with skin friction drag and drag due to lift, results in a much better overall estimate of aircraft drag than would otherwise be obtained.

Skin friction is based on the wetted area of the aircraft. When detailed three-dimensional measurements of an aircraft configuration are available, as, for instance, those obtained from a Cordax machine, the skin friction estimate which is based on strip theory will give satisfactory results. When such a description is not available, then the skin friction may be accurately estimated by using the cross-sectional area distribution of the aircraft. This allows an accurate estimation of the fuselage perimeter to be made.

Accurate calculation of drag due to lift requires a considerable amount of care. Whenever possible, detailed drawings of the general arrangement of the airplane should include a number of cross-sectional layouts of the fuselage, wing, and engine inlet(s). A geometrical description of each component of the aircraft along with a description of the juncture location between the fuselage, wing, and inlet(s) needs to be made. This can prove to be a difficult task for a highly blended aircraft configuration. A juncture location must be assigned between the wing/body/inlet to assure accurate representation of the fuselage camber line and fuselage/wing camber surface alignment. This is done in order to attain optimum aerodynamic performance for a given mission. Of particular importance is the need to accurately align the fuselage camber line and inner camber surface of the wing. Misalignment of the fuselage and wing can lead to grossly inaccurate estimates of the drag due to lift. In this procedure the engine inlet and duct can be represented by one, or more, axisymmetric bodies of revolution.

As was previously mentioned, leading-edge effects have been computed and found to exert a profound influence on estimating the aerodynamic forces on an aircraft.[8,14] Leading-edge thrust, as an aerodynamic phenomenon, shall now be considered and discussed. Conceptually, leading-edge thrust originated from considering the mathematical singularity which arises at the leading edge of a two-dimensional wing section. The method described is an extension of work done by Jones.[9,17] An analysis for a three-dimensional wing proceeds from considering the flow over the two-dimensional airfoils in a streamwise direction. One may then extend this analysis to include the flow over a cambered airfoil of finite thickness with a finite nose dimension.[8,9,17] Calculation of the total drag on two-dimensional airfoils will be made, assuming that the pressure and lift distribution are known; this will then be extended to consider a three-dimensional wing.

The vortex drag of a wing traveling at supersonic speeds is directly related to the induced drag resulting from similar distribution of lift at subsonic speeds. This approximation stems from the fact that at supersonic speeds, the form of the vortex wake following a given lift distribution and the induced field of motion in the vicinity of the wake far behind the wing is the same as at subsonic speeds. Total drag is found by integrating the product of the lift and inclination of the stream lines over the airfoil surface. In drag calculations the effect of the leading-edge suction must be included. By using the principle of superposition, the lifting pressure of a two-dimensional wing may be treated by considering camber or angle of attack of the mean line or thickness pressures for a symmetrical thickness distribution. Discontinuities in u and w may be dealt with separately and one is led to

$$C_{p_{\text{Lift}}} = \frac{\Delta p_l - \Delta p_u}{\frac{1}{2}\rho_\infty V^2} = \frac{4u}{V} \tag{1}$$

and

$$\frac{\mathrm{d}z}{\mathrm{d}x} = \frac{w}{V} \tag{2}$$

For a two-dimensional flat plate the integral for the drag takes the form[17]

$$D = 2\int_c \Delta p \frac{\mathrm{d}z}{\mathrm{d}x}\,\mathrm{d}x = -2\int_c \rho_\infty uw\,\mathrm{d}x \tag{3}$$

The product appearing in Eq. (3), which is the differential drag on an infinitesimal element may be represented as

$$-2uw = \text{I.P.}\,(u - iw)^2 \tag{4}$$

By considering an analytic function f of a complex variable, the velocity function for a two-dimensional flat plate inclined at an angle α relative to the freestream is[14,17]

$$f(\xi) = \frac{u - iw}{V\alpha} = i - \sqrt{\frac{1-\xi}{1+\xi}} \tag{5}$$

and

$$(u - iw)^2 = V^2\alpha^2\left(\frac{1-\xi}{1+\xi} - 2i\sqrt{\frac{1-\xi}{1+\xi}} - 1\right) \tag{6}$$

Leading-edge thrust is found by evaluating the contribution to the integral for drag at $\xi = -1$. This is found by considering the integral

$$D = \text{I.P.}\int_c \rho_\infty V^2\alpha^2\left(\frac{1-\xi}{1+\xi} - 2i\sqrt{\frac{1-\xi}{1+\xi}} - 1\right)\mathrm{d}\xi$$

The leading-edge singularity contributes the following term to the drag

$$D_{\text{l.e.}} = -2\pi\rho_\infty V^2\alpha^2$$

This analysis may be extended as follows: when the complex function for velocity takes the form just presented, the leading-edge contribution will be

$$D_{\text{l.e.}} = \text{I.P.}\rho_\infty \int_c \left[\frac{f^2(\xi)}{1+\xi} + \begin{matrix}\text{nonsingular}\\ \text{terms}\end{matrix}\right] \mathrm{d}\xi \tag{7}$$

