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THEORETICAL PREDICTION OF INTERFERENCE LOADING ON AIRCRAFT STORES

PART II — SUPERSONIC SPEEDS

BY
F. DAN FERNANDES

PREPARED FOR
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LANGLEY RESEARCH CENTER
HAMPTON, VIRGINIA 23365
UNDER
CONTRACT NUMBER NAS1-10374

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GENERAL DYNAMICS
Electro Dynamic Division

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SUPERSONIC SPEEDS

SUMMARY

A method is developed for theoretically predicting the loading on pylon-mounted stores in supersonic flow. Linear theory is used, without two dimensional or slender body assumptions, to predict the flow field produced by the aircraft wing, nose, inlet, and pylons. Aircraft shock wave locations are predicted, and their effect on the flow field is included through a transformation of the aircraft geometry. The interference loading is integrated over the store length by considering the local crossflow, its axial and radial derivatives, and buoyancy. Store moment calculations under an F-4 aircraft at Mach 1.2 are compared to wind tunnel data. The method is computerized, and program user information is included. A companion report presents the method in subsonic flow.

CONTENTS

SUMMARY	ii
LIST OF SYMBOLS	iv
LIST OF FIGURES	vi
INTRODUCTION	1
ACKNOWLEDGEMENT	2
METHOD PROCEDURE (supersonic)	3
STORE LOADING IN NONUNIFORM FLOW	4
METHOD EVALUATION	7
METHOD IMPROVEMENT	16
REFERENCES	17
AERODYNAMIC THEORY	18
A. Wing and Pylon Source Distribution due to Crossflow	18
B. Flow Field Due to Each Source Element	25
C. Flow Field Due to Fuselage Nose	30
D. Shock Wave Effects	36
APPENDIX A - Intersection of Missile Traverse Line with Aircraft Shock Waves	A-1
APPENDIX B - Computer Program STORLD User Information	B-1
APPENDIX C - Computer Program SHOCK User Information	C-1

LIST OF SYMBOLS

a	- defined by equation (10b)	FT
C_N	- store normal force coefficient, N/qS	
C_Y	- store yaw force coefficient, Y/qS	
C_m	- store pitching moment coefficient, m/qSD	
C_n	- store yaw moment coefficient, n/qSD	
C_l	- roll moment coefficient, l/qSD	
C_{NL}	- store normal force coefficient per unit length	(FT^{-1})
ΔC	- interference coefficient	
C'	- coefficient in tunnel axis system, pitches but does not roll with the test model	
C_p	- pressure coefficient, $\Delta p/q$	
D	- store reference diameter	
F	- defined by equation (12)	
G_s	- defined by equation (17)	
h	- defined by equation (10a)	FT
M	- Mach number	
q	- dynamic pressure	PSF
R	- local body radius	FT
r	- defined after equation (23)	FT
S	- reference area	FT^2
s	- denotes X, Y, or Z	FT
V_o	- free stream velocity	FT/SEC
u, v, w	- perturbation velocities in X, Y, Z directions, respectively	

u, v_r, v_θ	- perturbation velocities in cylindrical coordinates	FT/SEC
X, Y, Z	- aircraft coordinates	FT
X	- store traverse distance forward from the traverse line reference point	FT
α	- angle of attack	(degrees)
β	$-\sqrt{M^2 - 1}$	
θ	- surface slope from free stream direction	(radians)
θ	- cylindrical coordinate angle	
Σ	- area inside Mach forecone	
ϕ	- velocity potential function	FT ² /SEC
$\Delta()$	- incremental change	
$()_{A/C}$	- aircraft	

LIST OF FIGURES

		<u>PAGE</u>
1.	Aircraft and Store Geometry	8
2.	Interference Coefficient Prediction for Separate Aircraft Components and Comparison with Experiment - ARM Store Under F-4 Outboard Pylon	9
3.	Interference Coefficient Prediction for Separate Aircraft Components and Comparison with Experiment - ARM Store Under F-4 Inboard Pylon	11
4.	Comparison of Theoretical Interference Pitch and Yaw Coefficients with Experimental Data - RAM Store of F-4 Inboard Pylon	15
5.	Values of Aerodynamic Influence Coefficients (AIC's) Versus Box Location in Mach Forecone	23
6.	Order of Computing Mach Box Properties (u or w) on Wing and Diaphragm	24
7.	Comparison of Theory With Experiment for Pressure Coefficient on Axisymmetric Nose	33
8.	Equivalent Body Concept for Improvement of Linear Theory Flow Field Calculations - Supersonic Flow	38
A-1.	Comparison of Computer Shock Shapes with Experimental Data at Transonic Speeds	A-12
A-2.	Prediction of Detached Shock using Love's Method of Nose Modification	A-14
A-3.	Modification of Love's Method for Detached Shocks - Equivalent Cone Body is Constructed Through the Nose Apex and Through the Nose Point Where $\delta = \delta_{DET}$ (Cone Always Has Detached Shock)	A-15

	<u>PAGE</u>
B-1. Program STORLD Logical Flow	B-4
B-2. Program STORLD Input Format (Supersonic)	B-5
C-1. Program SHOCK Logical Flow	C-3
C-2. Program SHOCK Input Format	C-5

INTRODUCTION

The prediction method presented here for supersonic flow is the same as that of Reference 1 for subsonic flow, the only difference being that the aerodynamic theory used to predict the aircraft flow field is modified. Because of the differences in the theory, separate computer programs are used to implement the method in subsonic and supersonic flow. This report presents the aerodynamic theory for supersonic flow, presents the user information for the supersonic computer program, and presents comparisons between theory and experiment. Reference 2 contains the programming details for all the programs (one subsonic, two supersonic).

Much of the information contained in Reference 1 for subsonic flow applies equally well here. This includes the historical development of theoretical prediction methods, the basic procedure used in the method, and the details of predicting store loads in nonuniform flow. This information will not be repeated here, and this report is therefore supplementary to Reference 1.

Reference 1 states that notable omissions to the method thus far are aircraft wing-body interference and store-to-store-to-pylon mutual interference. This is true for the prediction method in supersonic flow as well as subsonic flow. It is hoped that these features will be added at a future time. These omissions do not affect method accuracy in a large number of practical cases. An additional omission, unique to supersonic flow, is that the store bow shock interaction

with aircraft shocks is not considered. This interaction can change the strength and impingement location of the aircraft shock onto the store to some extent. The effect would tend to be more important for blunt-nose stores of low fineness ratio. Neglected also is the possibility that the store bow shock may be reflected by the aircraft back onto the store. This could be significant if a blunt store is carried behind the wing leading edge. However, neglecting the store bow shock will not affect method accuracy in the majority of practical cases and avoids considerable complication of the method.

Additional features of the method which require further investigation are pointed out in the METHOD EVALUATION section, and are summarized under METHOD IMPROVEMENT. Although the method does have much room for improvement, its accuracy is sufficient to render it useful. This report enables the reader to apply the computerized method to his own particular problems. The computer codes are available through COSMIC (computer software management and information center). Requests should be directed to "COSMIC, University of Georgia, Athens, Georgia, 30601."

ACKNOWLEDGEMENT

The author thanks Dr. Darryl Trulin for his aid in formulating the method for predicting shock wave locations, as contained in Program SHOCK.

METHOD PROCEDURE (supersonic)

The method consists of predicting the flow field about the aircraft by using linear theory, with corrections for shock wave effects. FORTRAN computer programs have been written to calculate shock locations and to calculate singularity strengths to match the boundary conditions of a general jet fighter-bomber aircraft configuration in supersonic flow. The disturbance velocities are calculated by the program over the length of the store for each store position of interest, which gives the interference angle of attack and static pressure field as seen by the store. The program then integrates the effect of the variable flow field over the length of the store to give the interference coefficients. These interference coefficients are to be added linearly to the store free-air coefficients to obtain the total store loading at angle of attack during separation.

To predict the aircraft flow field, a supplementary computer program is first used to predict the locations of shock waves emanating from the nose, inlet, and pylons. APPENDIX A presents the theory used in these calculations. This shock position information is used to transform the aircraft into an "equivalent" linear theory representation. This transformation is explained in the AERODYNAMIC THEORY section. The geometry of the transformed aircraft is then input into the main prediction program. This program represents the aircraft wing, nose, inlet, and pylons by linear theory and calculates

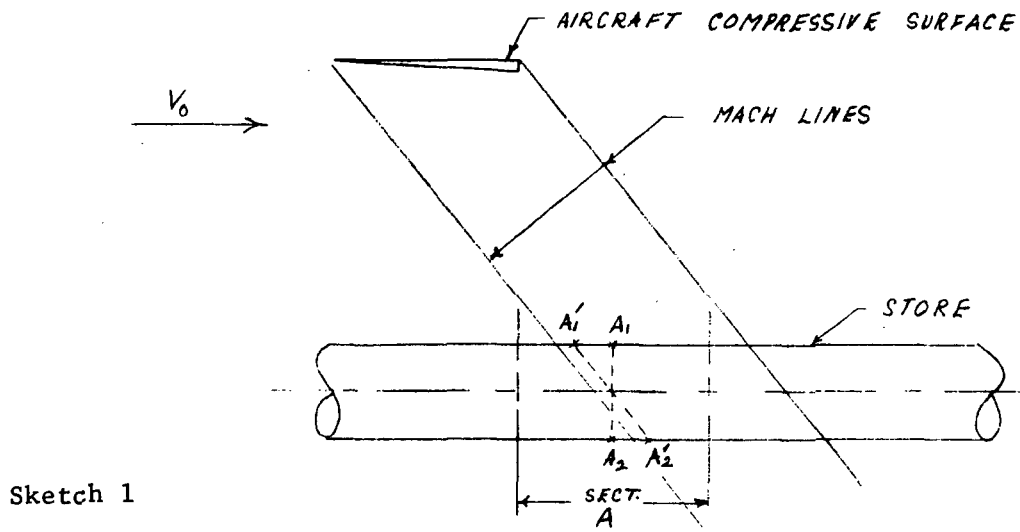
the resulting interference flow field as explained in detail also in the AERODYNAMIC THEORY section.

STORE LOADING IN NONUNIFORM FLOW (Supersonic)

The method for integrating the interference loading over the store in supersonic flow is nearly identical to that explained in Reference 1 for subsonic flow. The details of the method will not be repeated here. There is, however, one important feature of the store loading in supersonic flow which requires special attention; this is the loading on the store body produced by the variable static pressure in the interference field, termed the buoyancy loading.

The store buoyancy loading is obtained by integration of the static pressure field over the area of the store body. This is done numerically by computing the static pressure on opposite sides of the store at each of several axial sections. The buoyancy loading is proportional to the difference of static pressures from one side of the body to the other, integrated over the body length, as explained in Reference 1, Appendix E. The precision of the programmed numerical integration is adequate in subsonic flow because the static pressure varies smoothly and gradually with field position. However, the static pressure in supersonic flow can change greatly with small change in field position. This tends to introduce large and erroneous buoyancy loading predictions due to lack of precision in the numerical integration technique.

The sketch below illustrates this problem; an aircraft compressive surface induces a high pressure zone across the store between Mach lines. Pressures computed at points A_1 and A_2 indicate a large and



erroneous downward buoyancy loading on the store at section A. Two field points per section are not enough to accurately compute the true buoyancy loading on the store in such a variable pressure field unless a prohibitively large number of store sections are used.

To alleviate this problem, the supersonic prediction program was coded to shift the field points along Mach lines passing through the store axis at each section midpoint. These are shown as points A_1' and A_2' on the sketch. This method has the ability to calculate the loading due to pressure decay along Mach lines, but it does not account for the moment on the store produced by the fact that the

pressure field crosses the store at an oblique angle. The results of the calculations using this method showed that the buoyancy loading prediction was of negligible magnitude. More work is needed to determine the possible importance of the neglected portion of the buoyancy loading (that is, the moment produced by the obliquity of the pressure field).

The large variations of the interference flow field with position in supersonic flow also affect the precision of the loading prediction due to crossflow as well as that due to buoyancy. However, the buoyancy loading is by far the most affected since it depends on pressure differences over small distances. Precision of the cross-flow loading prediction is good if the store is divided into axial sections which are no larger in length than the axial dimensions of the important aircraft details.

METHOD EVALUATION

The ability of the computerized method to predict store interference coefficients was examined by comparing theory to wind tunnel data for the ARM and RAM stores in the influence of the F-4 aircraft. Figure 1 shows the geometry for the aircraft and the ARM and RAM stores. Succeeding figures present interference moment coefficient about the store reference center of gravity versus axial traverse distance of the store forward from the mate ($X = 0$) position.

Pitch and yaw moment comparisons between the prediction method and tunnel data are presented in Figures 2 and 3 for the ARM store under the F-4 aircraft at Mach 1.2. Figures 2a and 2b present pitch and yaw moments, respectively, for the store under the outboard pylon. Figures 3a and 3b present the corresponding data under the inboard pylon. The dashed lines are the results of the computerized predictions, showing the separate contributions of each aircraft component* (wing, nose, inlet, pylons) as well as the total. However, the experimental data does not extend far enough forward of the aircraft to measure the effect of the F-4 nose on the ARM store.

The experimental data of Figures 2 and 3 are the same as that used for data comparisons in Reference 3. The present theoretical prediction method is a much improved version of that used for the predictions of Reference 3. The major improvements in the method are: (1) locations of shock waves are estimated and are used to transform the aircraft into an equivalent geometry before linear theory is used

* Pylon crossflow contributions are small and are omitted from the component predictions for clarity, but are included in the theoretical total.

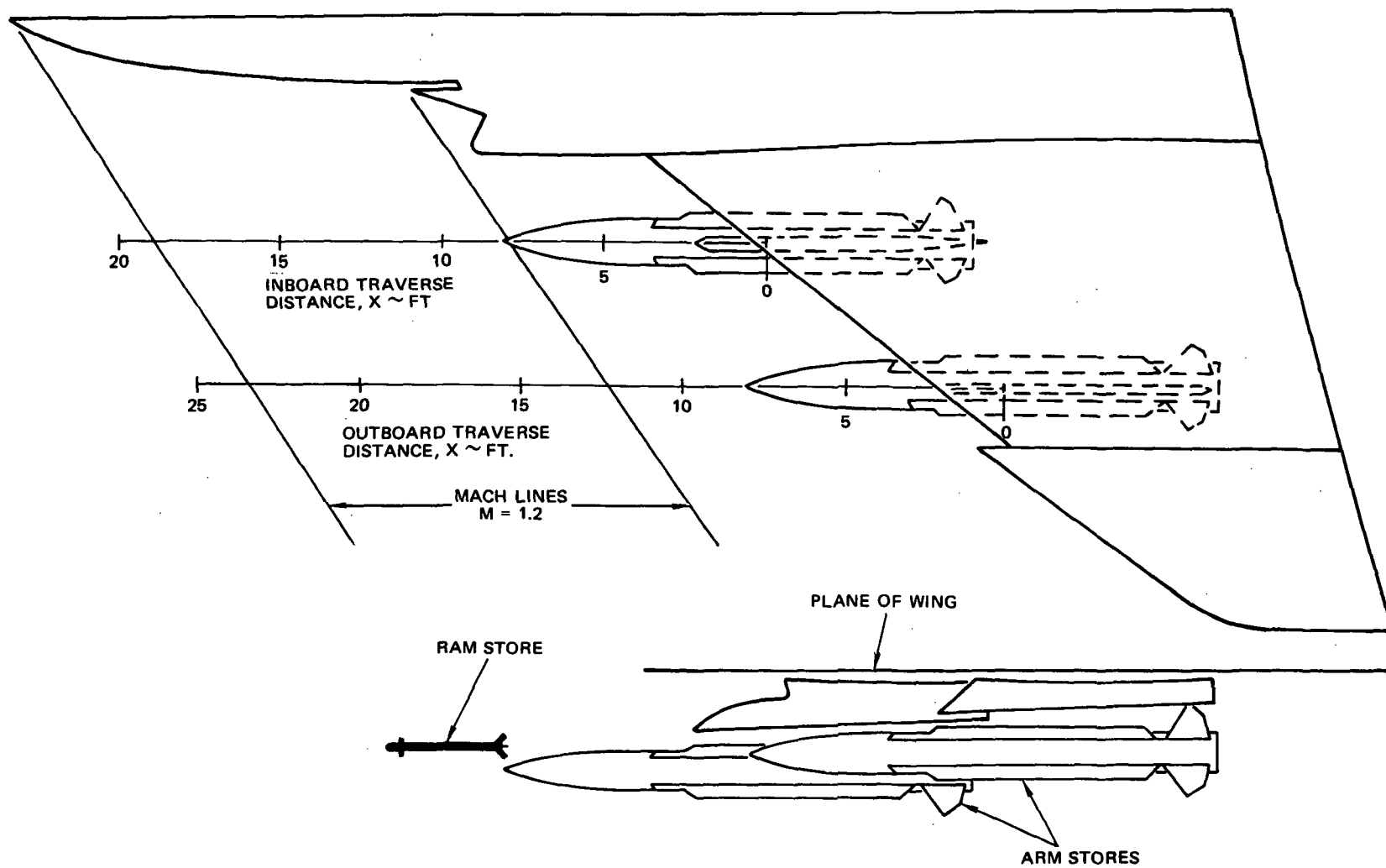
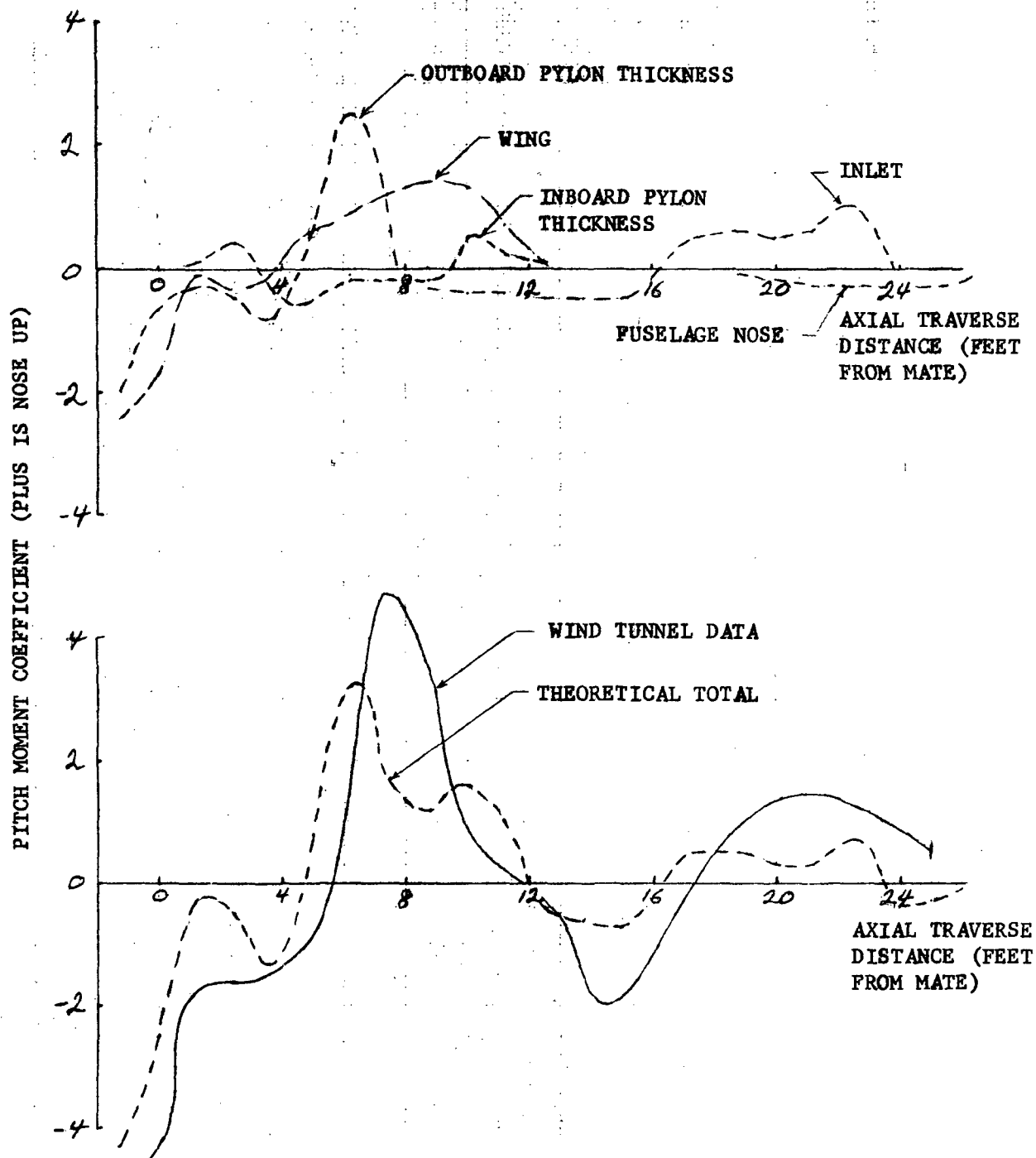


Figure 1. Aircraft and Store Geometry.

