DATA BASE FOR THE PREDICTION OF AIRFRAME/PROPULSION SYSTEM INTERFERENCE EFFECTS

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Results are presented from a study to define and evaluate the data base for predicting certain aerodynamic interference effects associated with airframe/propulsion system integration. The study was conducted for supersonic tactical aircraft with highly integrated jet propulsion systems, although some information is included for supersonic strategic aircraft and for transport aircraft designed for high-subsonic or low-supersonic cruise. Primary attention is paid to those interference effects which impact the external aerodynamics of the aircraft; however, information on other effects such as inlet internal performance is collected and organized, but not analyzed.

The study consisted of a literature search, the development of a framework to organize the data base (which is considered to include theoretical and empirical prediction methods as well as experimental data), and the evaluation of the state-of-the-art for three interference effects: inlet external drag, afterbody drag, and the effects of the airframe on the inlet flow field. Adequacy of prediction methods is assessed through comparison with experimental data.
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DATA BASE FOR THE PREDICTION OF AIRFRAME/PROPULSION SYSTEM INTERFERENCE EFFECTS

by

Nielsen Engineering & Research, Inc.

SUMMARY

Results are presented from a study to define and evaluate the data base for predicting certain aerodynamic interference effects associated with airframe/propulsion system integration. The study was conducted for supersonic tactical aircraft with highly integrated jet propulsion systems, although some information is included for subsonic strategic aircraft and for transport aircraft designed for high-subsonic or low-supersonic cruise. Primary attention is paid to those interference effects which impact the external aerodynamics of the aircraft; however, information on other effects such as inlet internal performance is collected and organized, but not analyzed.

The study consisted of a literature search, the development of a framework to organize the data base (which is considered to include theoretical and empirical prediction methods as well as experimental data), and the evaluation of the state-of-the-art for three interference effects: inlet external drag, afterbody drag, and the effects of the airframe on the inlet flow field. Adequacy of prediction methods is assessed through comparison with experimental data.

1. INTRODUCTION

There continues to be concern in the aircraft industry and the military services about the difficulty encountered in developing aircraft that successfully achieve the performance predicted in design studies, particularly for mixed mission aircraft with highly integrated...
jet propulsion systems. Part of the problem is the lack of an adequate
data base for the design studies which often leads to overly optimistic
performance predictions. The broad mission requirements for high per-
formance in the subsonic, transonic, and supersonic speed ranges, at high
and low altitudes, create the need for an extensive body of data for use
in mission performance studies. Of particular concern is the lack of
adequate theoretical or engineering methods for predicting the airframe/
propulsion system interference effects even though a significant body of
performance data for isolated components (wings, fuselage, engines, etc.)
is available.

Multi-mission aircraft configurations characteristically require that
the airframe and propulsion system be closely coupled for both structural
and aerodynamic efficiency. Consequently, there are large flow inter-
ference effects which significantly affect both lift and drag and
therefore cannot be ignored in performance predictions. The current
practice is to define these interference effects during development
testing after selection of a basic configuration. Because these inter-
ference effects are large, and in most instances adversely affect
performance, the codifying of methods for predicting interference effects
should help in meeting performance goals and avoiding expensive and time-
consuming redesign processes.

The selection of the "best" engine for a multi-mission aircraft is a
difficult and important task. Because the time required to develop and
qualify an engine is the same order as that for the airframe, it is
essential that the engine selection be made early in the aircraft
development cycle. Selection of the most effective engine cycle
parameters (i.e., compression ratio, turbine inlet temperature, bypass
ratio, etc.) is usually made on the basis of mission analysis studies.
These studies draw heavily upon existing performance data to establish
the best trade-offs for the most effective accomplishment of the desired
mission. Consequently, it is important that an adequate "data bank" of
component performance and related costs be used in these studies.
This report documents a study to define and evaluate the data base for predicting certain aerodynamic interference effects associated with airframe/propulsion system integration. Primary attention has been paid to interference effects which impact in an important way the external aerodynamics of the aircraft. For the sake of completeness, data for other effects which must be considered in predicting in-flight net thrust (e.g., inlet internal performance) have been collected and organized, but no analysis of this material is attempted. As used herein, the data base is considered to include theoretical and empirical prediction methods, as well as experimental data. This study is focused on interference effects for supersonic tactical aircraft with highly integrated jet propulsion systems, although some information is included for supersonic strategic aircraft and for transport aircraft designed for high-subsonic and low-supersonic cruise. No effort has been made to fully incorporate the substantial literature available for V/STOL aircraft, and hypersonic design point aircraft are specifically excluded.

In the first phase of this study, conducted under Contract NAS2-8874, a search of the classified and unclassified literature was made to identify the pertinent data. Use was made of the available indexing services, International Aerospace Abstracts (IAA), Scientific and Technical Aerospace Abstracts (STAR and CSTAR), the Government Reports Index (GRI), and the Technical Abstract Bulletin (TAB), as well as the computerized NASA RECON system. Searches were performed by the Defense Documentation Center. Additionally, letter and personal contacts were made with government and aerospace organizations actively engaged in the areas of interest. The latest available information pertaining to the F-14, F-15, and B-1 was obtained.