When $\xi \to -1$, Eq. (7) will give

$$D_{\text{l.e.}} = -\pi\rho_\infty c^2$$

with

$$c = u\sqrt{X - X_{\text{l.e.}}}$$

This may now be extended to three dimensions by considering a highly swept wing whose leading edge is given by Λ. Now, making use of Eq. (7) and accounting for compressibility effects, the differential thrust on a streamwise portion of the wings' leading edge becomes [17]

$$\mathrm{d}T/\mathrm{d}x = \pi\rho_\infty\sqrt{1-m^2}\, c^2 \tag{8}$$

and

$$\mathrm{d}T/\mathrm{d}y = \pi\rho_\infty\sqrt{1-m^2}\, c^2 \tan\Lambda \tag{9}$$

The pressure distribution of a wing with a subsonic leading edge in supersonic flow is found to exhibit an inverse square root behavior in a chordwise direction at a given spanwise station. [9,10,17,19] Theoretical predictions and experimental data show substantial agreement on this point. [20] A modified form of the pressure distribution will now be introduced such that inverse square-root variations will appear linear. The modified form is as follows

$$C_l = \lim_{x/c \to 0} -\frac{Cp}{4}\sqrt{\frac{X - X_{\text{l.e.}}}{C_{\text{loc}}}} = \frac{u}{V}\sqrt{\frac{X - X_{\text{l.e.}}}{C_{\text{loc}}}} \tag{10}$$

from which one readily finds that

$$C_l = -C/V\sqrt{C_{\text{loc}}} \qquad \text{and} \qquad C^2 = C_l^2 V^2 C_{\text{loc}}$$

Substitution into Eq. (8) will give

$$\frac{\mathrm{d}}{\mathrm{d}y}(C_T) = \frac{\mathrm{d}}{\mathrm{d}y}\left(\frac{T}{qS}\right) = \frac{\pi\rho_\infty}{qS}\sqrt{1-m^2}\, C_l^2 C_{\text{loc}} V^2 \tan\Lambda_{\text{loc}}$$

$$C_T = 2\int_0^1 \frac{b}{2S}\frac{2\pi}{S}\sqrt{1-m^2}\, C_l^2 C_{\text{loc}} \tan\Lambda_{\text{loc}} \mathrm{d}\eta$$

where $\eta = Y/(b/2)$.

Leading-edge thrust is a theoretical phenomenon related to a mathematical singularity at the leading edge of a wing. Thin airfoil theory states that leading-edge suction force, being infinite for a zero thickness wing, has the same product (or thrust) as an airfoil of finite dimension. A limiting case of the type described does not occur in the real world. If the radius of the airfoil nose is extremely small, flow separation will occur at very small angles of attack. [17,20]

Test data were used to assist in computing leading-edge thrust. [8,10,11] They serve as a basis for determining the percent of leading-edge thrust for a given configuration. The wings are considered as being either thin or having a moderately rounded leading edge (approximately 0.05 chord). This method is applicable to a thin wing with a slightly rounded leading edge. The contribution of camber and flat-plate pressure distribution needs to be considered in this type of calculation. [17] Camber is independent of angle of attack and, hence, invariant with respect to lift coefficient. The flat-plate contribution does depend on α and must be accounted for. The flat-plate pressures considered are the lifting pressures and are measured as the difference between the upper and lower surface pressure. This is done in the following manner:

$$\Delta C_p = \Delta C_{p_{\text{cam}}} + \alpha \cdot \Delta C_{p_{\text{f.p.}}} \tag{11}$$

For a given value of C_L the camber and flat plate pressures are computed directly in the form presented in Eq. (11).

Leading-edge thrust can then be estimated from the pressure distribution on the wing. Mathematically, the pressure distribution at the leading edge of a flat wing has a singularity. This result was deduced by Kaplun and Lagerstrom by employing the method of matched asymptotic expansions. [19] The first term arising from their solution is

$$\Delta C_p = C_p(x,0^-) - C_p(x,0^+) = 4\alpha\sqrt{(C-x)/x} \tag{12}$$

This expression has an inverse square-root singularity at the leading edge and approaches zero at the trailing edge as the square root of the distance. The C_p in Eq. (12) is that of a flat plate. Hence, Eqs. (11) and (12) may be phenomenologically related to one another.