FIGURE 2a

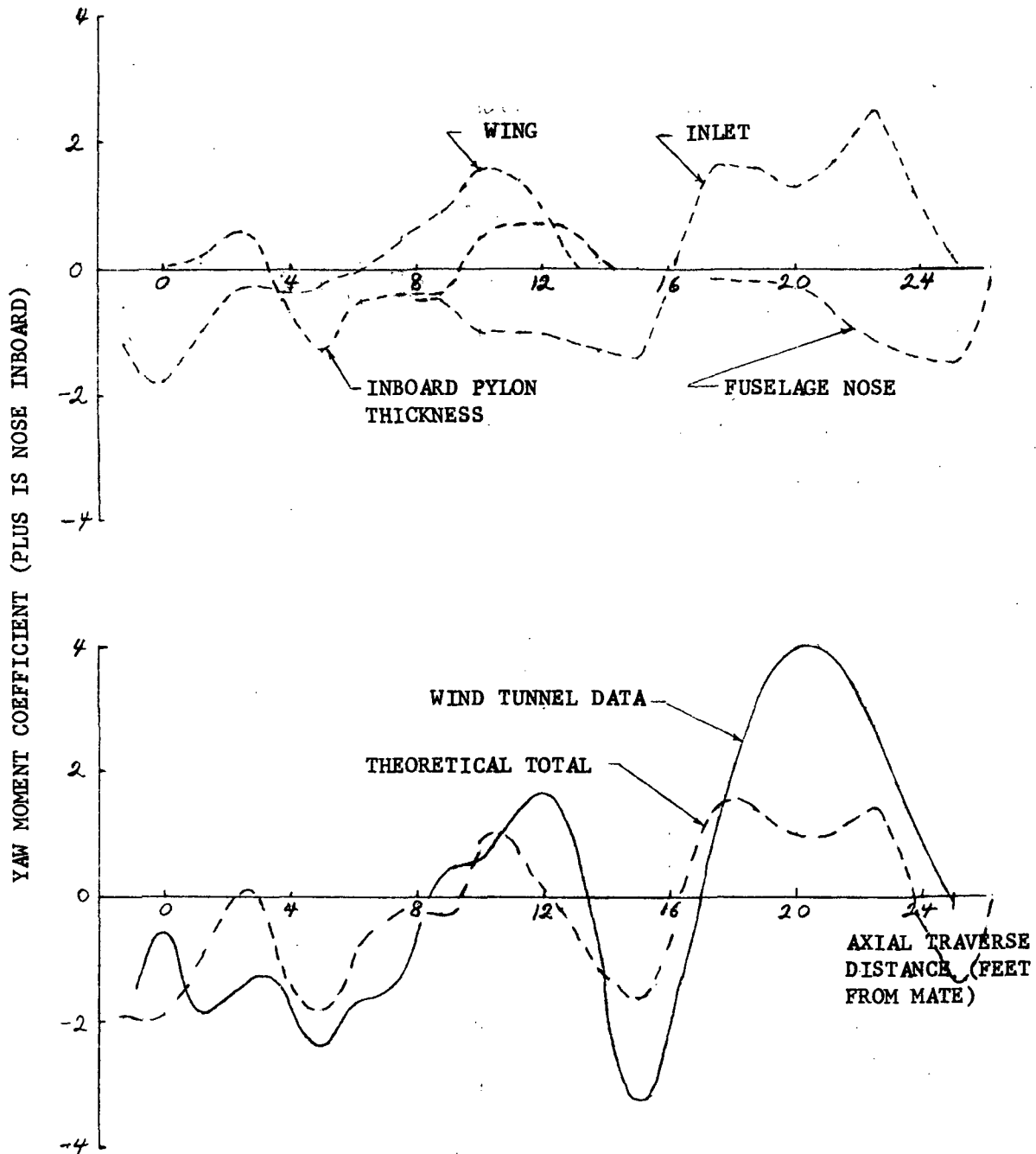


INTERFERENCE COEFFICIENT PREDICTION FOR SEPARATE
AIRCRAFT COMPONENTS AND COMPARISON WITH EXPERIMENT

(Mach 1.2, F-4 aircraft angle of attack = 0,
ARM store incidence = 0)

ARM STORE PITCH MOMENT UNDER F-4 OUTBOARD PYLON

FIGURE 2b.

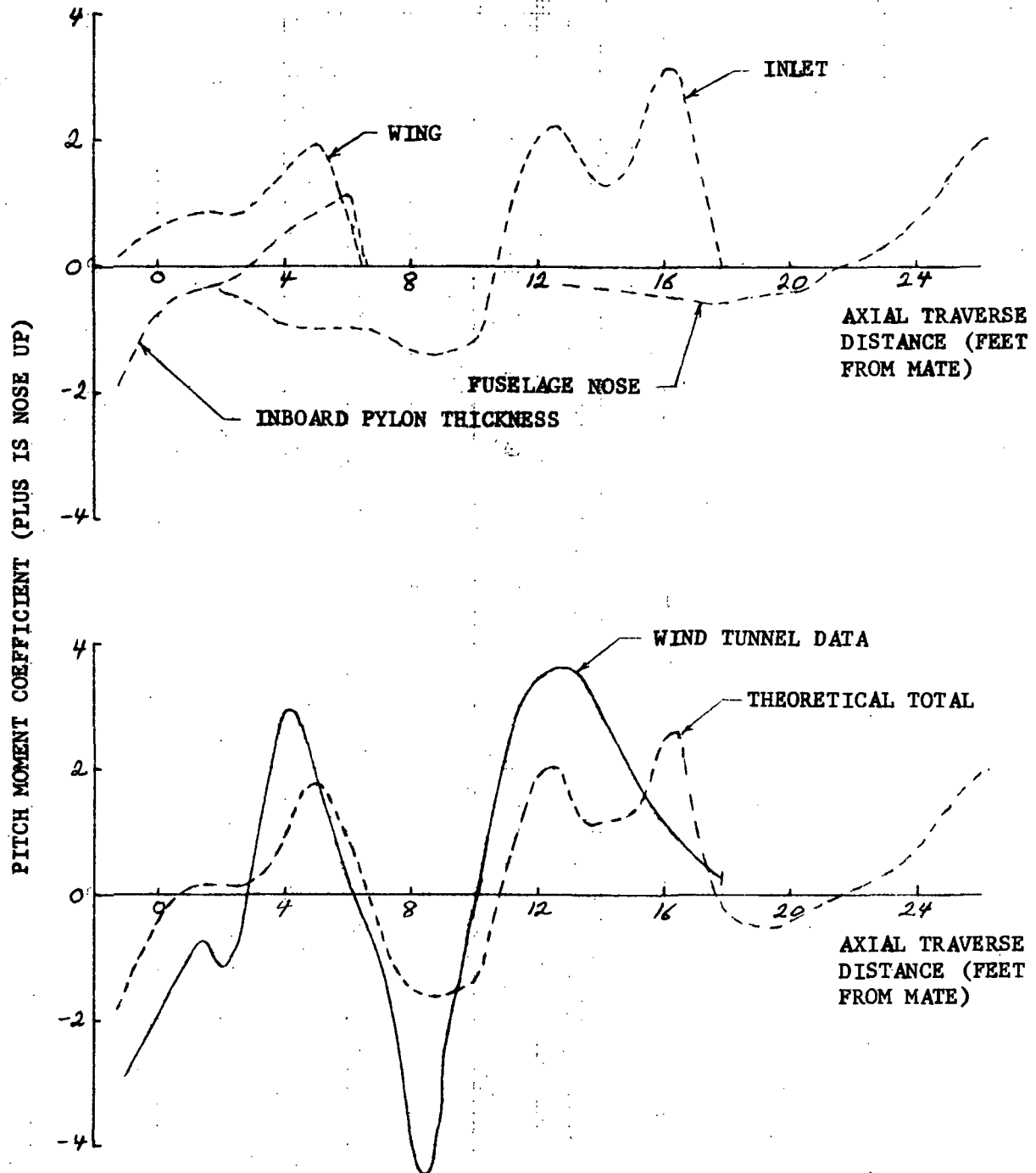


INTERFERENCE COEFFICIENT PREDICTION FOR SEPARATE
AIRCRAFT COMPONENTS AND COMPARISON WITH EXPERIMENT

(Mach 1.2, F-4 aircraft angle of attack = 0, ARM
store incidence = 0)

ARM STORE YAW MOMENT UNDER F-4 OUTBOARD PYLON

FIGURE 3a

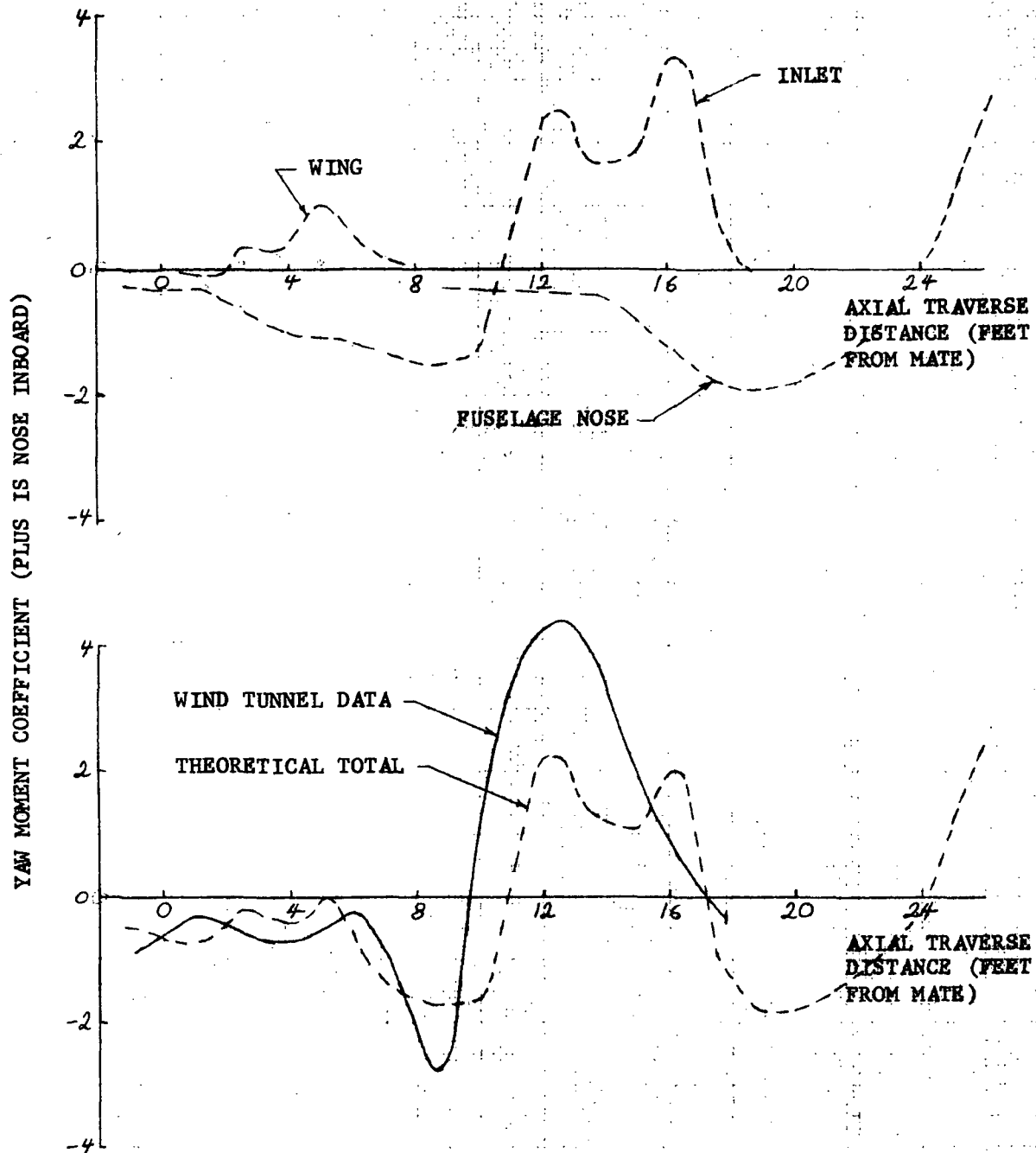


INTERFERENCE COEFFICIENT PREDICTION FOR SEPARATE
AIRCRAFT COMPONENTS AND COMPARISON WITH EXPERIMENT

(Mach 1.2, F-4 aircraft angle of attack = 0, ARM
store incidence = 0)

ARM STORE PITCH MOMENT UNDER F-4 INBOARD PYLON

FIGURE 3B



INTERFERENCE COEFFICIENT PREDICTION FOR SEPARATE
AIRCRAFT COMPONENTS AND COMPARISON WITH EXPERIMENT

(Mach 1.2, F-4 aircraft angle of attack = 0, ARM
store incidence = 0)

ARM STORE YAW MOMENT UNDER F-4 INBOARD PYLON

to predict the flow field; (2) interference flow velocities are computed at the store fin tips as well as at the body surface; (3) store loading due to the axial rate of change of crossflow is included. These theory improvements are responsible for the improved data agreement in Figures 2 and 3 as compared to that of Reference 3.

Comparing theory to experiment in Figures 2 and 3, the axial locations of the data peaks, as well as the zero-crossover points, are well predicted by the theory. This was not true for the data comparisons of Reference 3, indicating that the shock locations have been well predicted by the present theory. However, the magnitudes of the data peaks are consistently underpredicted by a factor of about two. This is true for the peaks produced by both pylon and inlet for both inboard and outboard stores.

The underprediction of the data peaks could be caused by one of two factors; either the disturbance flow field severity is underpredicted, or the method of computing the loading from the interference field tends to underpredict the loading. The linear theory used for flow field prediction is corrected for shock locations but not for their strengths; therefore, an underprediction of the flow field severity is quite possible. Also, it is quite possible that neglecting one part of the store buoyancy prediction (as explained in the previous section) has caused the underprediction. Further studies which compare flow field predictions directly with experiment are needed to better evaluate and improve the method.

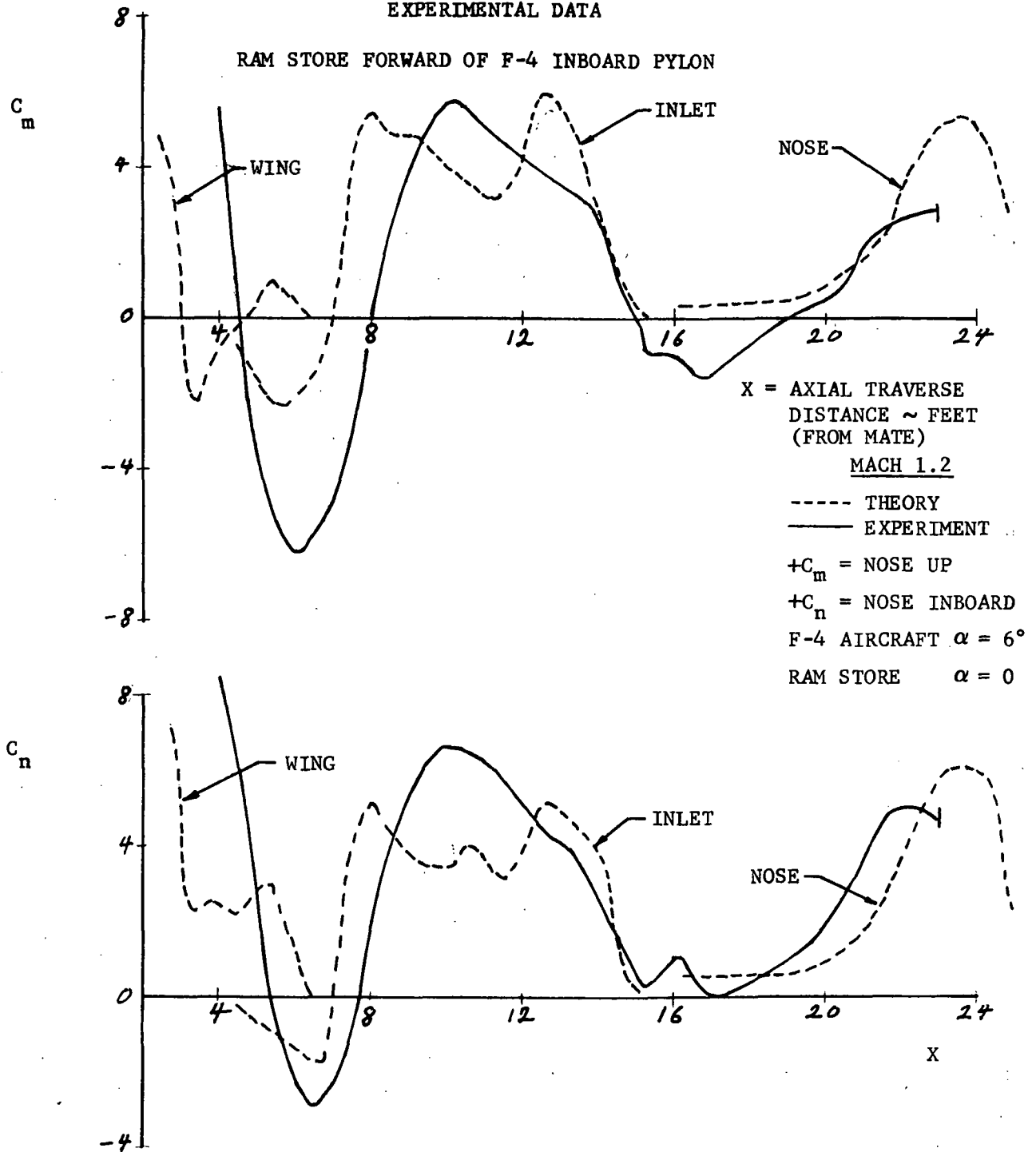
In describing the flow produced by the inlet ramp, some method of accounting for the re-turning of the flow at the inlet cowl must be included. This was done by assuming that no inlet ramp disturbance propagates into the Mach cone of influence of the inlet cowl lip (as was also done in Reference 3). This is, of course, a very simple representation of a complex interaction problem. The negative peaks in the moment data, at $X = 15$ feet on Figures 2 and at $X = 9$ feet on Figures 3, are most affected by the manner in which this inlet cowl interaction is described. This is an area where further work is needed.