Also in the first phase, a framework to organize the data base was developed, and the information assembled in the ways described above was categorized using this framework. A preliminary evaluation of the suitability for preliminary design purposes of the data bases for two interference effects was conducted. These effects, inlet external drag
and afterbody drag, are areas which exert a considerable influence on aircraft performance and where a significant need for correlation exists. The work conducted under the first phase was reported in reference 8.1.*

In the second phase of the study, conducted under Contract NAS2-9513, in-depth evaluation of the state-of-the-art was conducted for three interference effects: inlet external drag, afterbody drag, and the effects of the airframe on the inlet flow field. The literature search conducted in the first phase was updated and reports newly acquired were classified using the framework from the first phase. Additionally, existing empirically based techniques to predict nozzle/afterbody drag of single and twin engine installations were developed for implementation into the aerodynamic prediction computer program being developed by the NASA/Ames Research Center. This last work, and the evaluation of these nozzle/afterbody drag prediction techniques using data from the data base, is separately reported in ref. 8.2. The present report is the final report for Contract NAS2-9513 and documents the remainder of the work conducted in the second phase. Because a considerable amount of new material was generated in this phase, this report supersedes reference 8.1.

In sections 2, 3, and 4, the entire data base is presented and categorized by the type of interference effect treated and the general nature of the treatment used in each reference. The effects of interest are broken down into two main areas: those involving inlet/airframe integration (section 2) and those involving afterbody/airframe integration (section 3). Section 4 deals with some special topics. The documents cited in this report are categorized according to which of the areas (inlet or afterbody) is treated and appear in either the inlet/airframe reference list (Appendix A) or the afterbody/airframe reference list (Appendix B). The number assigned to a report in either of these

*References identified as 8.X are listed in section 8 of this report. The references comprising the data base appear in Appendices A and B and are identified as described later.
lists corresponds approximately to the order in which the report became available to us. A combined cross-index, alphabetized by author name, is contained in Appendix C. In this report, the list to which a particular reference number pertains is clear from the context in which it appears.

The remainder of the report consists of the analysis in some depth of three areas. Section 5 presents the data base for inlet external drag in more detail and contains an evaluation of its adequacy for preliminary design. Sections 6 and 7 similarly treat afterbody drag and the effects of the airframe on the inlet flow field, respectively.
LIST OF SYMBOLS

A  area
A/A*  contraction ratio required to bring a reference flow to sonic velocity isentropically
A_c  inlet capture area - the frontal area of an inlet, projected in the free-stream velocity vector direction; for axisymmetric inlets, the area is bounded by the cowl leading edge, for others it is bounded by the cowl leading edge, side plate leading edges and initial ramp leading edge
A_e  area at nozzle exit plane
A_m  maximum fuselage cross-sectional area
A_MI  maximum projected frontal area of inlet
A_MB  cross-sectional area at metric-break station
A_P  projected frontal area
A_SO  total nozzle flow area plus base area at nozzle exit station (both nozzles)
A_t  area at nozzle throat
A_W  wing reference area
C_D, C_D  boattail pressure drag coefficient, based on A_m and A_P, respectively [see eq. (27)]
C_D_A(+)  additive drag coefficient, D_add/\rho A_c
C_D_BL  bleed drag coefficient, based on A_c
C_D_BLD  boundary-layer-diverter drag coefficient, based on A_c
C_D_BFP  by-pass drag coefficient, based on A_c
C_D_C, C_D_C  cowl drag coefficient based on A_c and A_MI, respectively (see fig. 1)
C_D_CALAC  CALAC predicted aft end drag [see eq. (39)]
C_D_{DEB}  equivalent body pressure drag coefficient, based on A_P