Another approximation will now be introduced and discussed. When dealing with the pressure due to a flat plate and camber, it is reasonable to consider the forward portion of the wing, since the mathematical singularity relating to leading-edge thrust is localized at the leading edge. This, in turn, means that the forward portion of the wing may be considered as a flat plate oriented at an angle α' relative to the freestream. The local angle α' is, in general, different from α and varies with spanwise station η. The C_p distribution of a wing may be represented as

$$\Delta C_p = H\left(\frac{x}{c}\right)^{-1/2} + I\left(\frac{x}{c}\right)^{1/2} + J\left(\frac{x}{c}\right)^{3/2} + \ldots \tag{13}$$

which suggests forming a function $F(C_p)$ in a streamwise direction to the flow of the form:

$$F(C_p) = \frac{-C_p}{4}\sqrt{\frac{X - X_{\text{l.e.}}}{C_{\text{loc}}}} \tag{14}$$

Using Eq. (12) one has

$$\Delta C_p = 4\alpha\sqrt{\frac{c-x}{x}} = \frac{4\alpha}{\sqrt{x/c}}\sqrt{1 - \frac{x}{c}}$$

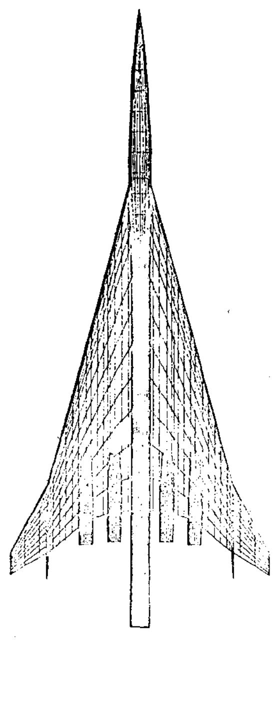


Fig. 2 NASA SCAT-15F.

with

$$\sqrt{\frac{1-x}{c}}=1-\frac{1}{2}\left(\frac{x}{c}\right)-\frac{1}{8}\left(\frac{x}{c}\right)^2-\frac{1}{16}\left(\frac{x}{c}\right)^3+0\left[\left(\frac{x}{c}\right)^4\right]$$

for $(x/c)^2<1$. One may then write

$$\Delta C_p=H\left(\frac{x}{c}\right)^{-1/2}+I\left(\frac{x}{c}\right)^{1/2}+J\left(\frac{x}{c}\right)^{3/2}+\ldots$$

where

$$H=4\alpha \qquad I=-2\alpha \qquad J=-\alpha/2.$$

Now consider a swept wing in supersonic flow as depicted in Fig. 1, where

$$\sqrt{X-X_{\text{l.e.}}}=\sqrt{x'/\cos\Lambda}$$

Now defining a function F normal to the leading edge of the wing

$$F(C_p)=\frac{\Delta C_p}{4}\sqrt{\frac{X-X_{\text{l.e.}}}{c}}=\frac{\Delta C_p}{4}\sqrt{\frac{x}{c}}=\frac{\Delta C_p}{4\sqrt{c/\cos\Lambda}}\sqrt{\frac{x'}{\cos\Lambda}}$$

and

$$\Delta C_p=4\alpha'\left(\frac{x'}{c'}\right)^{-1/2}-2\alpha'\left(\frac{x'}{c'}\right)^{1/2}-\frac{3}{16}\alpha'\left(\frac{x'}{c'}\right)^{3/2}+\ldots$$

For a flat plate

$$\Delta C_p=K\left(\frac{x}{c}\right)^{-1/2}+L\left(\frac{x}{c}\right)^{1/2}+M\left(\frac{x}{c}\right)^{3/2} \qquad (15)$$

$$F(C_p)\sim\alpha'-\frac{\alpha'}{2}\frac{x}{c}-\frac{3}{64}\alpha'\left(\frac{x}{c}\right)^2 \qquad (16)$$

In the region $x/c<<1$ the ΔC_p term is dominated by the first term in Eq. (15). Hence the F function is dominated by the term α' in the same region. The linear variation that occurs in F is a result of the second term (the square-root behavior in ΔC_p behavior). F is essentially a constant plus a linear function of x with the leading-edge term approximately equal to α.

Calculations of leading-edge thrust will be performed in the next section using F (C_p) as defined in Eq. (14). As defined, $F(C_p)$ has a finite value at the leading edge of the wing. Mathematically, the pressure at the leading edge exhibits an inverse square root behavior. A function C_l which is the limiting value of $F(C_p)$ as $X/C\to 0$ is used in determining the suction at the leading edge of the wing.

The function C_l is determined by forming a straight line through points as will be discussed in the next section. The points for X/C less than about 30% chord were used to determine the line for calculating C_l. In general, most of the points plotted in this region come reasonably close to approximating a straight line.

Experimental data indicate that some fraction of theoretical suction force should actually occur in a wind tunnel or flight test. A combination of wind-tunnel data for SCAT 15F[6] and a collection of data for several wings[8,11,12] over a wide range of supersonic speeds was used. These data were used to establish a fraction of the maximum theoretical leading-edge thrust to be applied to each drag polor at a given lift.