Figure 4 is a data comparison for the RAM store under the F-4 aircraft inboard pylon. Since RAM is launched from a tube and leaves the tube ahead of the pylon, it experiences no pylon interference. The primary aircraft interference is produced by the aircraft wing, nose, and inlet, which are shown separately on the figure.

The RAM data agreement of Figure 4 is very good for the fuselage nose interference. The inlet interference prediction is good for the X -location of the zero-crossover points, and the total magnitude of the positive moment disturbance is also well predicted. The negative peak in pitch moment at $X = 6$ feet is underpredicted, perhaps due to inlet cowl interaction as discussed previously for ARM. The wing interference prediction is not well located. This may indicate the extent to which the wing interference does not travel along the Mach line directions used by linear theory. No shock location adjustment is made for the wing interference flow field.

COMPARISON OF THEORETICAL INTERFERENCE
PITCH AND YAW COEFFICIENTS WITH
EXPERIMENTAL DATA

FIGURE 4



METHOD IMPROVEMENT

The data comparisons of the previous section indicate three areas of study where the prediction method could be improved. These are:

- (1) account for the strengths of flow field disturbances behind finite shock waves as they may differ from that given by linear theory;
- (2) account for the store buoyancy loading produced by the fact that the pressure disturbance strikes the store at an oblique angle;
- (3) investigate in more detail the flow field produced by the inlet ramp as affected by interaction with the inlet lip and cowl. Further studies to improve and evaluate the method should also include data comparisons between theory and experiment for the interference flow field as well as for store interference loadings.

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AERODYNAMIC THEORY

The aircraft wing, pylon, and jet inlet ramps are represented by source distributions according to linear theory. These aircraft surfaces are divided into small segments of constant source strength. The source strengths to represent wing and pylon thickness and inlet ramps are simply equal to the local surface slopes. The source strengths to represent wing and pylon crossflow are determined by the "Mach Box" numerical step procedure as explained in section A. Calculation of the flow field due to these source distributions is presented in section B. The aircraft fuselage nose is represented by a source and doublet distribution along the axis of symmetry as explained in section C. Section D explains how the linear theory representation of the aircraft is to be corrected for the effects of shock waves.

A. Wing and Pylon Source Distribution Due to Crossflow

The "Mach Box" step procedure of Reference 4 is used to evaluate the source distribution in the plane of the wing or pylon due to crossflow; wing crossflow is produced by camber, twist, and angle of attack; pylon crossflow is produced by interference with the wing. The method also yields the distribution of lift on the wing. In the Mach Box method, the principle of the diaphragm adjacent to the planform subsonic edges is used to account for flow field communication between upper and lower surfaces (after Evvard). The source strength on the planform itself is

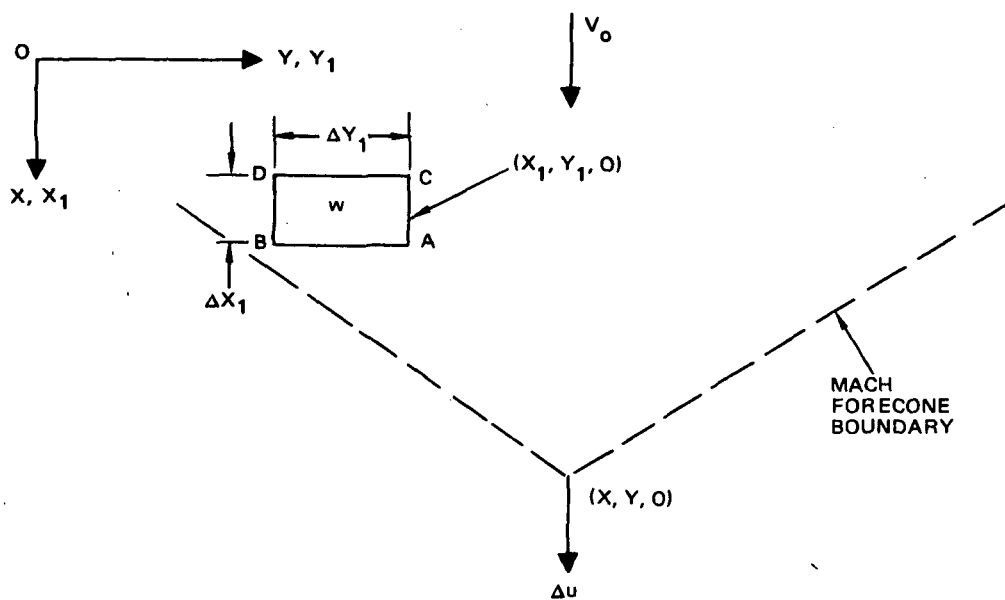
equal to the local surface slope, while the source strength on the diaphragm is calculated from the condition that the diaphragm can support no lift. The mathematical procedure for applying this Mach Box method to the pylon is identical to that for the wing, with some geometric differences. For simplicity, the method will be presented for the wing; geometric considerations needed for adapting the method to the pylon are discussed in APPENDIX B explaining pylon representation for computer program input.

To formulate the Mach Box method, consider a planar wing of arbitrary planform in the $Z_1 = 0$ plane. The velocity potential at any point $(X, Y, 0)$ on the wing is given by

$$\phi = \frac{-1}{\pi} \iint_{\substack{\text{FORE} \\ \text{CONE}}} \frac{w \, dX_1 \, dY_1}{\sqrt{(X-X_1)^2 - \beta^2 (Y-Y_1)^2}} \quad (1)$$

where w is the downwash velocity on the wing or diaphragm at point $(X_1, Y_1, 0)$ which lies inside the Mach forecone of point $(X, Y, 0)$.

Consider a small rectangular segment of constant downwash (w) in the $Z_1 = 0$ plane, as shown in sketch 2. Let $\Delta\phi$ be the velocity potential at point $(X, Y, 0)$ due to this segment and Δu be the axial perturbation velocity.



Sketch 2

$$\Delta u = \frac{\partial}{\partial x} (\Delta \phi) \quad (2)$$

Integrating equation (1) over the segment and differentiating with respect to X gives

$$\Delta u = \frac{w}{\pi \beta} \frac{Y - Y_i}{|Y - Y_i|} \cos^{-1} \left[\frac{|Y - Y_i|}{\frac{1}{\beta} (X - X_i)} \right] \Bigg|_{(A-B)-(C-D)} \quad (3)$$

where the arc cosine function is evaluated at corners A, B, C, and D as indicated.

Zartarian in Reference 4 divides the entire wing plane into these rectangular segments, with w and u considered constant inside each segment. Summing equation (3) for all segments lying inside the Mach forecone of the segment with center at (X, Y, 0), Reference 4 uses the notation

$$u_i = \sum_{\substack{j \text{ IN} \\ \text{FORECONE}}} \frac{w_j}{\beta} R(i, j) \quad (4)$$

where

$$R(i, j) = \frac{1}{\pi} \frac{Y - Y_i}{|Y - Y_i|} \cos^{-1} \left[\frac{|Y_i - Y_j|}{\frac{1}{\beta} (X_i - X_j)} \right] \Bigg|_{(A-B)-(C-D)} \quad (5)$$

The functions R are called Aerodynamic Influence Coefficients (AIC's) and are tabulated in Reference 4 for box segments which have Mach lines parallel to the diagonals; that is, $\Delta X_1 = \beta \Delta Y$, in Sketch 2. Such box segments are called "Mach Boxes," and their use allows tabulation of the R functions independent of Mach number and without consideration of partial boxes being inside the Mach forecone. The use of tabulated AIC values saves much

computation labor in determining wing diaphragm source strength and wing lift distributions. Figure 5 shows values of these AIC's versus position of the Mach boxes relative to the box being influenced.

Equation (4) is used to evaluate the source strength on the diaphragm, using the physical condition that the pressure coefficient on the diaphragm is zero:

$$C_p = \frac{-2u}{V_o} \quad (6)$$

Therefore $u = 0$ (7)

Equation (4) is also used to evaluate the wing lift distribution by evaluating u for each box, u being related to pressure coefficient by equation (6) above. As equation (4) is applied to each Mach box in the correct order, only one unknown appears in equation (4) each time (u or w), and that unknown is solved for explicitly. Figure 6 shows typical wing planforms with diaphragms and with Mach boxes numbered in the order in which they are considered.

The above steady state "AIC" method for obtaining wing source distribution and lift distribution has been coded in FORTRAN for this study. Checkout calculations of wing lift and lift distribution have been made which show the method to agree with other linear theory methods to about 4 percent, depending on the number of Mach boxes used.

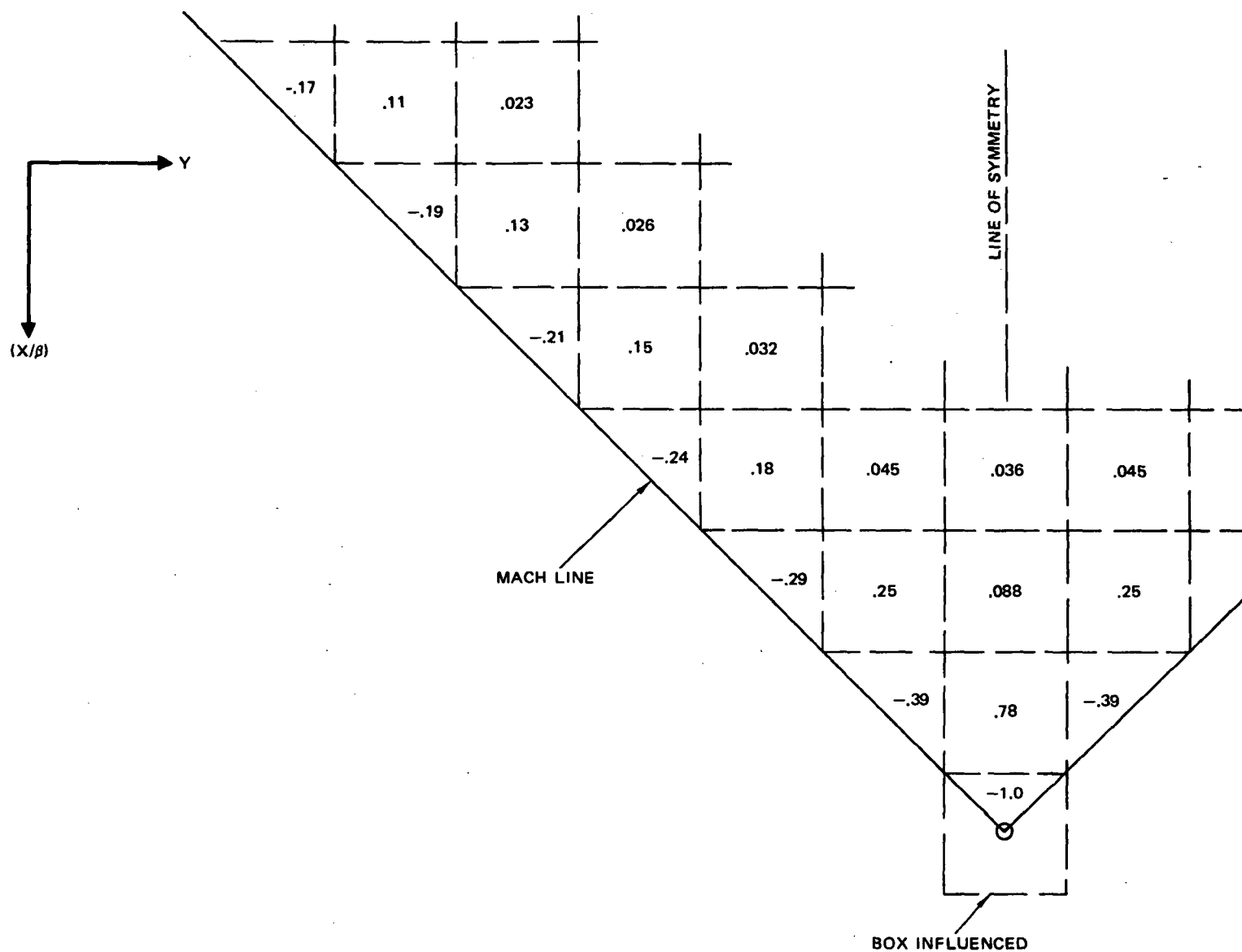
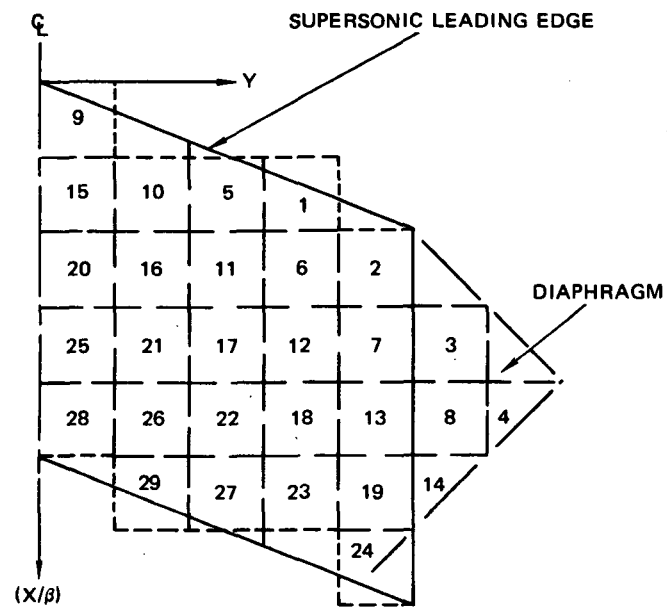


Figure 5. Values of Aerodynamic Influence Coefficients (AIC's) Versus Box Location in Mach Forecone.

--- MACH LINES
 — ACTUAL WING
 - - - MACH BOX WING



V_0

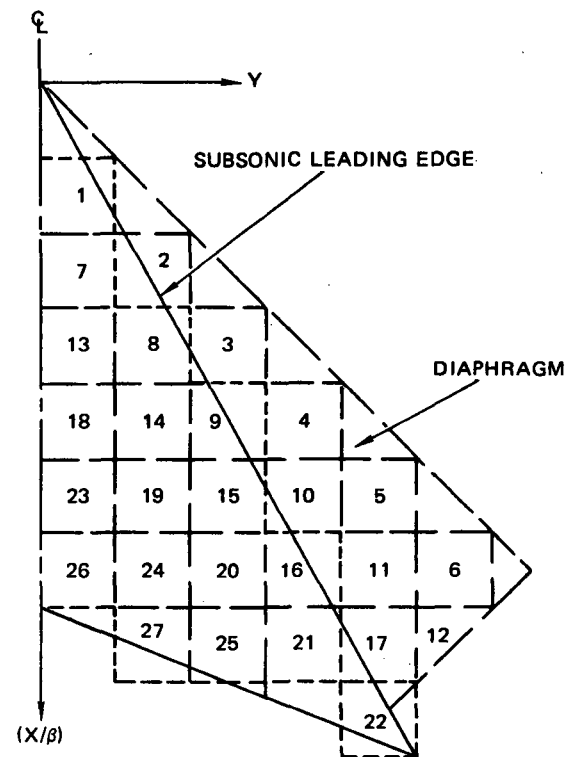
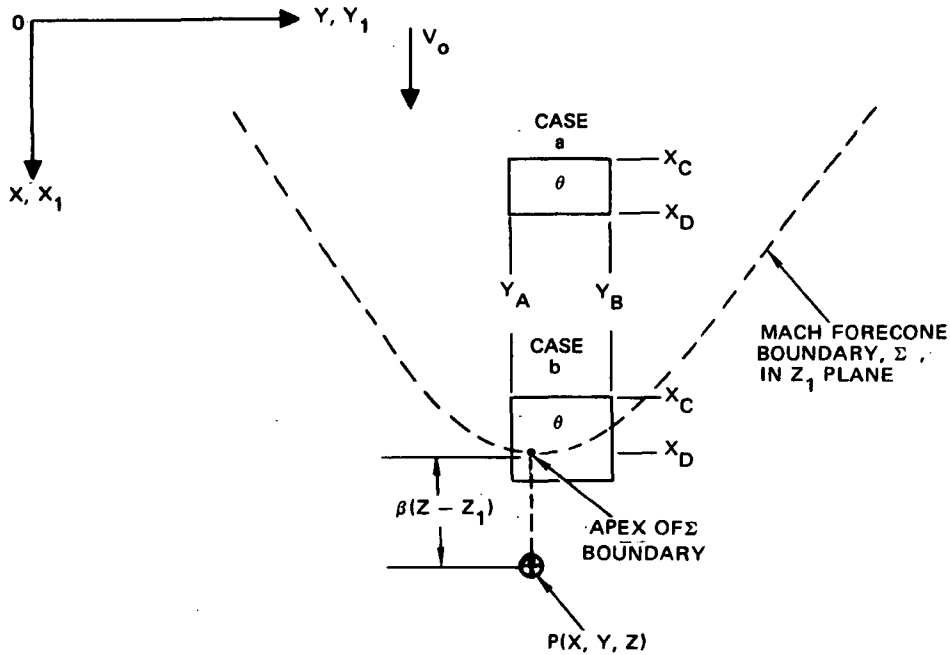


Figure 6. Order of Computing Mach Box Properties
 (u or w) on Wing and Diaphragm.

B. Flow Field Due to Each Source Element

Consider a small rectangular segment of source distribution in the plane of $Z_1 = \text{constant}$ and inside the Mach forecone (Σ) of point P, as shown in Sketch 3.



Sketch 3

Two cases are considered: (a) segment lies entirely inside Σ ; (b) segment lies on Σ boundary such that X_D is at the boundary apex.

Let the segment have constant slope θ related to downwash by

$$\theta = w/V_0 \quad (8)$$

The velocity potential at point P is given by

$$\frac{\phi}{V_0} = \frac{-\theta}{\pi\beta} \iint_{\Sigma} \frac{dx_i dy_i}{h} \quad (9)$$

where $h = \sqrt{a^2 - (Y - Y_i)^2} \quad (10a)$

$$a = \sqrt{\left(\frac{X - X_i}{\beta}\right)^2 - (Z - Z_i)^2} \quad (10b)$$

Equation (9) is integrated first in Y_i , giving

$$\frac{\phi}{V_0} = \frac{\theta}{\pi\beta} \int_{X_c}^{X_D} F dx_i \quad (11)$$

where

$$\begin{aligned} F &= - \int_{Y_A}^{Y_B} \frac{dY_i}{h} \\ &= S/N - 1 \left(\frac{Y - Y_i}{a} \right) \Big|_{Y_i=Y_A}^{Y_i=Y_B} \end{aligned} \quad (12)$$

Integration limits are shown in Sketch 3.