*All drag coefficients are based on \rho_\infty and the indicated area.
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<th>Description</th>
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<tr>
<td>$C_{D_{EXT}}$</td>
<td>inlet external drag coefficient, based on $A_c$ (see fig. 1)</td>
</tr>
<tr>
<td>$C_{D_{I}}$</td>
<td>inlet external drag coefficient, based on $A_w$</td>
</tr>
<tr>
<td>$C_{D_{INT}}$</td>
<td>inlet interference drag coefficient, based on $A_c$</td>
</tr>
<tr>
<td>$C_{D_{NADC}}$</td>
<td>NADC predicted aft end drag [see eq. (39)]</td>
</tr>
<tr>
<td>$C_{D_{MB}, C_{D_{MB}}}$</td>
<td>boattail pressure drag coefficient based on $A_{MB}$ and $A_{P_{MB}}$, respectively [see eq. (26)]</td>
</tr>
<tr>
<td>$C_{D_{SP}}$</td>
<td>side-plate drag coefficient, based on $A_c$</td>
</tr>
<tr>
<td>$C_{D_{SPIIL}}$</td>
<td>spillage drag coefficient, based on $A_c$ (see fig. 1)</td>
</tr>
<tr>
<td>$C_{D_{W}}$</td>
<td>total drag coefficient (pressure, friction and base drag) on aft end of airplane configuration, based on $A_w$</td>
</tr>
<tr>
<td>$\Delta C_{D}$</td>
<td>jet-off to jet-on drag coefficient increment, based on $A_{SO}$ (fig. 57)</td>
</tr>
<tr>
<td>$\Delta C_{D}$</td>
<td>pressure drag coefficient increment used in ESIP drag correlation (see figs. 52 and 53)</td>
</tr>
<tr>
<td>$\Delta C_{DS}, \Delta C_{DE}$</td>
<td>boattail drag increments due to plume shape and entrainment, respectively, based on $A_m$</td>
</tr>
<tr>
<td>$\Delta D$</td>
<td>drag increment from design to operating pressure ratio [CALAC subsonic correlation, eq. (26)] or from jet-off to jet-on operation [CALAC supersonic correlation, eq. (27)]</td>
</tr>
<tr>
<td>$C_{LS}$</td>
<td>lip suction coefficient, based on $A_c$ (see fig. 1)</td>
</tr>
<tr>
<td>$C_p$</td>
<td>pressure coefficient, $(P - P_\infty)/q_\infty$</td>
</tr>
<tr>
<td>$C_T$</td>
<td>nozzle thrust coefficient, ratio of actual gross thrust to ideal thrust, $F_{ID}$</td>
</tr>
<tr>
<td>$D$</td>
<td>fuselage maximum diameter</td>
</tr>
<tr>
<td>$D_{add}$</td>
<td>additive drag, $\int_{\text{cowl lip}} (P - P_\infty)dA_p$</td>
</tr>
<tr>
<td>$D_m$</td>
<td>maximum fuselage equivalent diameter, $\sqrt{4A_m/\pi}$</td>
</tr>
<tr>
<td>$D_{MB}$</td>
<td>metric break equivalent diameter, $\sqrt{4A_{MB}/\pi}$</td>
</tr>
<tr>
<td>$ER$</td>
<td>nozzle expansion ratio, $A_e/A_t$</td>
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LIST OF SYMBOLS (Continued)

\( F_\text{ID} \)  
nozzle ideal gross thrust based on the isentropic expansion of actual mass flow to free-stream pressure

\( FR \)  
boattail fineness ratio, \( L_A/D_m \)

\( H \)  
height of nonaxisymmetric nozzle cross section

\( \bar{h} \)  
nondimensional height above cone surface (see fig. 107)

\( \text{IMS} \)  
integral mean slope: area-weighted average of the rate of change of nondimensional afterbody area distribution, based on \( A_m \) [see fig. 54(b)]

\( \text{IMS}_T \)  
truncated integral mean slope: IMS which is determined using a maximum allowable slope which is a function of Mach number (see fig. 53)

\( \text{IMSA} \)  
integral mean slope, based on \( A_{MB} \) [see fig. 54(a)]

\( K_{\text{add}} \)  
additive drag correction factor [see eq. (3)]

\( K_1 \)  
drag coefficient parameter used in similarity correlation of jet-off boattail pressure drag [see eq. (26)]

\( K_2 \)  
drag coefficient parameter – increment in drag from design pressure ratio to operating pressure ratio [see eq. (26)]

\( K_3 \)  
drag coefficient parameter – increment in drag from jet-off to jet-on operations [see eq. (27)]

\( L_A \)  
length of afterbody

\( L_N \)  
length of nozzle

\( \text{LE,TE} \)  
denotes wing leading and trailing edges, respectively (see figs. 112, 113)

\( M \)  
Mach number

\( M_L \)  
local Mach number

\( M_\infty \)  
free-stream Mach number

\( \text{MFR} \)  
mass flow ratio – mass flow entering the inlet ratioed to free-stream flow through \( A_c \)
### LIST OF SYMBOLS (Continued)