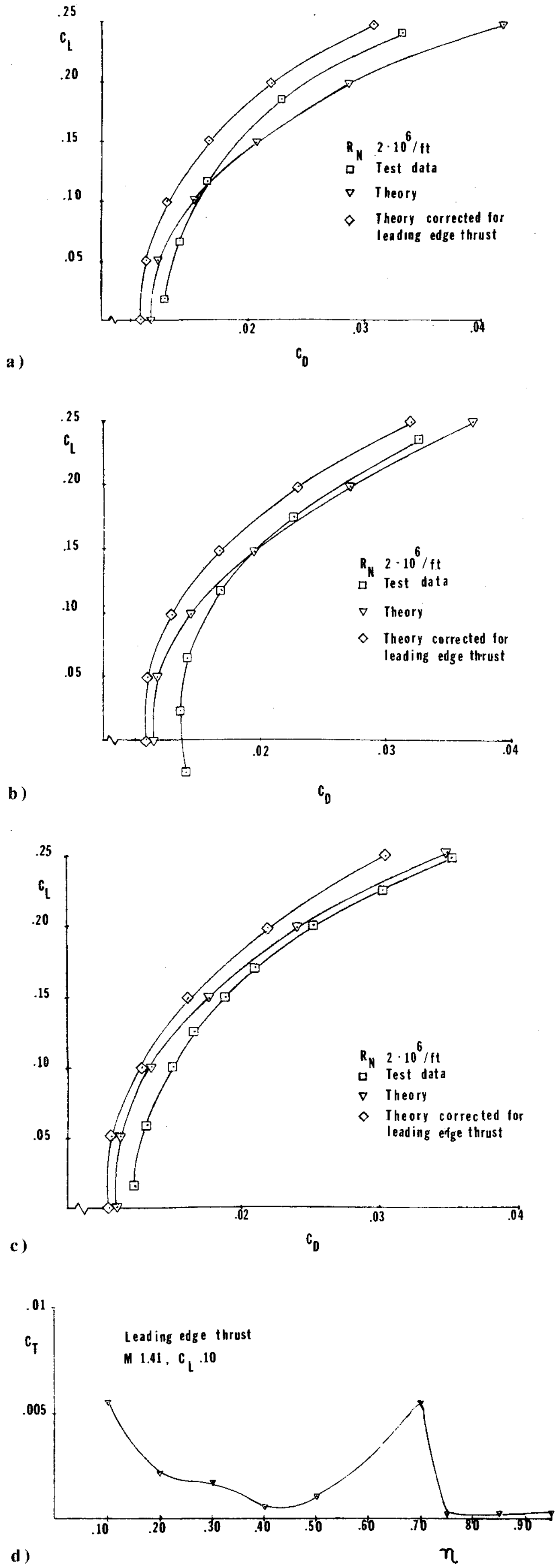


Fig. 3 NASA SCAT-15F test-theory comparison: a) $M=1.20$, b) $M=1.30$, c) $M=1.41$, d) leading-edge thrust calculation, $M=1.41$, $C_L=0.10$.

After determining C_l, the suction force at the leading edge may be determined by the equation

$$C_s = \frac{b}{2s} \cdot \frac{2\pi}{\cos} \Lambda_{\mathrm{loc}} \cdot C_{\mathrm{loc}} \tan \Lambda_{\mathrm{loc}} k C_l{}^2$$

which is similar to Eq. (9) except for the $\cos\Lambda_{\mathrm{loc}}$ which appears due to the fact that the leading-edge suction acts normal to the leading edge of the wing. The next step is to determine localized thrust per unit span by using the relation $C_t = C_s \cos\Lambda$. Total leading-edge thrust coefficient is found, finally, using the integral

$$C_T = 2\int_0^l C_t \mathrm{d}\eta$$

Calculations of leading-edge thrust will be presented in the next section.

III. Applications

Application of the computer program and analysis methods presented in the previous section will now be made to several different aircraft configurations. This will include the NASA SCAT-15F, SCIF-4, SCIF-5, and Boeing LES-213.