Equation (11) is differentiated with respect to X , Y , and Z to obtain the perturbation velocities u , v , and w , respectively. Using Leibnitz' rule for the derivative of an integral with variable limits, and using the variable s to represent either X , Y , or Z , equation (11) gives

$$\frac{1}{V_0} \frac{\partial \phi}{\partial s} = \frac{\theta}{\pi\beta} \int_{X_c}^{X_D} \frac{\partial F}{\partial s} dx_i + \frac{\theta}{\pi\beta} F \Big|_{X_i=X_D} \frac{\partial X_D}{\partial s} \quad (13)$$

For case (a) of Sketch 3, X_D is not a function of s , and the last term is zero. For case (b) of Sketch 3, X_D is at the apex of the Σ boundary and is given by

$$X_D = X - \beta (Z - Z_1) \quad (14)$$

Evaluating F at X_D from equation (12) gives

$$\begin{aligned} F|_{X=X_D} &= \sin^{-1}(-1) - \sin^{-1}(1) \\ &= -\pi \end{aligned} \quad (15)$$

For case (b) then, equation (13) becomes

$$\begin{aligned} \frac{1}{V_0} \frac{\partial \phi}{\partial s} &= \frac{\theta}{\pi \beta} \int_{X_c}^{X_D} \frac{\partial F}{\partial s} dX, \\ &\quad - \frac{\theta}{\beta} \frac{\partial}{\partial s} [X - \beta |Z - Z_1|] \end{aligned} \quad (16)$$

Denote the integral in equation (16) as G_s .

$$G_s = \int_{X_c}^{X_D} \frac{\partial F}{\partial s} dX, \quad (17)$$

and define

$$\begin{aligned} \Delta X &= X - X_1 \\ \Delta Y &= Y - Y_1 \\ \Delta Z &= Z - Z_1 \end{aligned} \quad (18)$$

Using equation (12) for F, the G functions are

$$G_x = \int_{x_c}^{x_D} \left[\frac{-\Delta Y \Delta X}{\beta a^2 \sqrt{a^2 - \Delta Y^2}} \right]_{Y_1=Y_A}^{Y_1=Y_B} dx, \quad (19a)$$

$$G_Y = \int_{x_c}^{x_D} \left[\frac{1}{\sqrt{a^2 - \Delta Y^2}} \right]_{Y_1=Y_A}^{Y_1=Y_B} dx, \quad (19b)$$

$$G_Z = \int_{x_c}^{x_D} \left[\frac{\Delta Y \Delta Z}{a^2 \sqrt{a^2 - \Delta Y^2}} \right]_{Y_1=Y_A}^{Y_1=Y_B} dx, \quad (19c)$$

Integrating equations (19) gives

$$G_x = \left\{ \frac{\Delta Y}{|\Delta Y|} \cos^{-1} \left[\frac{|\Delta Y|}{a} \right] \right\} \bigg|_{Y_1=Y_A}^{Y_1=Y_B} \bigg|_{x_1=x_c}^{x_1=x_D} \quad (20a)$$

$$G_Y = \left\{ -\beta \cosh^{-1} \left[\frac{\Delta X}{\beta \sqrt{\Delta Y^2 + \Delta Z^2}} \right] \right\} \bigg|_{Y_1=Y_A}^{Y_1=Y_B} \bigg|_{x_1=x_c}^{x_1=x_D} \quad (20b)$$

$$G_Z = \left\{ \frac{-\beta}{2} \frac{|\Delta Y \Delta Z|}{\Delta Y \Delta Z} \right. \\ \left. \times \cos^{-1} \left[\frac{2 \Delta Y^2 \left(\frac{\Delta X}{\beta} \right)^2}{(\Delta Y^2 + \Delta Z^2) \left[\left(\frac{\Delta X}{\beta} \right)^2 - \Delta Z^2 \right]} \right] \right\} \bigg|_{Y_1=Y_A}^{Y_1=Y_B} \bigg|_{x_1=x_c}^{x_1=x_D} \quad (20c)$$

For case (a) of Sketch 3, the perturbation velocities are given by use of the above G functions in

$$\frac{1}{V_0} \frac{\partial \phi}{\partial s} = \frac{\theta}{\pi \beta} G_s \quad (21a)$$

and for case (b)

$$\frac{1}{V_0} \frac{\partial \phi}{\partial s} = \frac{\theta}{\pi \beta} G_s - \frac{\theta}{\beta} \frac{\partial}{\partial s} (X - \beta |\Delta Z|) \quad (21b)$$

where $s = X, Y, \text{ or } Z$

$$\frac{\partial \phi}{\partial s} = u, v, \text{ or } w, \text{ respectively}$$

The velocity field due to a planar source distribution of variable strength $\theta (X_1, Y_1)$ is found by dividing the surface into small rectangular segments, applying equations (21) to each segment by assuming θ constant inside each segment, and summing over all segments. For cases where the planar surface does not lie in a plane of constant Z_1 , a rotation of Y-Z coordinates again allows the use of equations (21), followed by a corresponding resolution of the v-w velocity vectors back into the Y-Z directions.

If in Sketch 3 the source segment fills the Σ boundary as a two-dimensional unswept surface, the G functions of equations (20) are zero, and equation (21b) reduces to the two dimensional values.

$$\frac{u}{V_o} = \frac{-\theta}{\beta}; \quad \frac{v}{V_o} = \text{zero}; \quad \frac{w}{V_o} = \theta \quad (22)$$

Also, note that G_x evaluated at $\Delta Z = 0$ may be used in equation (21a) to calculate the AIC values of Figure 5.

Equations (20) for calculating perturbation velocities in supersonic flow have been coded in subroutine "DIFIN" of the prediction program. They are used by the program to compute the flow field produced by the aircraft wing thickness and crossflow, and by the pylon crossflow. The source elements represented by these equations are rectangular and are therefore not swept. Flow equations for source elements with sweep are given by Woodward, Reference 12. These equations have also been programmed for the present prediction method, and are contained in subroutine "SWP." They are used by the program to compute the flow field produced by pylon thickness and by the inlet ramps. Sweep values may be either greater or less than the sweep of the Mach lines. The advantage to using swept elements is that fewer elements can be used to accurately represent a swept surface. The equations of Reference 12 reduce to Equations (20) for zero sweep.

C. Flow Field Due to Fuselage Nose

The fuselage nose is considered as a pointed body of revolution. The flow field due to thickness envelope is described by placing a line-source distribution of variable strength on the axis of symmetry. For the special case of a cone, Reference 5 gives the flow field as

$$\frac{u}{V_o} = -g \cosh^{-1} \left[\frac{X}{Br} \right] \quad (23a)$$

$$\frac{v_r}{V_o} = g \beta \sqrt{\left(\frac{x}{\beta r}\right)^2 - 1} \quad (23b)$$

where X = distance behind apex

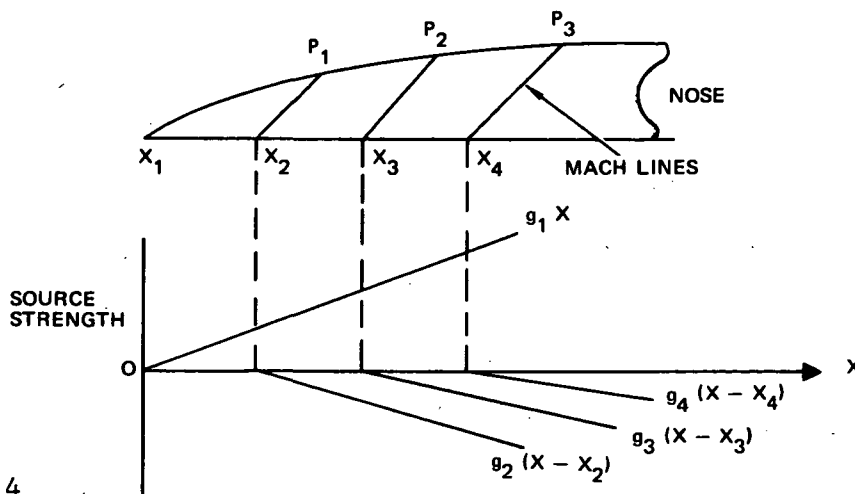
r = distance from axis

and the source strength increases linearly with X , having a slope of g .

The constant g is determined by the boundary condition

$$\frac{v_r}{V_o + u} = \frac{dR}{dx} = \text{cone slope} \quad (24)$$

Flow around a nose of arbitrary shape can be obtained by a superposition of the above conical solutions according to a numerical step procedure of von Karman and Moore (as contained in Reference 5, page 232). In this method, the axial source distribution is made up of several cone contributions as shown in Sketch 4, with cone apex locations spaced at intervals along the axis.



Choose body surface points P_i on the Mach cone from the apex X_{i+1} of each linear source distribution as shown in the sketch. Let u_{ij} , v_{rij} be the velocities at surface point i due to conical source contribution j . Then due to the rule of forbidden signals in supersonic flow,

$$u_i = \sum_{j=1}^i u_{ij} \quad (25a)$$

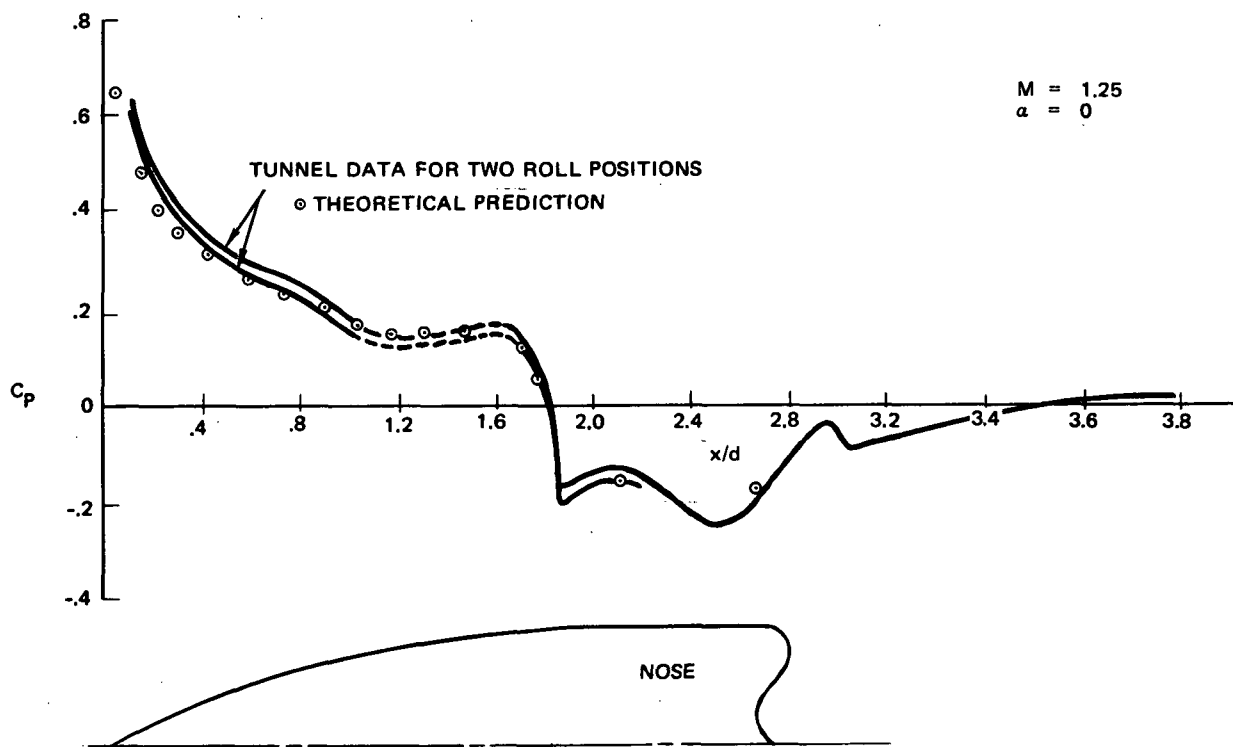
$$v_{ri} = \sum_{j=1}^i v_{rij} \quad (25b)$$

The source strengths g_i are found by solving the boundary condition, equation (24), at each surface point i in turn from front to rear. Using equations 25 and 23 in 24 and solving for g_i gives

$$g_i = \frac{\frac{dR}{dx}|_i \left\{ v_0 + \sum_{j=1}^{i-1} u_{ij} \right\} - \sum_{j=1}^{i-1} v_{rij}}{\left(\frac{v_{rii}}{g_i} \right) - \frac{dR}{dx}|_i \left(\frac{u_{iii}}{g_i} \right)} \quad (26)$$

Sample computerized calculations using the above method are shown in Figure 7 for the pressure distribution over an ogive nose at Mach 1.25. Pressure coefficient is related to surface velocity by

Figure 7. Comparison of Theory With Experiment for Pressure Coefficient on Axisymmetric Nose.



$$C_p = \frac{-2u}{V_o} - \left(\frac{v_r}{V_o} \right)^2 \quad (27)$$

The method is seen to give good surface pressure results.

To compute the nose velocity field due to angle of attack, an analysis similar to the one presented above is made using line doublets of constant slope in place of the line sources. Sketch 4 is applied to this doublet distribution as well if the source strengths g_i are replaced by the corresponding doublet strengths, which we shall call c_i . Reference 6 gives the flow field due to a single such line doublet as that for a cone:

$$\frac{u}{V_o \alpha} = c \beta \cos \theta \sqrt{\left(\frac{X}{\beta r} \right)^2 - 1} \quad (28a)$$

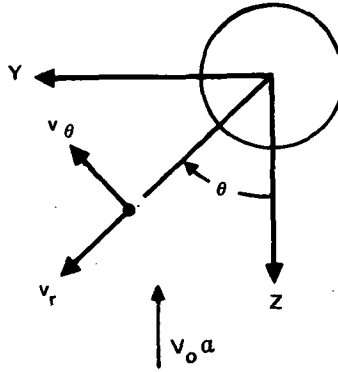
$$\frac{v_r}{V_o \alpha} = \frac{-c \beta \cos \theta}{2} \left\{ \beta \cosh^{-1} \left(\frac{X}{\beta r} \right) + \frac{X}{r} \sqrt{\left(\frac{X}{\beta r} \right)^2 - 1} \right\} \quad (28b)$$

$$\frac{v_\theta}{V_o \alpha} = \frac{c \beta \sin \theta}{2} \left\{ \beta \cosh^{-1} \left(\frac{X}{\beta r} \right) - \frac{X}{r} \sqrt{\left(\frac{X}{\beta r} \right)^2 - 1} \right\} \quad (28c)$$

where X is again the axial distance behind the cone apex, and the boundary condition to be satisfied on the body is

$$\left(\frac{v_r}{V_o \alpha} \right) - \cos \theta = \left(\frac{u}{V_o \alpha} \right) \frac{dR}{dX} \quad (29)$$

The sign convention used for α and θ is shown in Sketch 5.



Sketch 5

Since v_θ does not appear in equation (29), the $\cos \theta$ may be factored out, giving

$$\frac{v_r}{V_o \alpha \cos \theta} - 1 = \frac{u}{V_o \alpha \cos \theta} \frac{dR}{dX} \quad (30)$$

For several such line doublets, equations (28) are written in the form of equations (25). These may then be substituted into equation (30) and solved for each doublet strength c_i in order of increasing i . This gives:

$$c_i = \frac{1 + \frac{dR}{dX} \Big|_i \sum_{j=1}^{i-1} \frac{u_{1j}}{V_o \alpha} - \sum_{j=1}^{i-1} \frac{v_{r1j}}{V_o \alpha}}{\left(\frac{v_{r1i}}{V_o \alpha c_i} \right) - \frac{dR}{dX} \Big|_i \left(\frac{u_{1i}}{V_o \alpha c_i} \right)} \quad (31)$$

The resulting c_1 values are then used in equations (28) to compute the velocity field produced by each line doublet.

It should be noted that the method cannot be applied to cases where the Mach angle from the nose tip lies inside the body. Thus there is an upper bound on Mach number and nose bluntness for applying the method.

D. Shock Wave Effects

The surfaces of jet aircraft produce shock waves in supersonic flow. These shock waves are curved in the three-dimensional flow field and approach the slope of the Mach wave in the far field. They may also be detached forward of the disturbance-producing component at low supersonic Mach numbers, and subsonic flow may exist behind them. Important shock effects are produced by the aircraft fuselage nose, the inlet ramps, and the leading edges of the pylons.

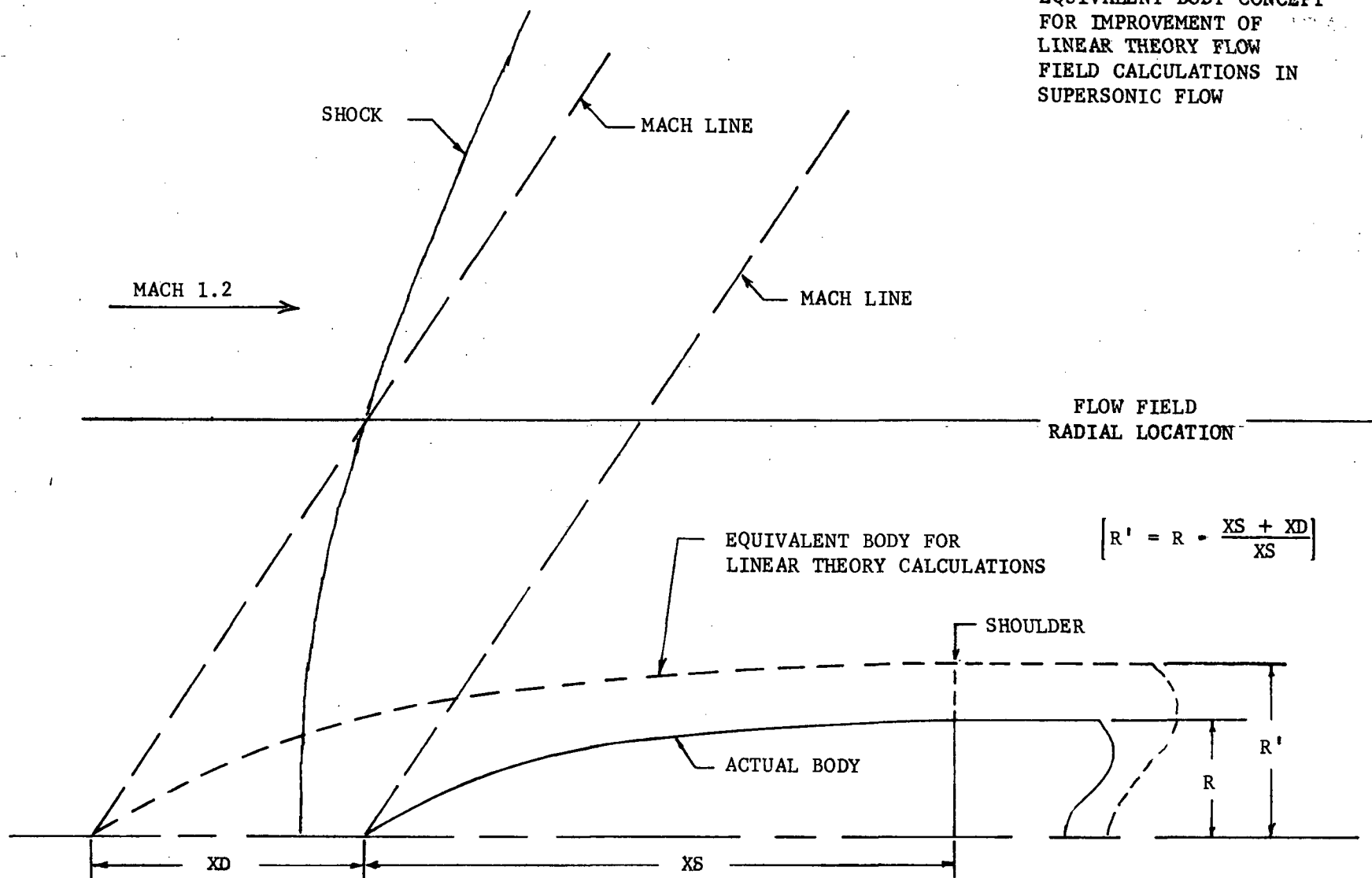
The use of linear theory for predicting supersonic flow fields is severely restricted in accuracy if no correction is made for these shock waves. This problem was encountered in Reference 3. This occurs because linear theory predicts that all disturbances travel along Mach waves; shock waves lie ahead of Mach waves and can produce a considerable forward "shift" of the disturbance flow field ahead of the linear theory solution. Also, the severity of the disturbance tends to be greater than the linear theory prediction.