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<tr>
<td>NPR</td>
<td>nozzle pressure ratio - ratio of nozzle total pressure to free-stream static pressure at nozzle exit</td>
</tr>
<tr>
<td>NSPR</td>
<td>nozzle static pressure ratio, $p_e/p_\infty$</td>
</tr>
<tr>
<td>$P,p$</td>
<td>static pressure</td>
</tr>
<tr>
<td>$P_b$</td>
<td>base pressure</td>
</tr>
<tr>
<td>$P_e$</td>
<td>nozzle exit static pressure</td>
</tr>
<tr>
<td>$P_P/p_\infty$</td>
<td>ratio of local to free-stream pitot pressure</td>
</tr>
<tr>
<td>$P_{TL}/P_{T_\infty}$</td>
<td>ratio of local to free-stream total pressure</td>
</tr>
<tr>
<td>$P_t,p_t$</td>
<td>total pressure</td>
</tr>
<tr>
<td>$q$</td>
<td>dynamic pressure</td>
</tr>
<tr>
<td>$R$</td>
<td>maximum body radius (see fig. 88)</td>
</tr>
<tr>
<td>Re</td>
<td>Reynolds number</td>
</tr>
<tr>
<td>$r$</td>
<td>radial distance from body centerline</td>
</tr>
<tr>
<td>$r_{ce}$</td>
<td>radius of the cowl lip leading edge</td>
</tr>
<tr>
<td>$S/D$</td>
<td>nozzle spacing ratio - ratio of nozzle centerline to centerline distance to equivalent diameter of nozzle at exit station</td>
</tr>
<tr>
<td>$T$</td>
<td>temperature</td>
</tr>
<tr>
<td>$V_\infty$</td>
<td>free-stream velocity</td>
</tr>
<tr>
<td>$\frac{V}{V_\infty}$</td>
<td>local sidewash velocity (see Table XV)</td>
</tr>
<tr>
<td>$W$</td>
<td>width of nonaxisymmetric nozzle cross section</td>
</tr>
<tr>
<td>$\frac{W}{V_\infty}$</td>
<td>local upwash velocity (see Table XV)</td>
</tr>
<tr>
<td>$X$</td>
<td>axial distance along body or afterbody</td>
</tr>
<tr>
<td>$Y$</td>
<td>lateral distance from centerline of body (see fig. 97)</td>
</tr>
<tr>
<td>$Z$</td>
<td>distance below centerline of body (see fig. 97)</td>
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\[ \alpha \]
angle of attack

\[ \alpha_i(i=1,11) \]
covariance coefficients, used in NADC extension of CALAC afterbody drag method [see eq. (39)]

\[ \alpha_L \]
local upwash angle (see Table XV)

\[ \beta \]
age of sideslip; also boattail trailing edge angle

\[ \beta_L \]
local sideslip angle (see Table XV)

\[ \gamma \]
ratio of specific heats

\[ \epsilon_L \]
local total-flow angle (see fig. 88)

\[ \theta \]
boundary-layer momentum thickness, or circumferential angle (see fig. 88)

\[ \theta_B \]
mean boattail angle - mean angle over a distance corresponding to one-third of nozzle exit radius

\[ \theta_{\text{BTM}} \]
maximum nozzle boattail angle

\[ \mu \]
viscosity

\[ \rho \]
density

\[ \phi, \phi_2, \phi_B \]
velocity potentials associated with transonic equivalence rule (see fig. 110)

\[ \phi_L \]
local roll angle (see fig. 88)

Subscripts

\[ B \]
referring to a basic configuration

\[ \text{corr} \]
pertaining to the corrected value of nozzle pressure ratio [see eq. (12)]

\[ e \]
referring to the nozzle exit

\[ \text{eff} \]
effective

\[ \text{f.e.} \]
pertaining to fully expanded flow in a nozzle

\[ j \]
referring to conditions in a jet
LIST OF SYMBOLS (Concluded)

L  pertains to local conditions
l  pertaining to the cowl lip plane
NB referring to the metric break station
m referring to the maximum cross-sectional-area station
p referring to projected frontal area
ref pertaining to the reference value
t referring to the nozzle throat station
∞ free stream
2. INLET/AIRFRAME INTERACTION EFFECTS

In Table I, reports dealing with interactions of the inlet and airframe are categorized by the specific effect they treat and by report type, as described below.

2.1 Report Types

Review Papers attempt to enhance understanding of a given effect by analyzing in depth data first presented elsewhere, assessing the coherence of data from several sources, judging the applicability of existing prediction methods by drawing comparisons with data, etc. In short, they attempt to synthesize and analyze previously available information. Primary Data Sources are reports presenting detailed experimental data. What is meant by sources of Empirical Prediction Methods and sources of Theoretical Prediction Methods is clear once the (somewhat arbitrary) distinction is made between the approaches: empirical methods are based primarily on correlations of experimental data, and theoretical methods are taken to be solutions (finite difference, method of characteristics, integral, etc.), of the appropriate equations of motion. Theoretical methods may include combinations of several methods modified by and patched together with empirical considerations when the primary emphasis is on solution of the governing flow equations.

2.2 Definition of Effects

The inlet/airframe interaction effects appearing in Table I are defined in the following outline. In general, a reference is categorized under the lowest subdivision of an effect that is appropriate. If, however, the information in a reference is not sufficiently detailed, the reference is classified under a higher order heading. For example, if drag data were taken for an inlet that had an operating boundary-layer-control system and the drag associated with this boundary-layer control was not separated from the drag associated with mass spillage around the
inlet, the reference is classified as one dealing with inlet drag, not with external or boundary-layer control drag.