SCAT-15F

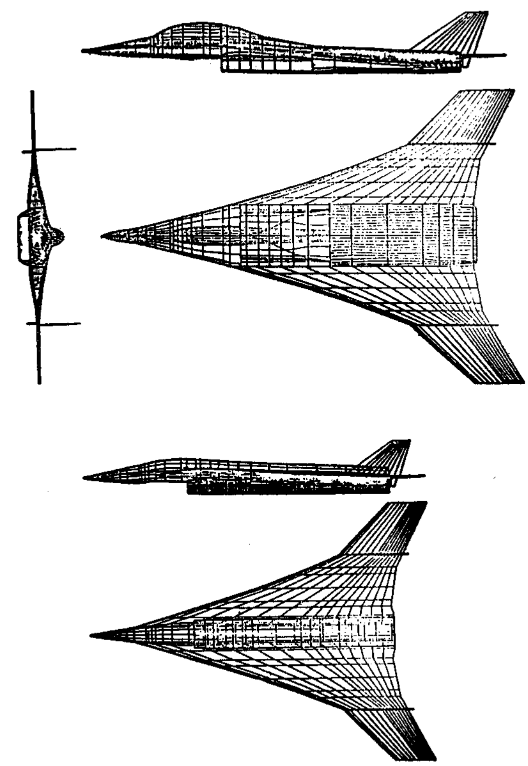
The NASA SCAT-15F (Fig. 2) was developed as a long-range SST. It has a leading-edge sweep of 74 deg and was designed to cruise at $M = 2.70$ with $C_L = 0.085$.[6,7] Analysis of this configuration was initially performed using the representation shown in Fig. 2. The configuration was then represented as a decambered, circular body of revolution with a flat wing mounted in the xy plane. Zero lift wave drag calculations indicated very small differences.

At low supersonic Mach numbers theoretical methods overpredicted the drag on SCAT-15F. A correction for leading-edge thrust, using the method considered in the preceding section was made. Calculation of C_T and application to drag polars at $M = 1.20$, 1.30, and 1.41 are shown in Fig. 3.

SCIF-4

SCIF-4 (Fig. 4), is an outgrowth of an SST developed several years at NASA Langley.[21] It has a highly swept leading edge and was designed for optimum performance at $M = 1.80$ and $C_L = .12$. Analysis of this configuration was performed at $M = 1.20$, 1.80, and 2.16. An increment due to inlet diverter drag was also computed.

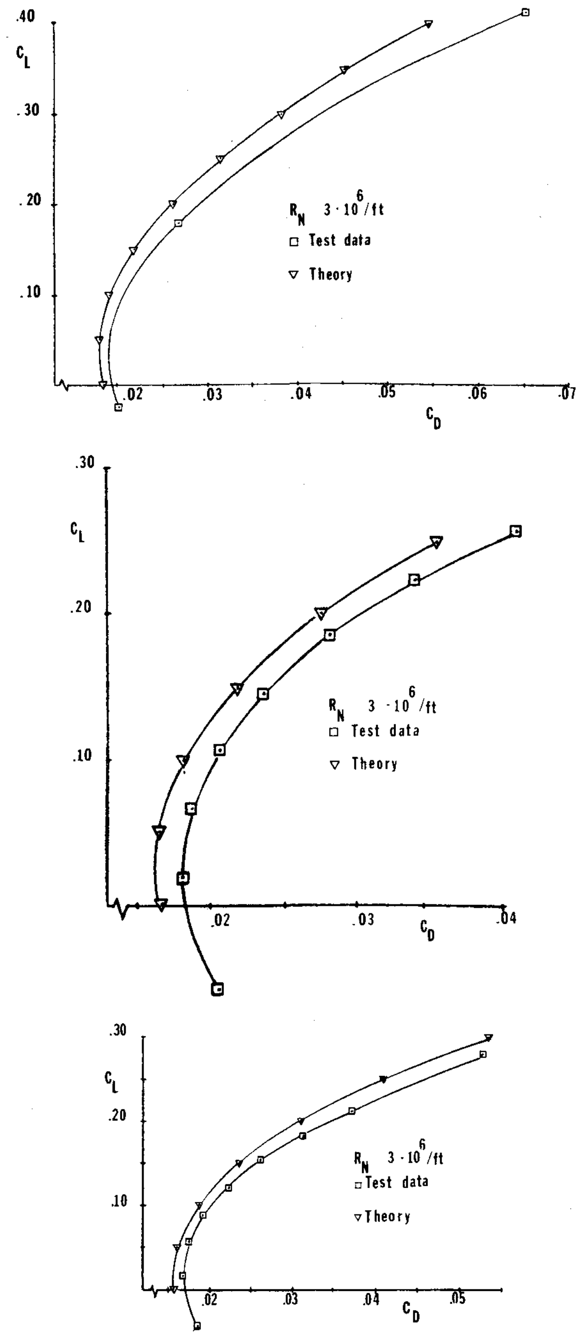
Drag due to lift was computed by dividing the wing and fuselage at $x = 4.55$ and $y = 0.827$ in. from the nose tip. A camber line was computed using the coordinates of the centroids of the fuselage cross-sectional areas. The inlet was defined as two axisymmetric bodies of revolution. Corrections were made at $M = 1.20$ and 1.80 to account for leading-edge thrust acting on the subsonic leading edge. Test-theory comparisons are summarized in Fig. 5.



a)

b)

Fig. 4 NASA SCIF-4: a) wind-tunnel model, b) drag-due-to-lift representation.



a)

b)

c)

Fig. 5 NASA SCIF-4 test-theory comparison: a) $M = 1.20$, b) $M = 1.80$, c) $M = 2.16$.