A method is introduced here for accounting for the effects of shock waves on the flow field. The method applies the linear theory solution to a body which has been transformed from the original body. The method is illustrated in Figure 8, which shows a body of revolution at Mach 1.2. In the transformation, the actual body is "stretched" forward from the shoulder so that the Mach wave from the apex of the equivalent body intersects the shock wave at the location where the flow field is to be calculated. The transformed (equivalent) body is geometrically similar to the actual body, and the shoulder location is preserved. The linear theory solution for the equivalent body has these desirable features: (1) the forward Mach wave gives the correct location for the initial disturbance; (2) the expansion Mach wave from the shoulder is in the correct location; (3) the total magnitude of the disturbance, being proportional to body radius, is increased by a factor that is proportional to the distance of the shock wave ahead of the Mach wave. As the actual body is made more slender, the shock wave approaches the Mach wave and the transformed body approaches the actual body. Since the linear theory solution for a slender body is accurate, the method gives the proper solution for slender bodies as a limit case.

This equivalent body concept should be looked upon as an engineering approximation to solve a very difficult flow problem. No rigorous solution exists for the far flow field of bodies with curved, possibly

FIGURE 8

EQUIVALENT BODY CONCEPT
FOR IMPROVEMENT OF
LINEAR THEORY FLOW
FIELD CALCULATIONS IN
SUPersonic FLOW



detached, shock waves. The method of characteristics, as well as characteristic line modifications to linear theory, cannot be applied where subsonic flow exists extensively behind the shock. The accuracy of the present method and the limits of its application have not been well examined. It can be reasoned that the method should break down as the radius of the transformed body nears the flow field location to be calculated. Justification for the method as applied here is that it does give improved accuracy over linear theory. An area of further study on the method would be to use the shock inclination angle to predict the flow properties immediately behind the shock, and use this information in some simple manner to improve the accuracy of the linear theory prediction for pressure rise and flow angularity. This has not been considered here.

To apply the equivalent body method, the location of the shock wave must be known. A separate computer program has been written for this purpose. This is Program SHOCK and is explained in the appendix. It supplies the value XD noted on Figure 8. This value completely determines the equivalent body.

The equivalent body concept is applied to planar surfaces in the same manner as for axisymmetric bodies; that is, the leading edge of the surface is displaced forward while the slope and shoulder location are retained. In this way the method is applied to the aircraft inlet, pylon, or any compressive surface. Figure 8 may be considered as an end view for planar surfaces. The computer program STORLD transforms

the aircraft nose to the equivalent nose using XD as input. The other aircraft surfaces must be transformed by the user prior to input into the program. This is because the planar aircraft surfaces are inputted as elementary strips with no implied shoulder location or sense of axial ordering.

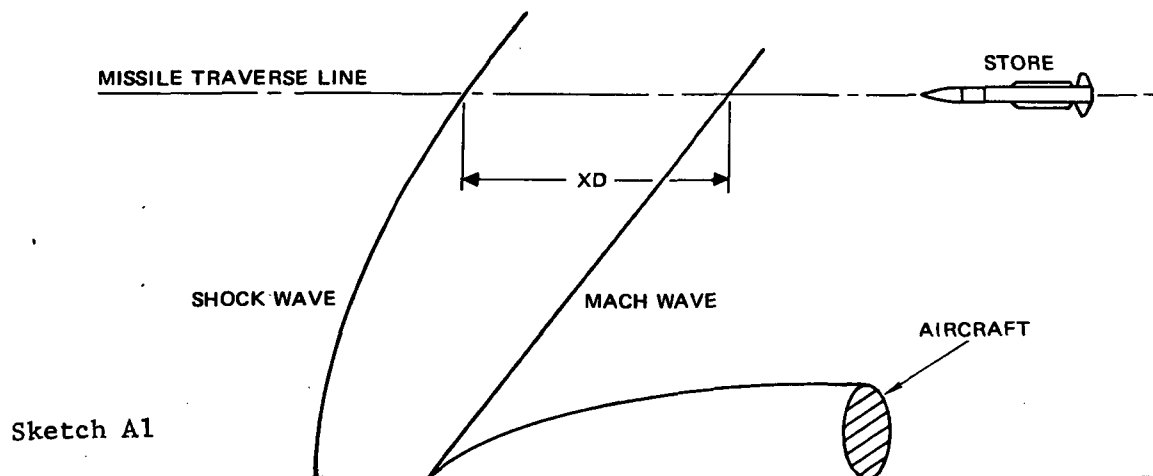
APPENDIX A
INTERSECTION OF MISSILE TRAVERSE LINE WITH
AIRCRAFT SHOCK WAVES

CONTENTS

	Page
GENERAL DISCUSSION OF SHOCK GEOMETRY	A-3
ATTACHED SHOCKS	A-4
DETACHED SHOCKS	A-7

APPENDIX A
INTERSECTION OF MISSILE TRAVERSE LINE WITH
AIRCRAFT SHOCK WAVES

To improve the supersonic linear theory predictions, shock waves are used as the mechanism to propagate the aircraft compressive disturbances, rather than the Mach waves used in linear theory. The purpose of this appendix is to describe the methods used in computer program "SHOCK" to predict the axial distance between the Mach wave and the shock wave at the store traverse line, marked "XD" on the sketch below.

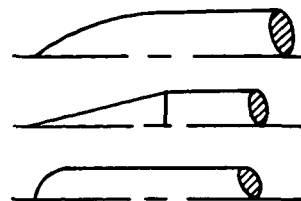


Two distinct classes of shocks are examined: the attached shock and the detached shock. The methods used to predict the shape of these shocks are those developed in References 8 and 9. The shock generating aircraft components to be analyzed by the computer program are the nose, the inlet ramp or centerbody, and the pylons. To simplify the problem these components are represented by one of three geometric shapes:

1) An ogive forebody followed by a cylinder

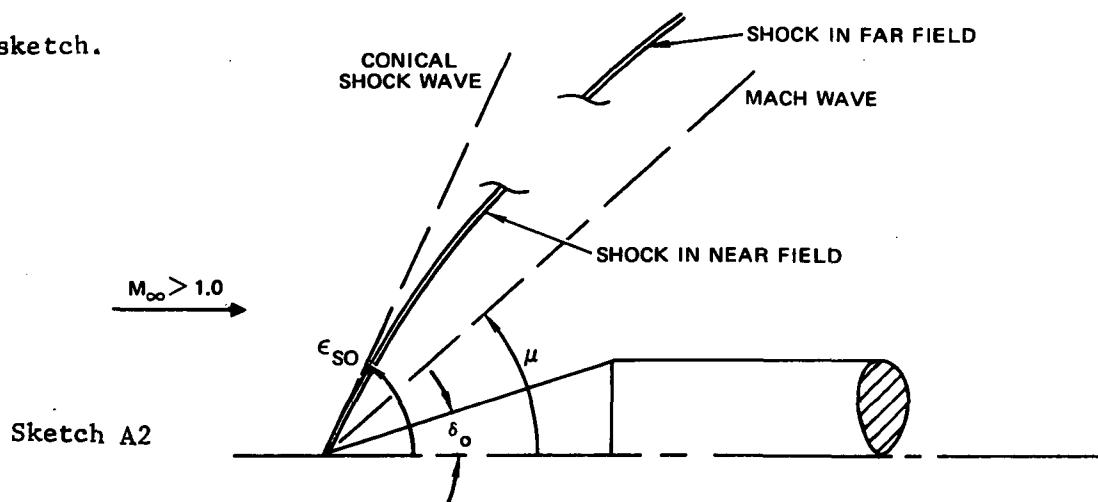
2) A cone-cylinder

3) A hemisphere-cylinder



GENERAL DISCUSSION OF SHOCK GEOMETRY

To examine the general features of a shock wave about a body in supersonic flow, consider the cone-cylinder shown in the following sketch.



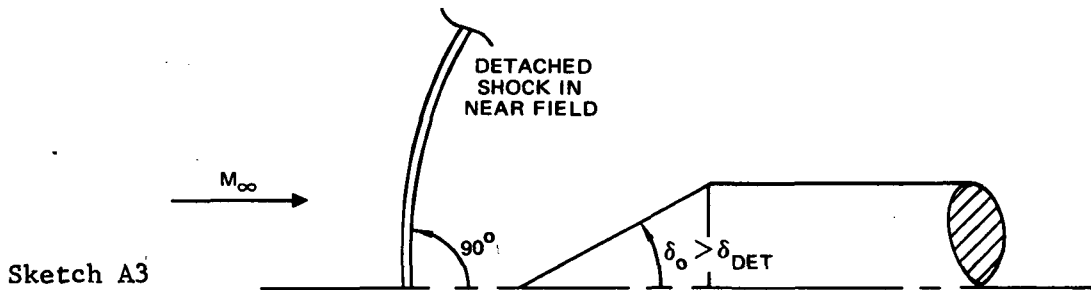
If δ_0 is small enough for the supersonic flow considered, the shock wave will be attached as shown above. The shock wave about this cone-cylinder has a curved outline and lies between the conical shock of an infinite conical body and the Mach cone of an infinitesimal disturbance. At the apex of the body the shock wave is tangent to the conical shock and then curves away from this cone until its slope approaches that of the Mach cone in the far field. If δ_0 were increased continuously, a maximum body cone angle of δ_{DET}^* would be attained that would just support an attached shock. A slight increase in δ_0 beyond this value of δ_{DET} causes the shock wave to move forward

*This cone angle can be found for a given freestream Mach number in

of the body, i.e., a detached shock exists. This shock wave would have the general shape shown below



DETACHED SHOCK IN FAR FIELD



The detached shock would have a shock angle of 90° at the body centerline and would have a slope approaching that of the Mach wave in the far field.

The above discussion for cone-cylinder shocks also applies to ogives. The pertinent angle δ_0 for the ogive nose that determines whether the shock is attached or detached is measured at the nose apex. The hemisphere-cylinder body has a blunt nose with $\delta_0 = 90^\circ$. Therefore, the shock for this body is always the detached type.

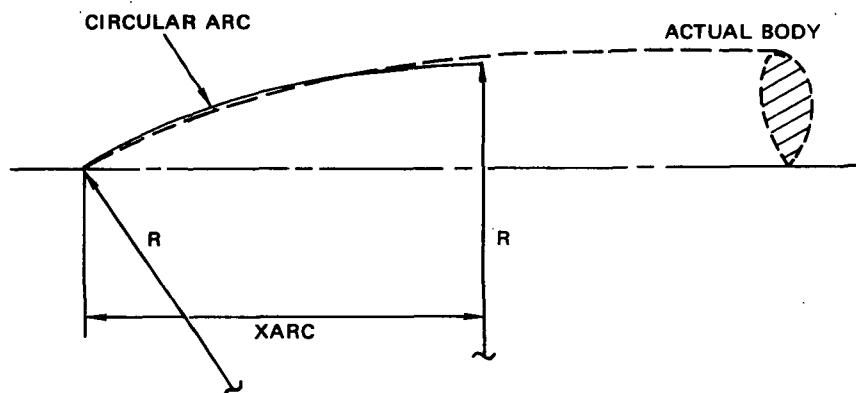
ATTACHED SHOCKS

In Reference 8 a method is given to predict the attached shock geometry about axisymmetric bodies having smoothly contoured noses. The shock equation, equation (A-1), is developed in dimensionless coordinates x_1 and y_1 . It is a curve-fit equation such that the

slope of the shock at the apex has the conical shock slope, and the shock slope in the far field approaches that of the conical Mach wave.

$$y_1 = (\tan \epsilon_{so}) [\ln (1 + x_1)] + (\tan \mu) [x_1 - \ln (1 + x_1)] \quad (A-1)$$

The authors of Reference 8 note that the forward portion of the nose essentially determines the shape of the shock. They suggest representing the forward portion of the nose by a circular arc nose that is tangent to the given nose at its apex and approximates the actual nose shape over this forward nose section as shown below.



Sketch A4

A simple scale relationship between physical shock coordinates and dimensionless shock coordinates is suggested in Reference 8;

$$\frac{x}{XARC} = K x_1 \quad (A-2)$$

$$\frac{y}{XARC} = K y_1 \quad (A-3)$$

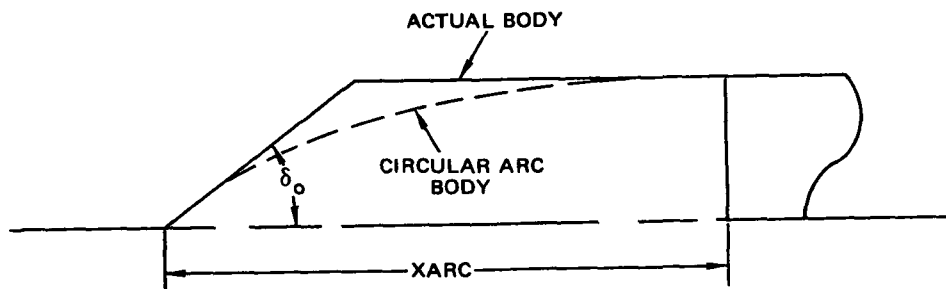
where (x, y) is the physical location of the shock in length units.

Tabular values of K were developed in Reference 8 by fitting the shock shape defined by equations (A-1), (A-2), and (A-3) to the shock calculated from a method of characteristics solution for circular arc bodies for various combinations of freestream Mach number and δ_0 . To apply the method to an arbitrary body, it is first approximated as a circular arc body. The shock shape is then specified by finding the appropriate K value in the table of Reference 8 as a function of Mach number and δ_0 , and using this K value in the following equation, obtained by combining equations (A-1), (A-2), and (A-3).

$$\frac{y}{K(XARC)} = (\tan \epsilon_{so}) \left[\ln \left(1 + \frac{x}{K(XARC)} \right) \right] + (\tan \mu) \left[\frac{x}{K(XARC)} - \ln \left(1 + \frac{x}{K(XARC)} \right) \right] \quad (A-4)$$

For the low supersonic Mach number regime, where subsonic flow exists behind the shock, K values are not available from Reference 8 because the method of characteristics does not apply. It was therefore necessary to extrapolate the K values of Reference 8 into this lower Mach number regime for the present application. Fortunately, the shock location was found not to be sensitive to the K values chosen in the extrapolation. The complete K value table is stored in the computer program. For δ_0 less than 10 degrees, and for δ_0 less than 15 degrees at Mach numbers less than 1.5, the linear theory solution is used in place of this K-value table.

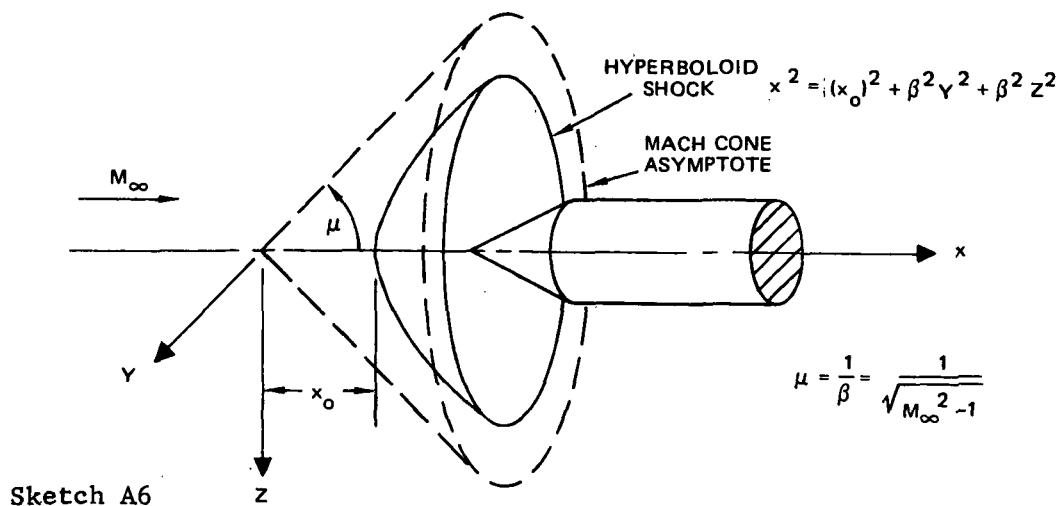
A second modification to the method of Reference 8 as used here should be noted. This concerns the choice of a circular arc to approximate the body forward portion. The "eye-ball fit" method of Reference 8 has been replaced by a uniform procedure in the computer program. As shown in the sketch below, the circular-arc nose used by the program is the one that is tangent to the original nose at both the apex and the aft body. This procedure effectively extends the method to bodies with little or no forward curvature, such as cone-cylinders.



Sketch A5

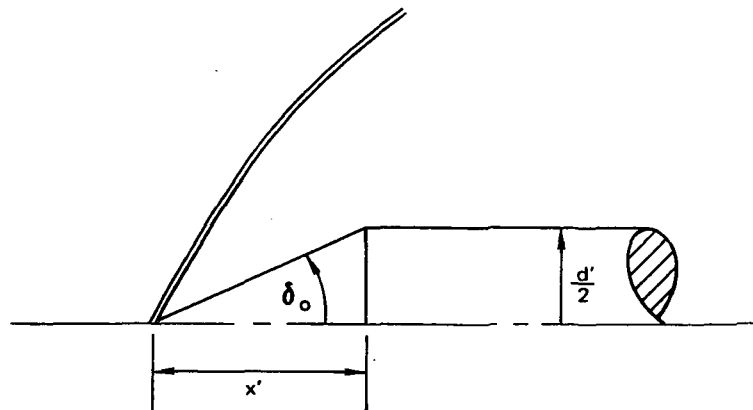
DETACHED SHOCKS

The method used to predict the detached shock geometry is that proposed by Love in Reference 9. The shock shape is assumed to be a hyperboloid as shown in the following sketch.



To completely specify this shock for a particular problem, two location points must be predicted. The first point to be determined is the shock apex location and the second is the location of the apex of the Mach cone asymptote.

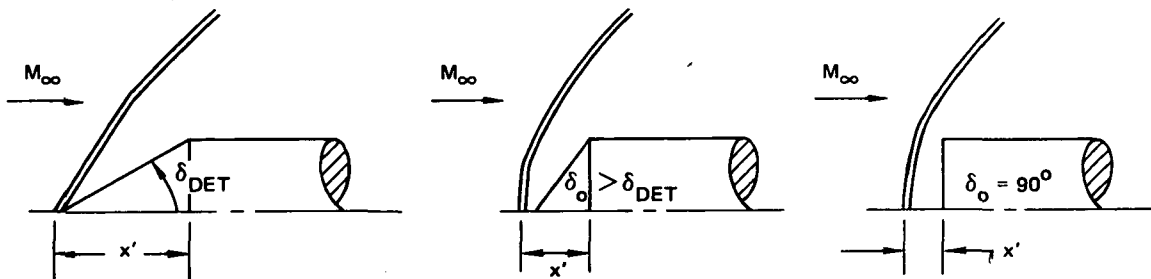
To examine the general approach in evaluating the shock standoff distance, consider the cone-cylinder with an attached shock as shown in the following sketch.



The cylinder diameter, d' , can be related to x' by equation (A-5)

$$\frac{x'}{d'} = .5 \cot \delta_o \quad (\text{A-5})$$

The angle δ_o can be increased to δ_{DET} before the shock will detach. If δ_o is increased beyond δ_{DET} , the apex of the detached shock and the cone apex both recede toward the cylinder as shown in the sketch below.