I. Airframe Effects on Inlet "Free-Stream" Conditions

The flow entering an installed inlet may differ from the free stream in Mach number, angularity and stagnation pressure. In general, these differences may be ascribed to:

A. Forebody effects, for close-coupled inlets mounted forward of the wing, or

B. Forebody-wing effects, for under-wing-mounted inlets in the wing's compression field. References dealing with combined forebody-wing effects on an inlet are classified in this category.

II. Inlet Drag

The drag attributable to an installed inlet operating at an arbitrary Mach number and mass flow ratio (MFR) is broken down as follows:

A. External Drag \( (C_{D_{EXT}}) \) is defined to be the sum of the drag on the stagnation streamline, additive drag \( (C_{D_A}) \), the drag on the external surfaces of the cowl \( (C_{D_C}) \) and for two-dimensional inlets, the drag on the external surfaces of the side plates \( (C_{D_{SP}}) \). \( C_{D_C} \) and \( C_{D_{SP}} \) are usually taken to be integrated pressure drag although some investigators include skin friction. The relationships of these quantities with the commonly used terms spillage drag and cowl-lip suction are schematically illustrated in Figure 1 for an axisymmetric inlet (thereby avoiding the complication of the side-plate drag). Spillage drag \( (C_{D_{SPIL}}) \) is shown to be the change in external drag as mass flow ratio is decreased from the reference value appropriate for the Mach
number in question. Cowl lip suction ($C_{LS}$) is the decrease in
 cowl drag ($C_D$) as the mass flow ratio is decreased from this
 reference value. If a reference treats any of these inter-
 related terms, it is classified under external drag. Exactly
 which drag quantity is presented is described in a later
 section.

B. Bypass drag is defined to be that portion of inlet drag associ-
 ated with the air passing through the bypass system. A bypass
 system provides an alternative to spillage for engine-inlet flow
 matching. Bypass drag results from the momentum loss of the by-
 passed air and the drag of any external surfaces installed for
 purposes of directing the bypassed air's exhaust.

C. Boundary-layer bleed drag is defined analogously to bypass drag
 except it is associated with the air removed from the inlet by
 porous surfaces or discrete holes or slots for the purpose of
 boundary-layer control.

D. Interference drag is the drag increment on airframe surfaces
 adjacent to the inlet that is a function of inlet mass flow
 ratio.

E. Boundary-layer diverter drag is the drag of the device provided
 to prevent the boundary layer developed on upstream surfaces
 (forebody or wing) from entering the inlet.

III. Inlet Internal Performance

The inlet exists to provide high-quality airflow to the engine
 throughout the flight envelope. This function must be, of course,
 accomplished while maintaining acceptable levels of drag and installed
 weight. The measures generally used for the quality of the internal
 airflow are.
A. Total pressure recovery; that is, how much of the free-stream total pressure is maintained through the diffusion process to the compressor face. Recovery and the level of distortion (discussed below) are enhanced by implementation of a boundary-layer bleed system. However, any improvement must be in the proper balance with the added complexity, weight and drag such a system entails.

B. Stable mass flow range. The inlet must be capable of providing the range of mass flows demanded by the engine. The low-flow limit for an inlet is set by either an engine maximum-allowable distortion criterion or the onset of an instability known as inlet "buzz". The high-flow limit is determined by engine distortion tolerance for inlet supercritical operation.

C. Distortion. There exists several definitions of terms used to quantify the nonuniformity (both stationary and dynamic) of the airflow in the inlet at the compressor face location. This measure of inlet performance is important because an engine will typically have limits specified in terms of one or more of these criteria above which stable operation is impossible. Even below this level, engine performance may be adversely impacted by some finite amount of distortion.

IV. Inlet Lift and Moment Effects

The aerodynamic forces acting on an inlet system can result in lift, pitching, yawing, and rolling moments as well as drag. The effects considered here are:

A. Inlet lift and moments due to airflow spillage.

B. Inlet lift and moments due to bypassed air.
C. Inlet lift and moments due to boundary-layer bleed flow.

D. Interference lift and moments. Analogously to interference drag, these are the increments on surfaces adjacent to the inlet and are functions of mass flow ratio.

V. Inlet Weight

While not strictly an interaction, this effect is considered because it is an important factor to be considered in evaluating inlet concepts. Sophisticated variable geometry inlets may enjoy benefits with respect to the previously listed effects, but if the accompanying weight penalty is too severe there is no net advantage to the integrated airframe/propulsion system.

3. AFTERBODY/AIRFRAME INTERACTION EFFECTS

In Table II, reports dealing with interactions of the afterbody (including the engine exhaust nozzles) with the airframe are categorized by the specific effects they treat and by report type in a manner similar to that of the inlet/airframe interaction effects in Table I.

3.1 Report Types

The report types are Review Papers, Primary Data Sources and Empirical and Theoretical Prediction Methods as in Table I.