SCIF-5

The NASA SCIF-5 (Fig. 6), was developed as a fighter aircraft using the SCAT-15F wing planform.[14,15] It is a highly cambered vehicle with an appreciable amount of wing-body blending. Skin friction was computed using the geometric input data obtained by Cordax measurements. Zero lift wave drag and drag due to lift were calculated using the method outlined in the previous section. Test-theory comparisons at $M = 2.60$ and 2.96 are summarized in Fig. 7.

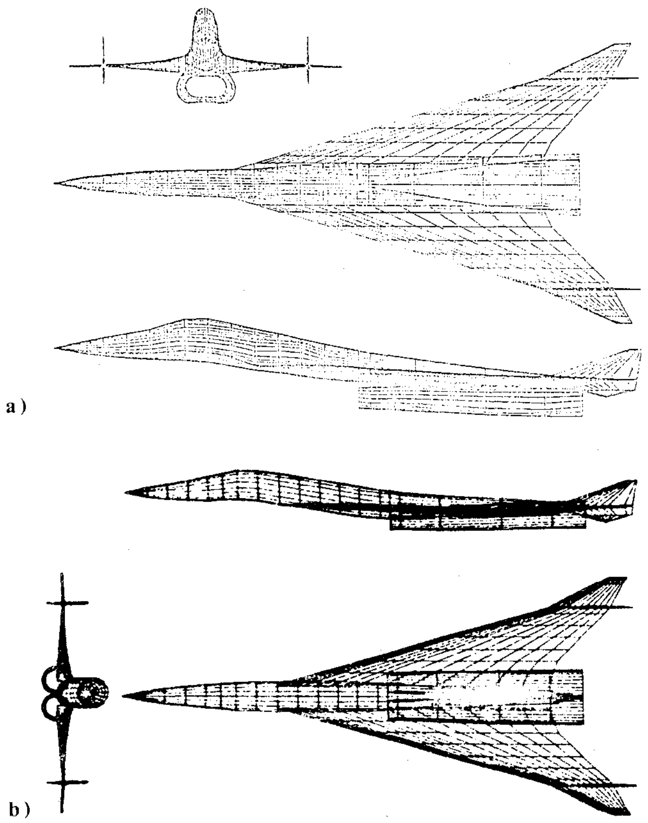


Fig. 6 NASA SCIF-5: a) wind-tunnel model, b) drag-due-to-lift representation.

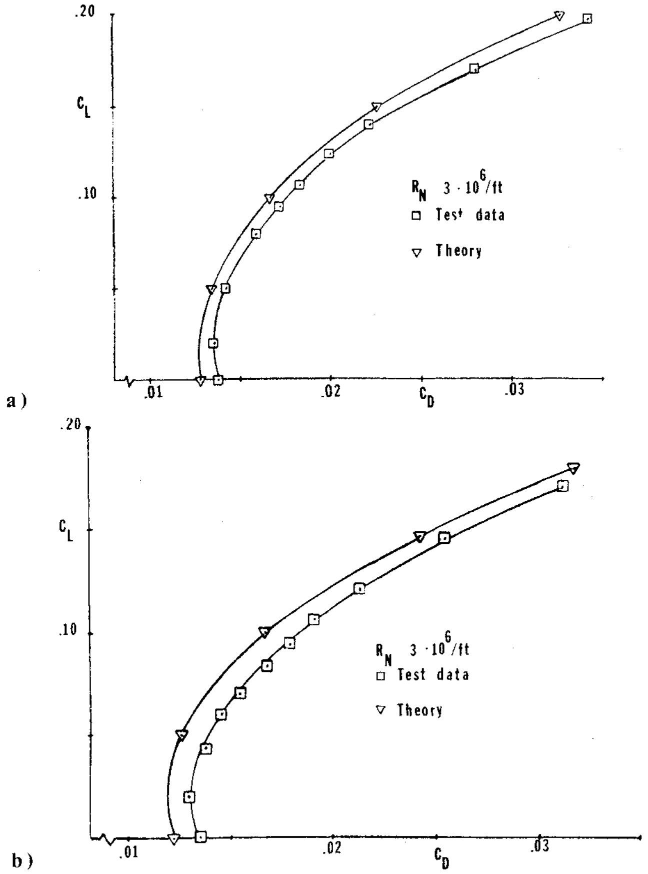


Fig. 7 NASA SCIF-5 test-theory comparison: a) $M = 2.60$, b) $M = 2.96$.