Sketch A8

The maximum value of x' for the detached shock can be expressed by equation (A-6).

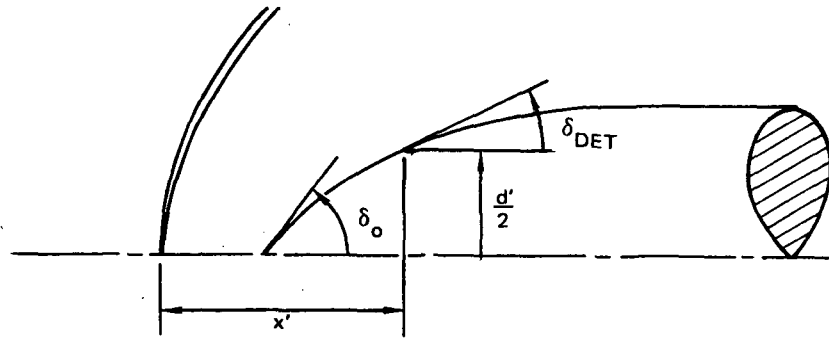
$$\left. \frac{x'}{d'} \right|_{\max} = .5 \cot \delta_{DET} \quad (\text{A-6})$$

Equation (A-7) is proposed in Reference 9 as a modified form of equation (A-5) for the case of a detached shock.

$$\frac{x'}{d'} = .5 (C) \cot \delta_{DET} \quad (\text{A-7})$$

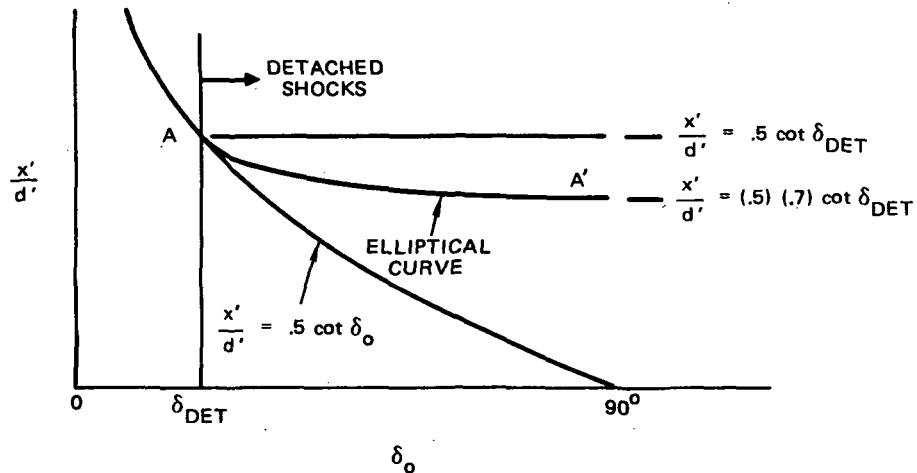
The value of C is an empirically derived value. For $\delta_o \gg \delta_{DET}$, a value of $C = 0.7$ appears to be appropriate for the cone-cylinder. To apply this equation to ogive and hemisphere noses, the measurements of x' and d' are taken from the body station at which the local body slope

is equal to $\tan \delta_{DET}$ as shown below.



Sketch A9

For the hemisphere nose, the value of C is found to be a function of M_∞ and a tabular relation is given in Reference 9. If the ogive is very blunt, i.e., the shock apex is not close to the nose apex, a value of $C = 0.7$ is again found to be appropriate. Love has proposed a modification to this method for the cone-cylinder if the angle δ_0 is close to the value of δ_{DET} . This method can best be explained with the following sketch.



Sketch A10

For values of δ_0 between δ_{DET} and 90° , an elliptical curve is proposed to determine $\frac{x'}{d'}$ between the limits of $(.5) \cot \delta_{DET}$ and $(.5) (.7) \cot \delta_{DET}$. The elliptical curve is tangent to the curve $\frac{x'}{d'} = .5 \cot \delta_{DET}$ at the point A and the ellipse has a slope of zero at the point A'.

The second evaluation to be made to specify the shock geometry is the location of the apex of the Mach cone asymptote (see Sketch A6). Love has proposed a procedure that is a modified version of Moeckel's method from Reference 11. The equation given by Love in Reference 9 is

$$\frac{x_0}{d'} = \frac{\beta \sqrt{\beta^2 \tan^2 \epsilon_s - 1} \left(\frac{x'}{d'} + \frac{\tan \eta}{2} \right)}{\beta^2 \tan \epsilon_s - \beta \sqrt{\beta^2 \tan^2 (\epsilon_s) - 1} + \tan \eta} \quad (\text{A-8})$$

$\epsilon_s \sim$ shock angle for sonic velocity after the shock

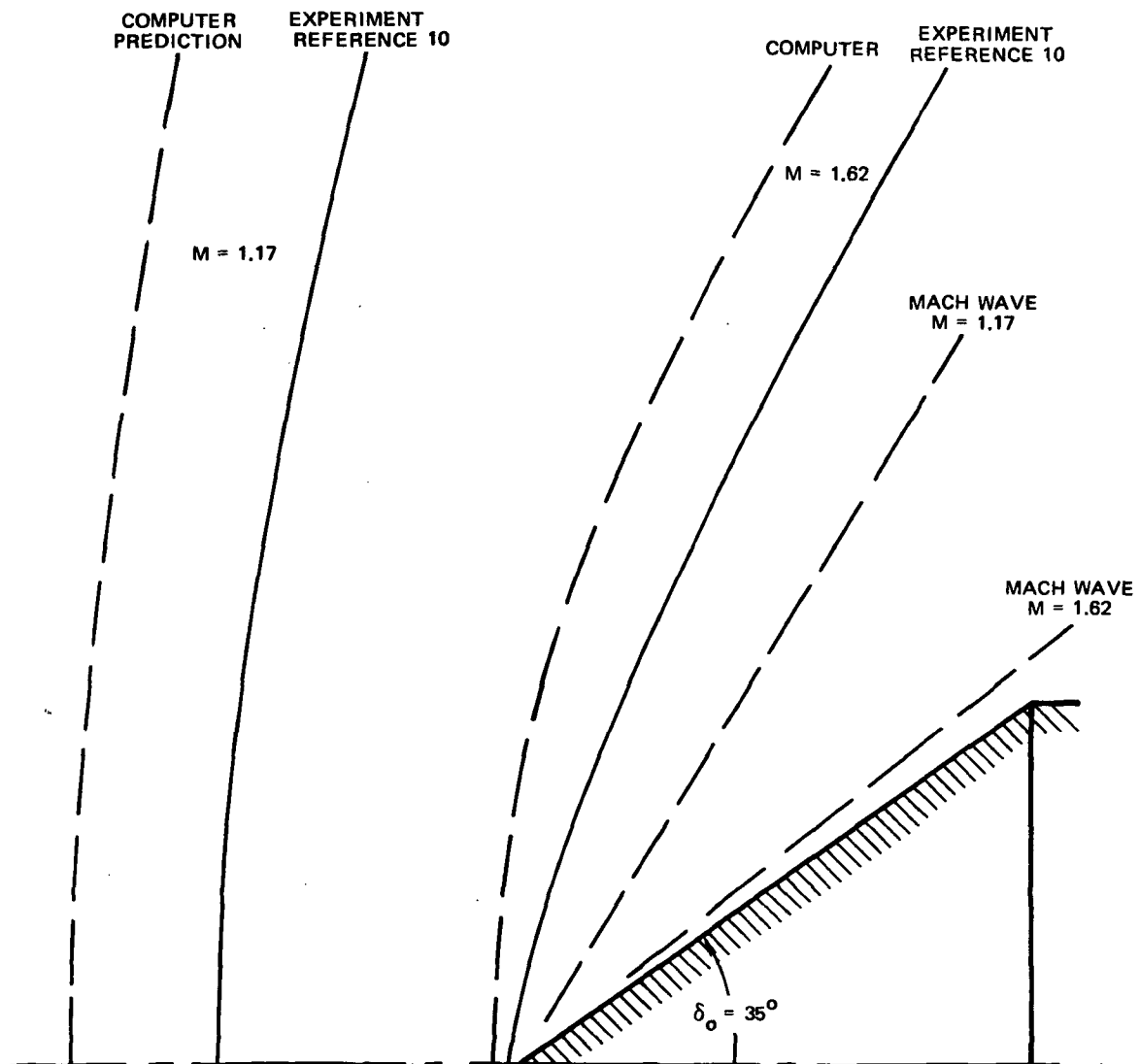
$\eta \sim$ correlation parameter plotted in Reference 9 as a function of M_∞

The shock location is then specified by a hyperboloid with apex at the stand-off distance x' and with Mach wave asymptotes originating at x_0 .

The method of Reference 9 for locating detached shocks, as presented above, had not been examined in Reference 9 for applicability at transonic speeds. A check on the method's accuracy at transonic speed was made for the present study and is shown in Figure A-1. This figure shows that the method predicts a shock ahead of the experimental shock, but provides a much more realistic disturbance location than the corresponding Mach wave.

Some difficulty was found in using Love's method as applied to an ogive nose for which the shock is only slightly detached. This difficulty arises because the diameter of the body at the effective shoulder (d' , see Sketch A9) lies very near the nose apex and

Figure A-1. Comparison of Computer Shock Shapes
With Experimental Data at Transonic Speeds.



becomes very small for detached shocks that are nearly attached.

This causes the Mach asymptote to be unrealistically close to the body (through Equation 8) which produces rapid bending of the shock wave to a position behind the actual shock. This effect is illustrated in Figure A-2. To correct this problem, the method used in the computer program is to always equate d' to the maximum body diameter.*

This effectively represents all bodies as cone-cylinder bodies for the purpose of computing detached shock location. Such a representation is shown in Figure A-3. This representation gives a shock location which lies somewhat ahead of the actual shock but which provides an adequate solution over a broader Mach range than is supplied by the method of Reference 9.

* This correction is not required and not performed for the special case of a hemisphere cylinder.

Figure A-2. Prediction of Detached Shock Using
Love's Method With No Modification.

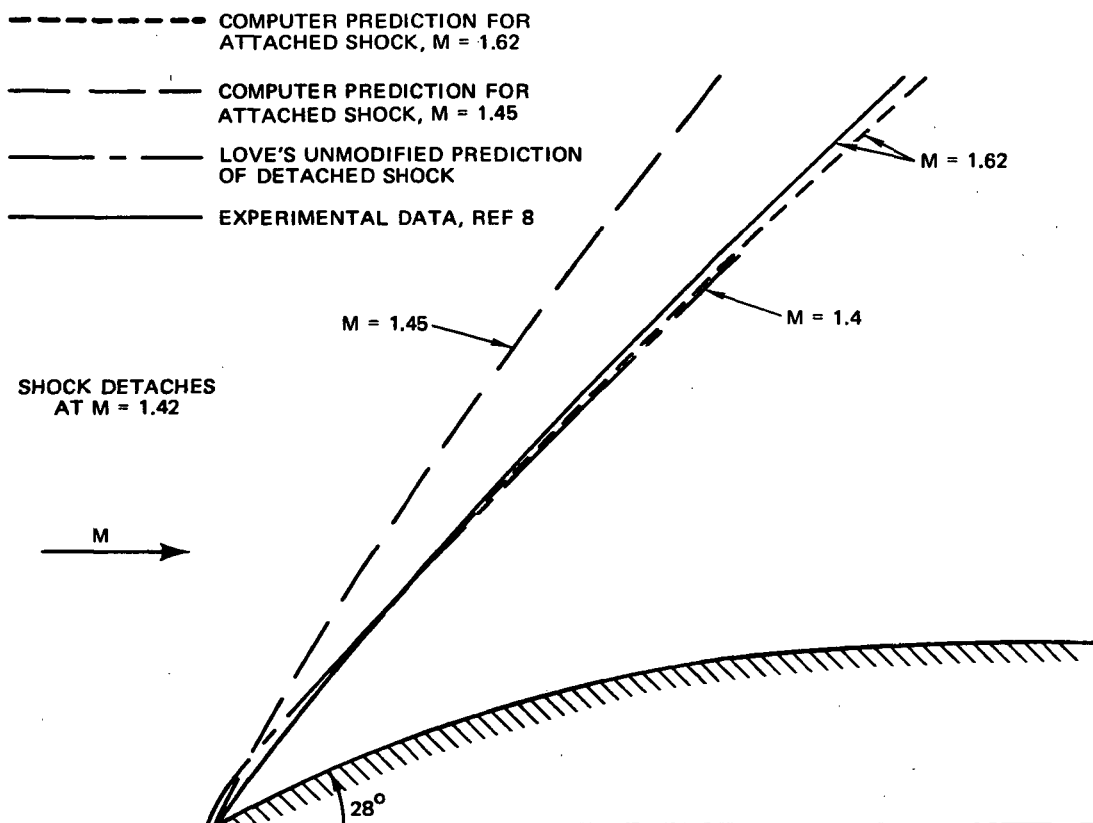
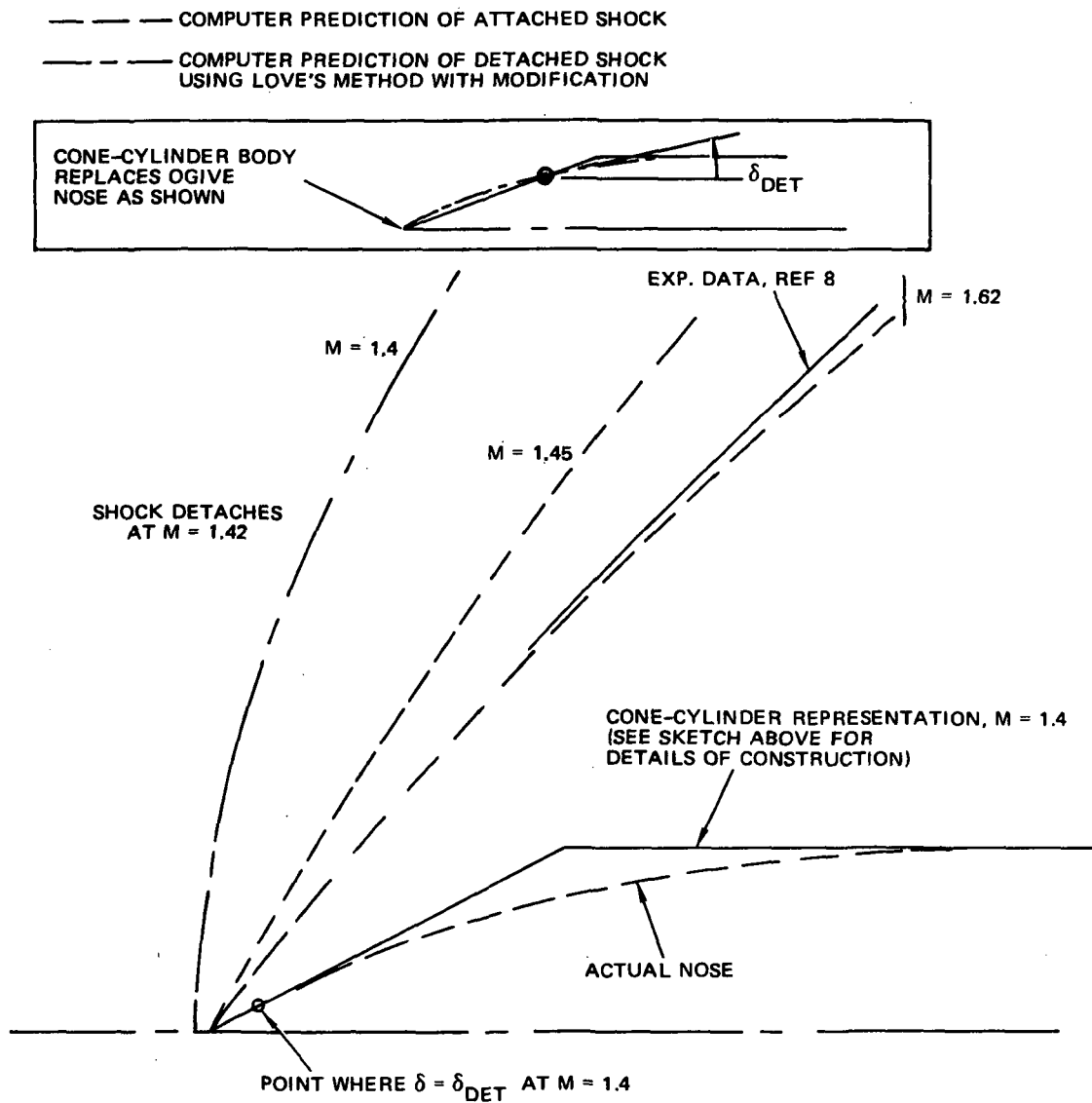


Figure A-3. Modification of Love's Method for Detached Shocks - Equivalent Cone Body is Constructed Through the Nose Apex and Through the Nose Point Where $\delta = \delta_{DET}$ (Cone Always has Detached Shock).



APPENDIX B - Computer Program STORLD
User Information (Supersonic Flow)

CONTENTS

	<u>Page</u>
INTRODUCTION	B-2
INPUT VARIABLE DEFINITIONS	B-3
OUTPUT VARIABLE DEFINITIONS	B-14
DISCUSSION OF PROGRAM USE	B-17
EXAMPLE INPUT VALUES	B-22

INTRODUCTION

Program STORLD is the main program for predicting the interference loading on aircraft stores. User information for the program version applied to subsonic flow is presented in Reference 1. This Appendix presents user information for applying the program to supersonic flow. All of the discussion of Reference 1 on program use in subsonic flow is also applicable to supersonic flow unless specifically mentioned in this Appendix. It is therefore recommended that the reader be familiar with the information of Reference 1 before attempting to apply the method to supersonic flow.

The supersonic version of program STORLD differs from the subsonic version in that many of the subroutines are different. Reference 2 contains a subroutine list for each version. A second difference is that the supersonic version requires a supplementary program, program SHOCK, to locate aircraft shock waves. User information for program SHOCK is presented in Appendix C.

INPUT VARIABLE DEFINITIONS

The X, Y, Z coordinate system used to represent the aircraft in supersonic flow is the same as that used in Reference 1 for subsonic flow. The logical flow of the program is also the same as Reference 1 and is repeated in Figure B-1 for convenience.

The input format for program STORLD is shown in Figures B-2. Many of these variables have the same definitions as in Reference 1 for subsonic flow. However, few of these variables will have the same numerical values for subsonic and supersonic representation of the same aircraft and store.

The input variables will now be defined. Dimension limits are given for index parameters; if these limits are exceeded, an error signal will result. The length units are given in feet (FT), although any consistent length unit may be used.