3.2 Definition of Effects

The afterbody/airframe interaction effects appearing in Table II are defined in the following outline. The references generally fall within one of two general categories, those applicable to single-engine configurations and those applicable to twin-engine configurations. Within
those two categories the references are classified as dealing with the effects of certain specific parameters. Some parameters such as nozzle pressure ratio are common to both types of configurations.

I. Single-Engine Configurations

The afterbody effects considered for single-engine configurations are associated with the effect of various parameters on the flow field around the afterbody, including the nozzle boattails.

A. Exhaust-plume effects are effects produced by the exhaust jet plume. The shape of the plume can create upstream disturbances which affect the flow field over the afterbody. Also, the velocity in the jet being higher than that of the surrounding flow results in flow entrainment which can also alter boattail flow fields. The specific effects considered are:

1. Nozzle-pressure-ratio effects are effects produced by the exhaust jet plume that can be attributed to the nozzle pressure ratio for a specific nozzle type.

2. Exhaust-temperature effects are effects produced by the exhaust jet plume that can be attributed to the exhaust jet temperature.

3. Miscellaneous other parameters or conditions may be capable of producing exhaust-plume effects. Included in this category are various methods of simulating an exhaust plume using solid stings, normal jet simulators, and other techniques.

B. Nozzle-type effects are the effects on afterbody drag which are attributable to the type of nozzle used. Any nozzle which can provide for efficient expansion of the exhaust gases at high
pressure ratio can be designed to give good performance at a specified design condition. Thus, an optimum nozzle can be chosen for that design condition. However, the merit of any particular nozzle type frequently depends upon its ability to operate efficiently at conditions other than design. This requires comparison of the relative merits of different nozzles with regard to afterbody drag, nozzle weight and complexity and overall mission requirements. Papers in this category provide information toward this comparison.

C. Base-flow effects are effects associated with the existence of a finite base area as may occur with a truncated nozzle plug or when the afterbody boattail terminates with a cross-sectional area larger than the nozzle exit area.

D. Empennage interference effects are effects produced by the alterations of the flow field around an isolated afterbody when horizontal and/or vertical stabilizer surfaces or other external supporting structure are installed. An example of such an effect is the increase in afterbody pressure drag noted in reference 46 for twin vertical stabilizers on a twin-engine configuration compared with the drag for a single vertical stabilizer on the same configuration. The single vertical stabilizer exerts less influence on the afterbody flow field due to its location on the plane of symmetry.

E. Mach-number effects are the effects of compressibility on the afterbody flow fields. References in this category are primarily those which provide specific information about the effects of compressibility on afterbody drag for complete configurations. Other references in other categories may also provide such information with regard to partial configurations.
F. Angle-of-attack effects are changes in the afterbody drag due to changes in angle of attack.

G. In this category are listed references which provide information on the effects of Reynolds number over a wide range of Reynolds number. Reynolds-number effects are important for scaling wind tunnel test results to full-scale flight speeds.

H. Boattail shape effects are effects that can be specifically attributed to the particular shape of a boattail.

II. Twin-Engine Configurations

A. Exhaust-plume effects. References for this effect are those which apply specifically to twin-engine configurations.

B. Nozzle-type effects. As for single-engine configurations, references in this category deal with the afterbody-drag effects which are attributable to the type of nozzle used.

C. Nozzle-spacing effects arise due to the mutual interference between the two nacelles/nozzles of the twin-engine afterbody.

D. Nozzle-location effects are associated with the location of the nozzle exit relative to the terminus of the fuselage interfairing on the empennage.

E. Fuselage-interfairing effects are associated with various shapes of interfairing, and the manner in which they are applied to the twin nacelle/nozzle combination.

F. Empennage interference effects are associated with horizontal and/or vertical stabilizing surfaces or other supporting structures.
G. Mach-number effects are the effects of compressibility on twin-engine configurations.

H. Angle-of-attack effects.

I. Reynolds-number effects.

J. Boattail-shape effects.

4. SPECIAL TOPICS

In surveying the literature assembled in the course of this work, a few topics emerged which are of general interest to those concerned with the subject of airframe/propulsion system interactions. References which touched on these topics are identified in this section.

4.1 Thrust-Drag Bookkeeping

To assess the net propulsive force acting on an airplane, it is necessary to account for each thrust and drag component once and only once. The failure to comply in a straightforward manner with this seemingly obvious requirement has, however, resulted in needless confusion in more than one aircraft development program. Several references are listed in Table III which discuss the necessity of this procedure and which describe detailed schemes to account for each thrust and drag item. Reports from the reference lists for Inlet/Airframe Interaction and Afterbody/Airframe Interaction are included.

4.2 Test Techniques

The development of experimental procedures to allow accurate measurement of interaction affects is discussed in several references which also appear in Table III. Sophisticated techniques are required to allow the separation of the various interaction effects while
minimizing the influence of undesirable interference effects brought about by partial, subscale model testing. Scale effects, simulation of the exhaust plume in afterbody testing, and modeling viscous effects are among the topics discussed.