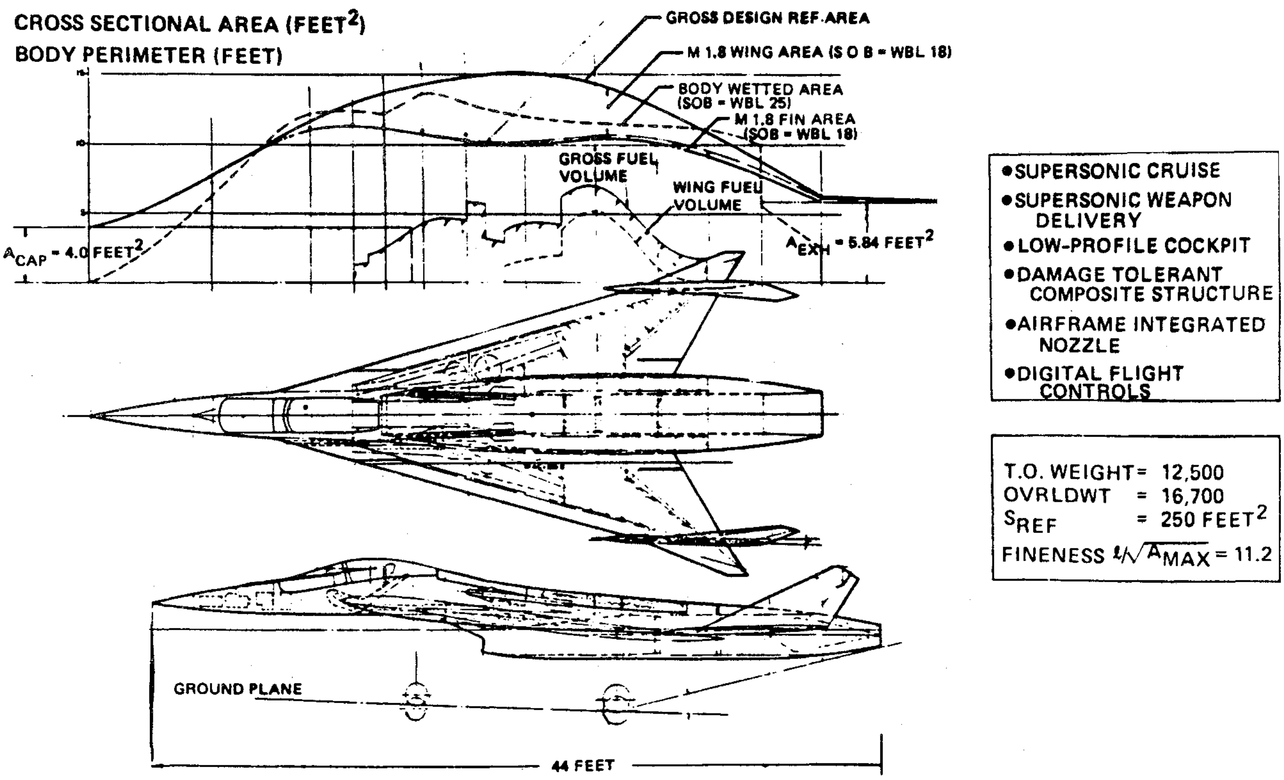


Fig. 8 Boeing LES-213 general arrangement.

LES-213

The Boeing LES-213 (Fig. 8) was developed as part of a study to determine the feasibility of the supersonic cruise concept. The initial vehicle utilized a wing planform similar to that of the NASA SCAT-15F and was designated as LES-210. Several design iterations were performed on this airplane which resulted in LES-213. Design of the aircraft was performed for a Mach number of 1.80, $C_L = 0.10$, and $C_M = 0.03$. The airplane configuration was tested by the Boeing Company.[16] Zero lift wave drag calculations indicated sizable differences in the wave drag as summarized in Table 1. Test-theory comparisons are summarized in Fig. 9 for $M = 1.50$, 1.80, and 2.20.