AIRCRAFT

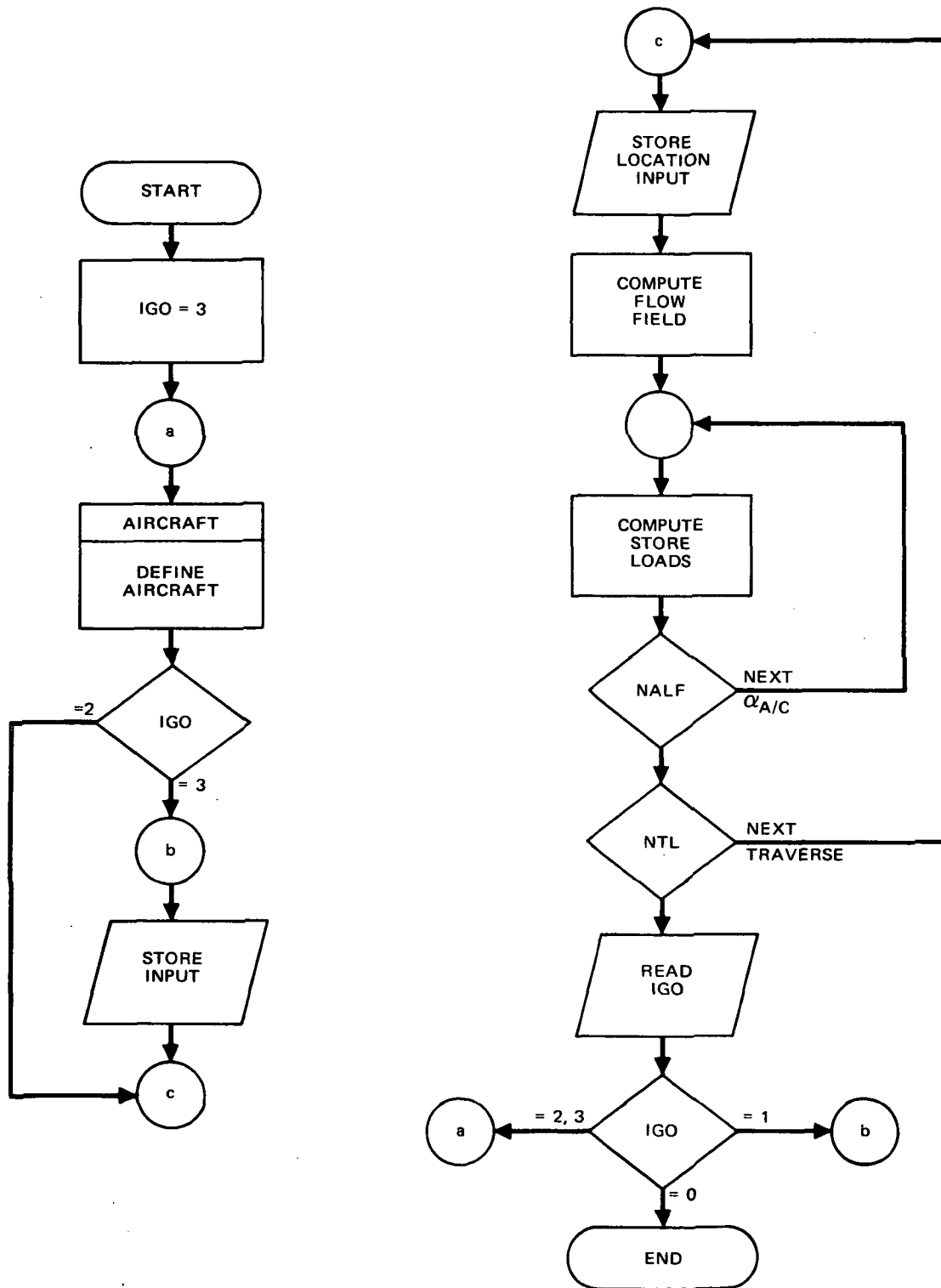
TITLE - To identify aircraft components included.

BM - Mach number for supersonic flow.

Wing

The aircraft wing is represented by a planar grid of Mach boxes as explained under AERODYNAMIC THEORY. Refer to Figure 6 for examples of wing representation. The wing is assumed to be extended inside the body to form an equivalent wing, and no wing-body interference is included. The wing leading edge is assumed to be a straight line from root to tip. Dihedral is neglected. The wing may have either a subsonic

Figure B-1. Program STORLD Logical Flow.



FORTRAN CODING AND DATA FORM

PAGE **A** OF

PROBLEM															AWO															EWO															WAP																																			
PROGRAMMER																														PHONE EXT.															DATE																																			
1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40	41	42	43	44	45	46	47	48	49	50	51	52	53	54	55	56	57	58	59	60	61	62	63	64	65	66	67	68	69	70	71	72	73	74	75	76	77	78	79	80	
1	AIRCRAFT TITLE																																																																															
2	BM																																																																															
3	INPUT FOR WING, PYLON, NOSE, STRIPS																				(SEE OTHER SHEETS FOR																																																											
4	INPUT FOR STORE																				INPUT THAT GOES HERE)																																																											
5	NTL →																																																																															
6	STORE TRAVERSE TITLE																																																																															
7	XM										YM										ZM										XT										ALPMW										(REPEAT NTL																													
8	NALF →										IB →										IA →										IW →																				(TIMES, OMIT)																													
9	ALFW(1)										-										-										-										-										ALFW(NALF)																				(IF NTL=0									
10	IGO →																				(=0, END ; =1, → LINE 4 ; =2 OR 3, → LINE 1)																																																											
11																																																																																
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25																																																																																
26	Note: an arrow (→) indicates that the value is to be placed to the far right of its input format zone. Variables so marked are fixed point (starting with																																																																															
27	I, J, K, L, M, N) and are read with an I10 format.																																																																															
28																																																																																
29	* Omit store data after first pass if IGO = 2																																																																															
30																																																																																

Figure B-2. Program STORLD Input Format (Supersonic). (Sheet 1 of 3)

FORTRAN CODING AND DATA FORM

PAGE 3 OF

PROBLEM		AWO		EWO		WAP	
INPUT FOR WING, PYLON, NOSE, STRIPS (SUPERSONIC FLOW)							
PROGRAMMER				PHONE EXT.		DATE	
	1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 21 22 23 24 25 26 27 28 29 30 31 32 33 34 35 36 37 38 39 40 41 42 43 44 45 46 47 48 49 50 51 52 53 54 55 56 57 58 59 60 61 62 63 64 65 66 67 68 69 70 71 72 73 74 75 76 77 78 79 80						
1	LU → MO →						
2	NSS → NSC →						
3	APEX S SM ZW						
4	CS(1) - - - - CS(NSS)						
5	TTC(1) - - - - TTC(NSS)						
6	ALFS(1) - - - - ALFS(NSS)						
7	XCAM(1) - - - - XCAM(NSS)						
8	THU(1) - - - - THU(NSC)						
9	DZDX(1) - - - - DZDX(NSC)						
10	CF(1) - - - - CF(NSC)						
11	DALFDX XALFO						
12	XMAX						
13	IG → IP → IW → (USE IG=0 IF NSS=0)						
14	DK(1,1) DK(2,1) DK(I,J) - - -						
15	((REPEAT CARD 14, I=1, ITR(J)), J=1, JDI)						
16	DW(1,1) DW(2,1) DW(I,J) - - -						
17	((REPEAT CARD 16, I=1, ITR(J)), J=1, JDI)						
18	X1 X2 X3 X4 Z1 Z2 YP XMAXP						
19	IG → IP →						
20	DKP(1,1) DKP(2,1) DKP(I,J) - - -						
21	((REPEAT CARD 20, I=1, IPS(J)), J=1, JDIP)						
22	DWP(1,1) DWP(2,1) DWP(I,J) - - -						
23	((REPEAT CARD 22, I=1, IPS(J)), J=1, JDIP)						
24	NBP →						
25	XNOSE YNOSE ZNOSE ALFNW XS XD						
26	XB(J) RB(J) (REPEAT LINE 26 FOR J=1 TO NBP) (OMIT IF NBP=0)						
27	NBT → NIN →						
28	XLIP YLIP ZLIP						
29	XST(J) YST(J) ZST(J) SST(J) THST(J) DXST(J) PHST(J) ALST(J)						
30	(REPEAT LINE 29 FOR J=1 TO NST, OMIT IF NST=0)						

Figure B-2. Program STORLD Input Format (Supersonic). (Sheet 2 of 3)

PAGE C OF

B-7

Figure B-2. Program STORLD Input Format (Supersonic). (Sheet 3 of 3)

leading edge or a supersonic leading edge. It is assumed that the store is not within the influence of the wing wake.

- LU, MO - Index limits used in determining the number of Mach box aerodynamic influence coefficients (AIC's) to be generated and used for computing wing source distributions. LU = chordwise index. MO = spanwise index. For initial runs, set LU and MO equal to max limits. Thereafter, LU and MO may be reduced to save run time if accuracy is not affected. Use MO = JDI, LU = 10 as a guide (see output list for JDI). For stacked runs, AIC's will not be redefined if LU and MO do not exceed values of previous runs.
(LU $\leq$ 20, MO $\leq$ 40)
- NSS - Number of spanwise sections into which semi-wing is divided for Mach box representation. (NSS $\leq$ 39)
- NSS must be chosen sufficiently small to allow storage space for the tip diaphragm (JDI $\leq$ 40, see output). Use of NSS = 39 causes the program to truncate the wing chord so that there is no tip diaphragm.
- NSC - Number of chordwise sections into which wing is divided for representing wing camber and thickness. NSC corresponds to NBV in subsonic flow. NSC = 0 for NSS $\neq$ 0 causes an error signal. (NSC $\leq$ 10)
- APEX - Axial coordinate of wing apex for equivalent wing, extended inside the body. This is the point where the projected leading edge intersects the Y = 0 plane. (FT)
- S - Semi-span of wing Mach boxes. S corresponds to SV in subsonic flow. S = wing semi-span over 2 · NSS. (FT)
- SM - Tangent of the sweep angle of the wing leading edge, from the normal to the free stream. (no units)
- ZW - Z coordinate of the plane of the wing. The wing is assumed to be planar. If the wing is not planar, define ZW as the coordinate of the wing directly above the store. (FT)
- CS - Chord length at each wing section. CS corresponds to C in subsonic flow. (FT)

- ALFS - Same as ALFV of Reference 1.
- TTC, XCAM,
THU, DZDX,
DALFDX,
XALFO - Definitions are the same as in subsonic flow, Reference 1.
Numerical values will probably differ, however.
- CF - Chord fractions. CF corresponds to CS in subsonic flow,
Reference 1. (no units)
- XMAX - X coordinate of most aft point on the wing at which Mach box
source values are to be computed. Evaluate XMAX from the
fact that Mach Boxes aft of the most rearward store Mach
forecone cannot affect the store. A small XMAX saves
run time for computing Mach box source values. The program
reduces XMAX to fit storage space if dimensions are
exceeded by the input value. Also, the program increases
XMAX to fall behind the leading edge tip location if the
input value is smaller than this. This is necessary in
order to satisfy indexing requirements. (FT)
- IG, IP,
IW - Control input; = 1 yes, = 0 no, to generate, punch, and write
values of DK and DW for wing. IG = 2 generates both DK
and DW. IG = 1 generates DK, reads DW. IG = 0 reads both
DK and DW.
- DK, DW - Mach Box source strengths on the leading edge diaphragm
and tip diaphragm of the wing. DK is due to camber, twist,
and/or superimposed crossflow. DW is per unit angle of
attack. Values are per unit free stream velocity. Run
time is saved by generating and punching these values on an
initial run, and inputting the punched data on subsequent
runs. All other wing input must be unchanged between runs.
(DK no units, DW per degree)

Pylon Crossflow

The following variables describe the pylon for the purpose of
computing wing-pylon crossflow interference. Unlike subsonic flow, the
pylon need not be placed at a juncture between wing sections. Cal-
culation of wing velocities normal to the pylon is actually more accurate
if the pylon is not too close to the juncture between Mach boxes; and

the program shifts the pylon spanwise for this calculation when necessary. The pylon dihedral angle is assumed to be 90 degrees. The pylon upper and lower edges are assumed to be parallel to the X axis. The pylon is approximated by two chordwise rows of Mach Boxes, with associated leading edge-tip-wake diaphragms. These Mach Boxes are imaged above the wing in the same manner as the pylon vortices in subsonic flow (see Figure 12 of Reference 1).

- $X_1, X_2,$
 X_3, X_4 - These are the X coordinates of the corners of the planform of the pylon.
 X_1 = upper leading edge corner.
 X_2 = lower leading edge corner.
 X_3 = lower trailing edge corner.
 X_4 = upper trailing edge corner. (FT)
- Z_1 - Z coordinate of pylon upper edge, at juncture with wing.
 Z_1 = ZW if wing has no thickness. (FT)
- Z_2 - Z coordinate of pylon lower edge. (FT)
- YP - Pylon spanwise location. (FT)
- XMAXP - X coordinate of the most aft point on the pylon or in its wake where pylon Mach Box values are to be computed. Equate XMAXP to the store base most aft location. The program reduces XMAXP to fit storage space if dimensions are exceeded by the input value. (FT)
- IG, IP,
DKP, DWP - These variables for the pylon have the same definitions as the corresponding variables for the wing (refer to wing input).

Nose

The variables listed in the data input format, Figure B-2, for nose input have the same definitions as those for subsonic flow,

Reference 1. Also, the same numerical values can be used for a given nose geometry. This is true with the exception of the variables defined below.

- XS - Axial distance from nose apex to nose shoulder (see Figure 8) (FT)
- XD - Axial distance between nose bow shock and forward Mach wave at the store traverse line. XD is calculated by program SHOCK as explained in APPENDIX C. Program STORLD transforms the nose geometry to an "equivalent body" for linear theory calculations (see Figure 8) (FT)

Thickness Strips

Thickness strips are used to describe the aircraft inlet ramp and pylon thickness in supersonic flow, similar to the representation of these components in subsonic flow. Strip geometry is shown in Figure 13 of Reference 1. The geometric input variables used to describe these thickness strips are defined the same for subsonic flow and supersonic flow. However, the flow field produced by these strips is calculated differently by the program in supersonic flow, and this causes some of the supersonic numerical input to be different from that used in subsonic flow. One difference is that, in subsonic flow, each strip is represented by a swept line source at the midchord. This requires that the strip chord (DXST) be kept small relative to the distance of the strip from the store, as explained in Reference 1. In supersonic flow, each strip is represented by a planar source distribution of parallelogram shape. It is therefore not necessary to keep DXST small. Another difference is the necessity to transform the aircraft compressive surfaces into an "equivalent" shape to apply linear

theory where shock effects exist. This transformation is performed by the program for the aircraft nose but must be performed by hand for surfaces represented by thickness strips, since the thickness strips are not ordered axially, nor do they have a defined shoulder location. This transformation modifies the input values of XST and DXST for supersonic flow relative to their values for the same aircraft in subsonic flow (XST is decreased and DXST is increased in a manner to preserve geometric similarity, as explained in section D of the AERODYNAMIC THEORY).

In calculating the flow field produced by the inlet, the supersonic program computes zero interference from the inlet ramp if the field point is inside the Mach cone of the inlet lip. This accounts for the re-turning of the flow by the inlet lip (approximately). Internal spillage is not included. The inlet lip Mach cone has apex at XLIP, YLIP, ZLIP (input). Choose this as the point on the inlet lip which lies closest to the store. Use of NIN is the same as in the subsonic program.

Store

Definitions of input variables used to represent the store in supersonic flow are the same as in subsonic flow. The input sheet of Figure B-2C is identical to that of Reference 1, and the user should use Reference 1 for definitions to these variables. For store components which are slender such that slender body theory is used to predict the aerodynamic coefficients, numerical values for these coefficients in supersonic flow will be the same as in subsonic flow. For store

components which are not slender, these coefficients will change with Mach number. Use Appendices E and F of Reference 1 as a guide for evaluating these coefficients.

Store Location

Program input for store location and computation mode in supersonic flow is identical to that in subsonic flow. Refer to Reference 1 for definitions of input variables from NTL through IGO.

OUTPUT VARIABLE DEFINITIONS

Output of the supersonic program is different from that of the subsonic program in the way the wing, nose, and pylon intermediate values are outputted. The wing, nose, and pylon output is defined here. Refer to Reference 1 for output definitions related to the thickness strips, store, velocity field, and interference coefficients.

Wing Parameters

Zartarian

Coeff's - These are the AIC values of Reference 4, computed by the equations derived in section A* of this report. These coefficients are used to compute the DK and DW matrices on the wing and pylon diaphragms. Section A explains the diaphragm concept.

II, JDI - Index limits for the Mach Boxes on the diaphragm of the wing. II is the tip diaphragm maximum chordwise index and JDI is the maximum spanwise index. Computation and use of II and JDI are explained in Reference 2 under Subroutine DIAFRM. Program will reduce values of II and JDI if limits are exceeded.

Wing section C_L 's

Wing total

C_L - Wing lift coefficients (no units). Section C_L 's are computed at each spanwise section J, J = 1 to NSS. They are referenced to the local chord of the Mach Box wing. Total C_L is referenced to the area of the Mach Box wing. The Mach Box wing approximates the actual wing as shown in Section B*. These C_L values apply only to that portion of the wing which lies ahead of the final output value XMAX. Therefore, if these C_L values are to apply to the inputted wing planform, XMAX must lie entirely behind the wing. Wing lift is not computed if DK and DW values are read in (see below).

Wing DK, DW

Matrices - These are the source values for the Mach Boxes used to represent the wing and diaphragm. DK is due to wing camber, twist, and/or superimposed crossflow (no units).

* Contained under AERODYNAMIC THEORY.

DW is due to unit angle of attack (per degree). Values are printed versus J, the spanwise index for the Mach boxes. Values at each J are printed from the leading Mach wave back to the wing trailing edge, in order. The wing wake is not considered. The program has the option to punch out DK and DW values on the diaphragm, running from the leading Mach wave back to the wing leading edge. Punched values are then used as input to save run time (see input).

- ILE, ITE - Chordwise index of Mach boxes on wing at leading and trailing edges at each spanwise section J, J = 1 to NSS. These values are also defined on the tip diaphragm, J = NSS + 1 to JDI. On the tip diaphragm, ILE = ITE + 1, and ITE is defined as the maximum chordwise index at each section.
- YS - Y coordinate of wing sections and tip diaphragm sections, J, J = 1 to JDI. YS is taken to the mid-span of each section.
- XS - X coordinates of the wing leading edge at the mid-span of each wing section J, J = 1 to NSS.
- CS, ALFS,
TTC, XCAM,
THU, DZDX,
CF - See input list.
- DT Matrix - Wing source values due to thickness, camber, and twist. The DT source grid is not a Mach Box grid but is a grid of rectangular boxes using the CF (input) chordwise divisions. The Mach box spanwise divisions are retained. The DT matrix lies entirely on the wing and has dimensions NSS x NSC. Note that the flow field produced by wing camber and twist is computed using DK on the diaphragm and DT on the wing. This is done for computational efficiency.

Pylon Parameters

- XMAXP - X coordinate of most aft point where pylon Mach Box source values are to be computed (input). If input value of XMAXP causes a dimension over-run, the program reduces XMAXP and notes this fact. Pylon flow field calculations beyond XMAXP are not valid.
- IIP, JDIP - Index limits for the pylon Mach boxes, similar to II and JDI for the wing.

- APEXP - Apex of pylon Mach box system, equals X coordinate of forward Mach line at $Z = Z1$. Refer to AERODYNAMIC THEORY, Sect. A.
- ILEP, ITEP - Pylon Mach box chordwise index at leading and trailing edges. Corresponds to output values of ILE and ITE for wing. Spanwise direction for pylon is defined as vertically down. Pylon is divided into two spanwise sections, $J = 1, 2$. On tip diaphragm, $J = 3$ to JDIP.
- IPS - Pylon Mach box maximum chordwise index at each spanwise section. IPS includes pylon wake. IPS is used to write DKP and DWP matrices and is used to compute pylon flow field. Note, wing Mach box system uses ITE for these operations because wing wake is not considered.