4.3 Wind Tunnel to Flight Comparisons

An important means of assessing the success of test techniques is the comparison of wind-tunnel data to data obtained from actual flight vehicles. However, acquisition of accurate data for interference effects is particularly difficult and expensive in flight, so there are not a large number of such comparisons. Those comparisons we have uncovered appear in Table III.

4.4 Boundary Layer Methods

Many of the prediction methods in the data base use a particular boundary layer calculation method to account for viscous effects. The boundary layer methods so used are also listed in Table III.

4.5 Inviscid Flow Methods

Methods of calculating the inviscid flow about various kinds of bodies are also listed in Table III.

5. DATA BASE FOR INLET EXTERNAL DRAG AND THE EVALUATION OF ITS ADEQUACY FOR PRELIMINARY DESIGN

As previously shown in figure 1, inlet external drag* consists of the sum of integrated cowl pressure drag and additive drag:

\[ C_{D_{EXT}} = C_{D_C} + C_{D_A} \]  

*For a nonaxisymmetric inlet, the drag on surfaces such as sideplates is sometimes treated independently or alternatively is included as part of \( C_{D_C} \). The latter usage is adopted here for simplicity.
This relation can be expressed in a number of alternative ways, including

\[ C_{D_{EXT}} = C_{D_A} - C_{LS} + \left( C_{D_C} \right)_{ref} \]  \hspace{1cm} (1b)

\[ = \left( C_{D_A} \right)_{ref} + \left( C_{D_C} \right)_{ref} + \int_{MFR_{ref}}^{MFR} \frac{\partial C_{D_{EXT}}}{\partial (MFR)} \, d(MFR) \]  \hspace{1cm} (1c)

\[ = \left( C_{D_A} \right)_{ref} + \left( C_{D_C} \right)_{ref} + K_{add} \left[ C_{D_A} - \left( C_{D_A} \right)_{ref} \right] \]  \hspace{1cm} (1d)

by incorporating the definitions for spillage drag

\[ C_{D_{SPIL}} = C_{D_{EXT}} - \left( C_{D_{EXT}} \right)_{ref} \]  \hspace{1cm} (2)

and the \( K_{add} \) factor

\[ K_{add} = \frac{C_{D_{SPIL}}}{C_{D_A} - \left( C_{D_A} \right)_{ref}} \]  \hspace{1cm} (3)

The data base for inlet external drag consists of experimentally determined values and theoretical or empirical prediction techniques for any of the quantities in Eqs. (1-3). In the following sections, we first present the applicable experimental data and describe methods ("Partial Prediction Methods") which result in the prediction of a component of \( C_{D_{EXT}} \) (e.g., \( C_{LS} \) or \( \frac{\partial C_{D_{EXT}}}{\partial (MFR)} \)). We then present methods which result in the prediction of the detailed flow field about an inlet ("Flow-field Methods"), thereby allowing calculation of \( C_{D_{EXT}} \) by simple quadrature. The requirements placed on the data base for use in preliminary design
are discussed, and finally, the data base is evaluated with respect to these requirements.

5.1 Presentation of the Data Base

5.1.1 Experimental data

The experimental data base for inlet external drag is presented in Table IV. The types of inlets and ranges of test conditions are described for each set of data. The Primary Data Sources from Table I are included which present detailed experimental data for inlet external drag or a component (for example, additive drag or cowl drag). Also included are the data sources which are listed under Inlet Drag in Table I because the data presented are not further broken down into components.

Each inlet in Table IV is described by a string of characters, some of them subscripted, which represents the main characteristics of the inlet. These descriptors, and the characteristics they represent, are listed in Table V. Absence of a particular descriptor for a given inlet means that characteristic is not applicable to that inlet, or, as in the case of the design Mach number, that it was not reported. The items in Table V are self-explanatory, with the exception of the conventions for sideplate shape. These are schematically illustrated in the following sketches of the upstream portion of two-dimensional, external-compression inlets with two fixed horizontal ramps and a design Mach number of 2.4. Note that the absence of the descriptors for a bypass system and a boundary-layer-control system implies that these systems do not exist in these inlets.

\[ E_r H_x S_r S_{2.4} \]
Other examples of this system are: the string $M_C^3 v^2 B P2.5$ which describes an axisymmetric mixed-compression inlet with three conical external compression surfaces with two variable, a boundary-layer-control system consisting of porous elements, an operational bypass system, and a design Mach number of 2.5; and the string $P_0$ which describes a pitot inlet whose capture area is neither circular nor rectangular.

The column in Table IV after the inlet characterization identifies the drag component(s) presented for that inlet in the reference in question. The next column shows the test configuration; that is, whether the inlet was tested alone or in the presence of a forebody, mounted on an entire fuselage, etc. Next come a series of columns containing the ranges of Mach number, unit Reynolds number, angle of attack and angle of sideslip covered for each inlet. Finally, the remaining columns show whether sufficient data exist to evaluate the separate effects of cowl
shape, sideplate shape, ramp-angle or capture area variations on the drag component presented. Special circumstances are identified in the notes following Table IV.