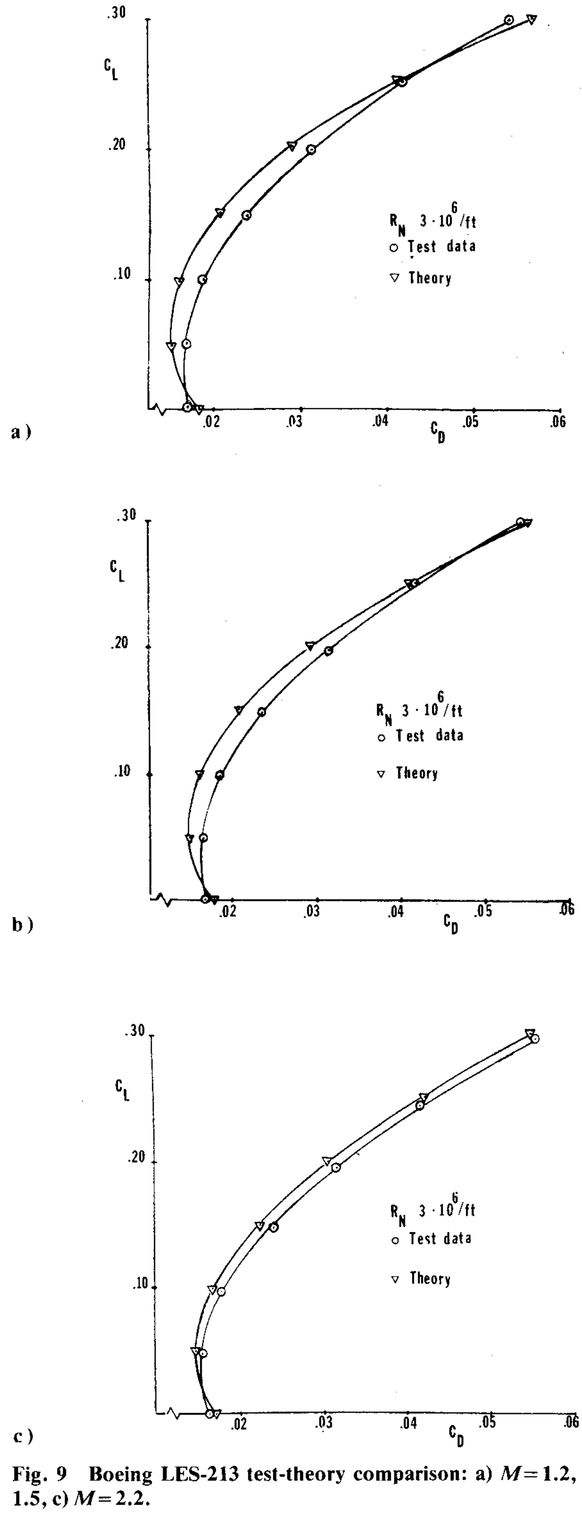


Fig. 9 Boeing LES-213 test-theory comparison: a) $M = 1.2$, b) $M = 1.5$, c) $M = 2.2$.

Table 1 LES-213 zero lift wave drag

M	Cambered	Uncambered
1.50	0.0186	0.0063
1.80	0.0182	0.0061
2.00	0.0177	0.0059
2.20	0.0174	0.0057

IV. Discussion of Results

Comparisons between experimental data and theoretical calculations were, in general, good. Some of the attributes and limitations of the modeling method and computational system will be discussed.

As would be expected at $M = 1.2$-1.4 the test-theory comparisons were not as good as they were for higher Mach numbers. This is due largely to loss of accuracy in the linearized analysis methods at the lower Mach numbers. It should be noted, however, that accounting for leading-edge thrust greatly improves the test-theory comparison on drag due to lift.

At $M = 1.6$-2.6 comparisons were better than at lower speeds. In this speed regime test-theory comparisons for the four vehicles considered were good. The improved correlation may be attributed to the fact that the flow over all, or most, of an aircraft configuration is supersonic. For high supersonic speeds another limitation on linearized theory is encountered.

At $M = 2.96$ and above, test-theory comparisons on SCIF-5 and SCAT-15F deteriorated. Comparisons on SCAT-15F at $M = 3.95$ and 4.63 showed an increasing gap as Mach number increased.[13] At these speeds the linearization and mean surface approximation have a rapidly vanishing range of applicability.

Leading-edge thrust can exert a substantial influence on the drag-due-to-lift characteristics of an aircraft. This is especially so when considering an airplane with highly swept wings at low supersonic speeds. Computations clearly indicate the existence of singularities exhibiting an inverse square root nature in the forward portion of a wing, which is also verified by experimental data.[20]

V. Conclusions

A method of calculating leading-edge thrust for a highly swept wing in supersonic flow has been presented and applied. One must keep in mind the class of aircraft for which this method is suitable. This includes 1) the wing must be slender, i.e. the perturbation velocities remain small compared to the freestream velocity, and 2) the wing must have a subsonic leading edge. Leading-edge thrust must be computed at a sufficient number of stations to adequately define its spanwise variation and distribution.

A method of inputing and analyzing a configuration with an appreciable amount of blending has been described and demonstrated. Limitations of the computer program used in analyzing a highly blended vehicle have been discussed. Based on the configurations considered it seems reasonable to assume that the modeling technique presented has limitations consistent with linearized supersonic theory. The method described and applied to SCIF-4, SCIF-5, and LES-213 resulted in favorable agreement between test and theory.

A need exists for an improved computational method to calculate the flow about a vehicle of arbitrary shape. Difficulties exist in adequately defining the juncture between the wing and body and analyzing the forces acting in this region using linearized potential theory. The situation is further complicated when considering an engine inlet and duct of arbitrary shape. Presently, there is no program that can explicitly evaluate the aerodynamic forces on an engine using linearized aerodynamics. Programs are now evolving that will address these problem areas. These programs employ either exact boundary conditions or nonlinear governing equations.

Acknowledgment

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