Pylon
DKP, DWP
Matrices

- These are the source values for the Mach boxes used to represent the pylon and diaphragm. Values are proportional to crossflow at the pylon produced by the wing. DKP is due to wing camber, twist, and thickness (no units). DWP is due to wing unit angle of attack (per degree). Values are printed versus spanwise index J . $J = 1, 2$ is on pylon. $J = 3$ to JDIP is on tip diaphragm. Values at each J are printed from leading Mach wave back to where $X = XMAXP$, in order. These values include leading edge, tip, and wake diaphragms. Pylon Mach boxes are pictured in Reference 2 under subroutine DIAFRM. The program contains the option to punch out all these DKP and DWP values, so that they may be used as input values in subsequent runs, with reduced run time.

Nose Parameters

- XBL - Not used for supersonic flow.
- XB, RB - These are not the input values of the nose control points, but are the control points of the "equivalent" nose, transformed according to the method shown in Figure 8. If point XB, RB is outside the cone of influence from the nose apex, this fact is noted on the output and point XB, RB is ignored. In this way the nose shape is modified to be more slender so that the linear theory method can be applied. The user should verify that the more slender nose is not a poor approximation to the actual nose.

DISCUSSION OF PROGRAM USE

This section will discuss considerations the user should keep in mind for obtaining the most accurate results possible from the prediction program. Two major considerations mentioned in Reference 1 for subsonic predictions are also important for supersonic predictions. They are: 1) perform adequate sensitivity studies to verify that the input represents the aircraft and store accurately; 2) compare load predictions with test data for a similar store under the subject aircraft, if data is available.

Buoyancy and Computation Mode

The supersonic prediction program computes only part of the store buoyancy loading, as explained in the text. Test case calculations have shown this buoyancy prediction to be of negligible magnitude. However, the buoyancy prediction can become erroneously large in some cases, due to lack of precision in the method. For this reason it is recommended that buoyancy not be included in using the supersonic prediction program in its present state of development. This may be done by input of $IB = 0$ for computation mode. However, for stores with high aspect ratio fins, input of $IB = 0$ will prevent the program from computing the crossflow at the fin tips, which reduces the accuracy of the crossflow loading prediction. In this case the buoyancy loading can be set to zero by using the store input values $RS(I) = 0$, $I = 1$ to NMS . The computation mode inputs of $IB = 1$, $IA = 0$ will then give an accurate crossflow calculation and zero buoyancy.

Precision of the Prediction

The supersonic interference flow field can change quite rapidly with field location, such that the store sectional division length (DELX, input) may be too large for accurate calculation of store loads. To avoid this possibility, it is recommended that the precision of the calculation be tested by shifting field calculation points axially by a value of DELX/2.0. This may be done by incrementing the input value of XSTART by DELX/2.0; if the loading prediction changes, then a smaller DELX input value should be used.

Aircraft Shocks

Section D of AERODYNAMIC THEORY says aircraft compressive surfaces are to be transformed to equivalent shapes for applying linear theory where shock effects exist. This transformation is a function of store location, which disallows computer program run stacking of variable store location for fixed aircraft description. Redescription of the aircraft for every store location adds greatly to user effort. For this reason, a compromise is suggested; describe the aircraft appropriately for the store at the mate location; use this same aircraft description for store locations off mate. In this way the number of aircraft descriptions is reduced to the number of pylons used for carriage. While off mate prediction accuracies will be somewhat compromised, the effect on separation studies will in most cases be minor.

Run Time

Run time to compute the flow field for one store axial traverse from mate to a position forward of the aircraft, including buoyancy loading on the store and including all the aircraft components that the program is capable of describing, will average between 20 and 50

seconds (CDC 6400 central processing time). Run time for traverses below mate is less than this because the store is removed from the Mach cone of influence of the aft parts of the aircraft. Run time is approximately proportional to the number of aircraft singularities times the number of field points to be calculated. This is much like the subsonic prediction method (see Reference 1, Figure 15). However, run time cannot be computed this easily because aircraft singularities outside the Mach forecones of the field points do not contribute greatly to run time. Experience with test case calculations indicates that the supersonic program is about twice as expensive to run as the subsonic program for calculating flow fields and is about 3 times as expensive for calculating wing and pylon singularity strengths. These numbers apply to aircraft with subsonic wing leading edges. Run time will be less for wings with supersonic leading edges. Aircraft components listed in the order of most expensive to consider are: wing-ylon crossflow interference; wing with subsonic leading edge; fuselage nose; thickness envelope of pylon; inlet.

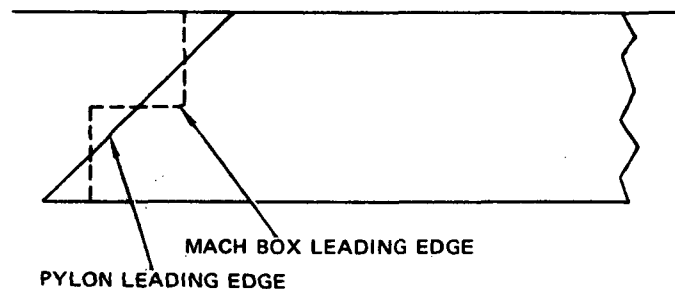
Store Location Limits

It is assumed that the store is not located within the influence of the wing wake. This assumption saves computer storage space and computing time. Also, it is assumed that the store is not located within the influence of portions of the wing which lie behind $X = X_{MAX}$ (input). And the store must not be influenced by the pylon or wake aft of the point $X = X_{MAXP}$ (input) when wing-ylon crossflow interference is computed. If input values of X_{MAX} or X_{MAXP} cause

available storage space to be exceeded, these values will be reduced by the program and this will be noted in the output. It is the user's responsibility to verify that the store is not so far aft that it is influenced by aircraft components which are not represented.

Wing-Pylon Crossflow Interference

The computer program uses two chordwise rows of Mach boxes to describe the pylon for wing-pylon interference calculations. If the pylon leading edge is swept, the Mach box approximation of the pylon leading edge is as shown in the sketch below.



Sketch B1

The sharp corners on the Mach Box leading edge can give unrealistic flow predictions for swept leading edges. This problem could be eliminated by using more than two rows of Mach boxes on the pylon. However, this leads to large run time and large storage space requirements (DKP and DWP matrices now take 4400 spaces each, for pylon and diaphragm). Wing-pylon interference does not contribute to store pitch loading, and it does not contribute a major portion of the yaw loading if the wing is at low angle of attack. Thus, even though the

prediction method may not predict accurate pylon crossflow for pylons with swept leading edges, it can nevertheless predict when pylon crossflow is important.

Conservative Estimates

A special feature of store interference loading in supersonic flow is that it can change drastically with store axial location. Comparison of test case data presented in this report shows that some prediction error can be expected in interference load axial location. Therefore, it is recommended that some axial shift of the predicted loading be made to increase carriage load when it is desired to make the prediction more conservative. This is especially necessary where loading variation with axial location is quite rapid, and it probably is a good procedure as applied to wind tunnel data as well.

Thickness Strip Inputs (for pylons, inlets)

The flow field predicted from linear theory for a source strip element in supersonic flow can change greatly with field location near the Mach cone of influence of the edges of the strip. For this reason it is important, when representing a continuous compression surface with more than one strip, that the strips fit together geometrically in a piecewise continuous manner. Unrealistic flow predictions may otherwise result. The strip elements are planar surfaces which are assumed to be parallel to the X-coordinate direction. This means that strips used in tandem to represent a compression surface must be coplanar to be piecewise continuous.

EXAMPLE INPUT VALUES

The following input index values were used in the F-4 aircraft and ARM and RAM store test case calculations presented in this report. These may serve as a guide to the user. Refer to the input list for definitions to these variables.

LU = 10	NST = 2 (inlet)
MO = 20	NST = 5 (inboard pylon)
NSS = 14	NST = 6 (outboard pylon)
NSC = 4	NMS = 12 (ARM)
NBP = 16	NMS = 6 (RAM)

The following shock shift values (XD) were predicted by Program SHOCK for the F-4 aircraft at Mach 1.2, and were used in the test case calculations.

	<u>XD for inboard stores (FT)</u>	<u>XD for outboard stores (FT)</u>
nose	4.7	6.0
inlet	4.0	4.8
inboard pylon	0.45	--
outboard pylon	--	0.40

APPENDIX C - Computer Program SHOCK
User Information

CONTENTS

	<u>Page</u>
INTRODUCTION	C-2
INPUT VARIABLE DEFINITIONS	C-4
OUTPUT VARIABLE DEFINITIONS	C-7
DISCUSSION OF PROGRAM USE	C-8

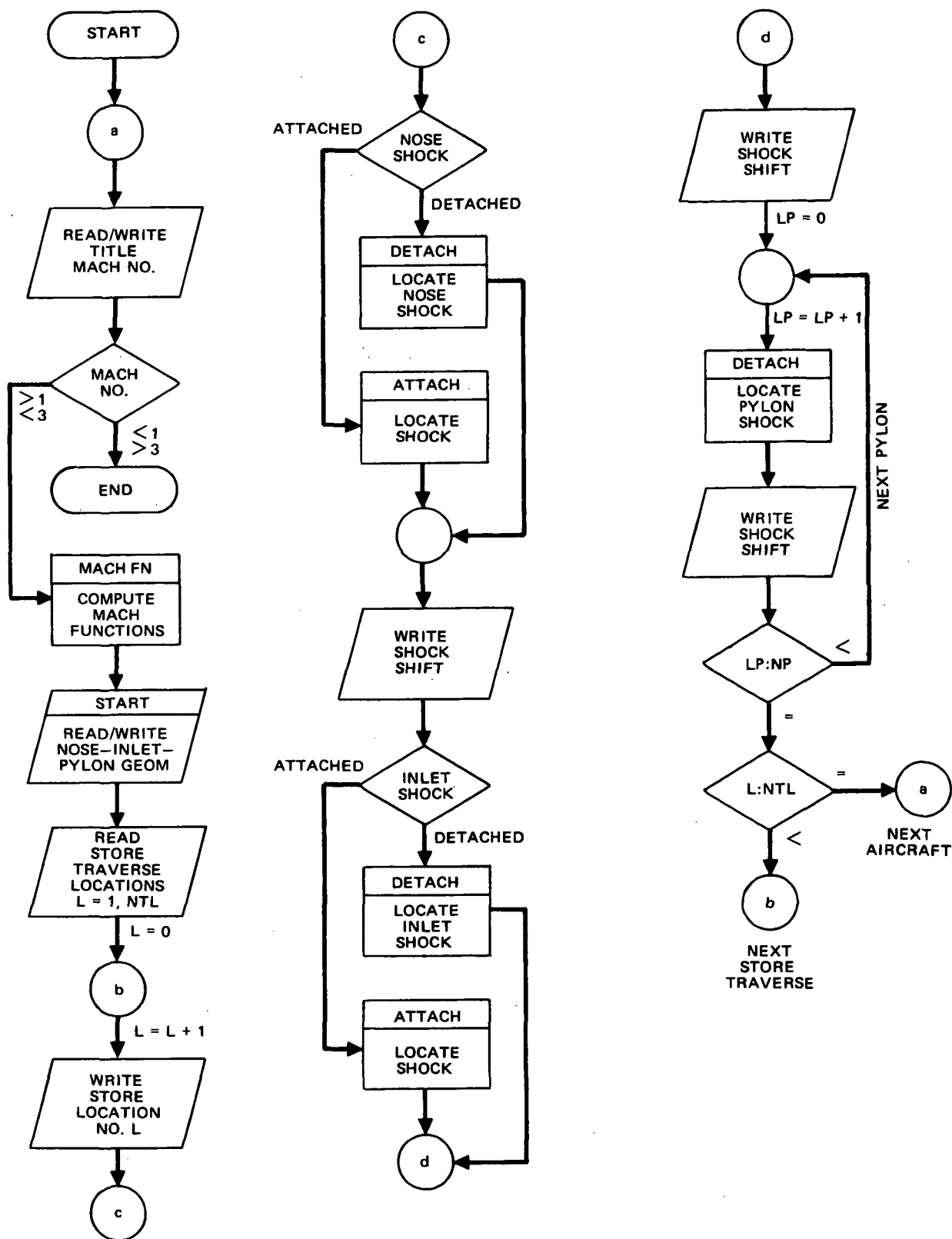
INTRODUCTION

Program SHOCK is supplementary to program STORLD. The purpose of program SHOCK is to calculate the locations of aircraft shock waves as they intersect the store. These shock locations are used in preparing the input data to program STORLD for representing the aircraft. Use of the shock location data is discussed in section D and Appendix B of this report. This Appendix will concern use of program SHOCK to obtain the shock locations.

The logical flow of program SHOCK is given in Figure C-1. The program considers the aircraft nose, inlet ramp, and pylons in that order. For nose and inlets, the program determines if the shock is attached or detached. Pylon shocks are assumed to be detached. The program then computes the shock location and outputs this in terms of a shift forward from the Mach wave at the intersection with the store. Run stacking for Mach number, aircraft geometry, and store location is allowed. The program is ended by inputting an off-limit Mach number.

Reference 2 contains a program analysis and test case runs for program SHOCK.

Figure C-1. Program SHOCK Logical Flow.



INPUT VARIABLE DEFINITIONS

For convenience, the coordinate system used to locate the aircraft and store for program SHOCK may be the same as that used for program STORLD. This allows some of the input numerical data to be used for both programs. See Reference 1, Figure 7, for coordinate system definition.

The input format for program SHOCK is shown in Figure C-2. These input variables will now be defined.

BM - Mach number, if outside the range 1 to 3 will stop the program.

Fuselage Nose

NBPP - Equals NBP if non-zero. NBP is the number of body points used to describe the nose. Input NBPP = 0 causes the program to use NBP as the last non-zero NBPP value from previously stacked runs, if any. (NBP $\leq$ 40)

ESO - Conical shock wave half-angle at the nose apex. For attached shocks, ESO is a unique function of nose apex angle and Mach number. (See for example Reference 7). For detached shocks ESO = 90. (degrees)

XNOSE, YNOSE,
ZNOSE - Coordinates of nose apex. (FT)

DELTA - Nose apex half-angle. (degrees)

XB, RB - Coordinates defining nose contour. XB is distance from nose apex along axis of symmetry. RB is nose radius at XB. Input values for program STORLD may be used here. (FT)

Inlet Ramp

The inlet ramp is inputted as a half-cone. If the inlet ramp is some other shape, it is to be approximated as a half-cone. This is explained later under DISCUSSION OF PROGRAM USE.

FORTRAN CODING AND DATA FORM

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PROGRAMMER																																																																						PHONE EXT.										DATE									
1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40	41	42	43	44	45	46	47	48	49	50	51	52	53	54	55	56	57	58	59	60	61	62	63	64	65	66	67	68	69	70	71	72	73	74	75	76	77	78	79	80										
TITLE																																																																																									
BM																																																																																									
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Note: an arrow (→) indicates that the value is to be placed to the far right of its input format zone. Variables so marked are fixed point (starting with I, J, K, L, M, N) and are read with an I10 format.																																																																																									

Figure C-2. Program SHOCK Input Format.

- XM, YM, ZM - Traverse line reference point. Traverse line passes through this point. (FT)
- ALFMW - Pitch incidence of traverse line relative to X coordinate. Plus ALFMW is nose down. Traverse line is assumed to have zero yaw incidence. (degrees)

OUTPUT VARIABLE DEFINITIONS

The following is an explanation of program SHOCK output, excluding input variables previously defined.

Mach Functions

- DDET - Cone half-angle at which shock just detaches. (degrees)
- ETA - Parameter η used in Reference 9 for determining detached shock asymptote location (see equation 8 of Appendix A). (degrees)
- ES - Shock angle to produce sonic velocity behind the shock. (degrees)
- C - Correction factor used in Reference 9 for calculating detached shock stand-off distance (see Equation 7 of Appendix A). (no units)

Nose Parameters

- slope - Derivative of RB versus XB. (no units)

Results

The traverse line location is printed. The shock is located by its distance ahead of the leading Mach wave at the traverse line; this distance is called DSHIFT. The equivalent body shape used to calculate the shock location is printed. For detached shocks, the stand-off distance and the asymptote origin are printed. See Appendix A for a definition of the detached shock asymptote.

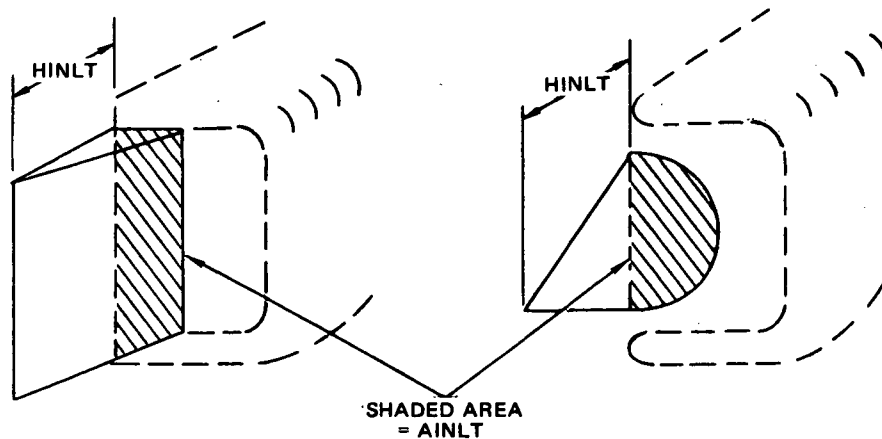
DISCUSSION OF PROGRAM USE

The computer program "SHOCK" predicts shock locations for axisymmetric bodies at zero angle of attack for three basic shapes: circular arc bodies, cone-cylinders, and hemisphere-cylinders. For detached shocks, the circular arc body is approximated as a cone-cylinder. For attached shocks, the cone-cylinder is approximated as a circular-arc body. To apply the programmed method to an aircraft it is necessary to approximate the aircraft shock-producing components by any of these three basic shapes. This requires some judgement on the part of the user to obtain good results. This judgement comes with experience, but the following discussion will act as a guide.

As a rule, if the field point is close to the body, then it is important to simulate the body geometry very near the body apex. If the field point is far from the body, then the body geometry further behind the nose apex, and the body maximum diameter, become more important. The user must verify that the body shape approximation being used by the program is a good approximation to the actual body over the portion of the body considered by the user to be most important. The program outputs the equivalent body shape it has used.

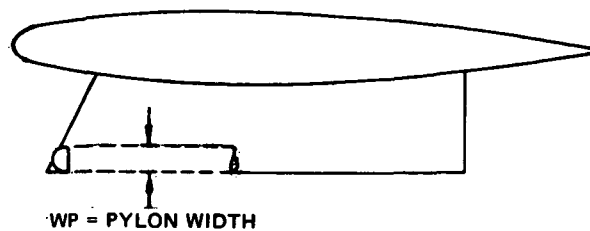
The input to program "SHOCK" reads the geometry of the aircraft nose, inlet ramp, and pylon leading edges. The nose shape is read as discrete points. The program then treats this body as a cone-cylinder for detached shocks, or as a circular-arc body for attached shocks. The inlet is read in as a length and an area for an equivalent

cone-cylinder. A wedge-type inlet ramp is thereby equated to a cone-cylinder as shown in the sketch below. For attached shocks, the program re-approximates the inlet ramp as a circular arc body.



Sketch C1

Pylons may be represented by an equivalent hemisphere-cylinder as shown in the following sketch.



Sketch C2

If the pylon leading edge cross section is more slender than a hemisphere shape, modification to the program input of pylon geometry is required. This can be done by inputting an equivalent hemisphere-cylinder shape of reduced diameter, which better fits the most forward portion of the pylon leading edge. If the pylon leading edge is not blunted but is

wedge shaped, the pylon leading edge may be inputted as if it were a nose, and the program will use a cone or arc body approximation.

Run Time

Program SHOCK run time is negligible.