5.1.2 Partial prediction methods

These methods, usually largely empirical, result in predictions for one or more of the components of $C_{D_{EXT}}$ indicated in Eq. (1). Prediction methods of this type have been categorized in Table VI by the inlet type and the speed range (subsonic, transonic, supersonic) to which they apply. Inlet type in this table refers to a pitot inlet of either axisymmetric or two-dimensional shape, or an inlet which is either of two-dimensional or axisymmetric geometry with an external compression surface. Application of prediction methods of this class to an inlet of more complicated shape (e.g., "kidney", "chin", half-axisymmetric) must be done on an ad hoc basis; the methods are, in general, derived for the simple inlet shapes just described. In Table VI, if a reference is listed as predicting $[C_{DA}]_{ref}$ or $[C_{DC}]_{ref}$, then only the reference value of that quantity is predicted by the method in that reference. On the other hand, if a reference is listed as predicting $C_{DA}$, then it is presumably valid at any mass flow (including the reference value).

A very brief description of some of the major features of each method shown in Table VI is contained in Table VII. For the convenience of the reader, the principal author's name is shown in parenthesis after the reference number. The information presented in Table VII does not, of course, allow for detailed understanding of the assumptions and limitations associated with each approach; for this purpose, the references themselves must be consulted. However, the general approach of each method is described, and important assumptions (such as independence of cowl lip geometry or a reference mass flow ratio of unity in lip-suction prediction methods) are shown. This allows essentially similar methods to be identified, a feature that is utilized in the comparison-to-data and evaluation processes to follow. For example, it is seen in Table VII
that a number of references (21, 71, 102, 106) involve the "pitot inlet analogy", a method based on the observation made in reference 59 that for pitot inlets, if the additive drag occurring behind the normal shock for subcritical operation in a supersonic free stream is figured in a particular way, lip suction is implicitly accounted for and external drag may be predicted very simply.

It should be noted that the Partial Prediction Methods are in general derived for use at zero angles of attack and sideslip in a uniform free stream. Adaptation to more complicated situations is often required, of course, and schemes for accomplishing this exist (see, for example, ref. 4), but the approximations involved are over and above those fundamental to the methods themselves. The comparisons to data of predictions using these methods shown in a later section are largely for the case of a uniform free stream at zero angles of attack and sideslip because published comparisons have not been found for the more complicated situations.

5.1.3 Flow-field methods

A complete description of the flow field in the vicinity of the inlet is provided by these methods; external drag and its components are available by straightforward integration of the calculated pressure distribution. Methods of this type are shown in Table VIII, where in a fashion similar to Table VI, they are categorized by general approach and the geometry and speed range to which they apply. Major features of the methods are listed in Table IX.

A few general comments about these methods are in order. Firstly, we obviously have not acquired every implementation of each general approach listed in Table VIII, but it is felt that all the important general approaches are represented. Secondly, all of the methods listed as applicable to 2-D geometries can handle non-zero angle of attack, but only two of the methods for other geometries (refs. 215 and 405) can do
so. This is so because only these methods (intended for use with subsonic design-point inlets) are designed to handle a three-dimensional flow field. Calculation of three-dimensional inlet flow fields is only just beginning, and descriptions of this work are very scarce in the open literature. Finally, all of the methods except that of reference 274 are inviscid, whereas in reference 274 the inviscid streamtube curvature method is coupled to a boundary layer analysis. This is obviously an option available for use with all of the other methods, but this refinement is a separate problem not discussed in the published accounts nor in this report. Analysis of the boundary layer methods available for use in inlet calculations and an evaluation of their success is clearly beyond the scope of the present work.

5.2 Evaluation of the Adequacy of the Data Base for Preliminary Design

5.2.1 Requirements

The functional dependence of external drag on geometry and average local flow conditions can be represented as

\[ C_{D_{EXT}} = f(\text{Inlet Geometry}, M_L, \alpha_L, \beta_L, \text{Re}_L, \text{MFR}) \] (4)

where the specification of Inlet Geometry includes such general features as inlet type (pitot, external- or mixed-compression), inlet shape (two-dimensional, axisymmetric, quarter round, etc.), and design Mach number as well as details such as required capture area, compression surface angles, and cowl and sideplate (if applicable) external profile and leading-edge radius. Even this complicated relationship has been simplified if the inlet is located in a nonuniform flow field (caused by forebody precompression, for example) by the assumption that the external drag may be related to the average local conditions.
In the preliminary design process, external drag must be evaluated for various candidate inlet concepts, usually not defined in total detail, to allow the most promising of them to be selected for continued development in depth. The criterion for selection is some ranking in terms of overall best performance for a specified mission where drag level is traded off against internal performance, complexity and weight. To be useful for preliminary design, a data base must therefore satisfy three requirements:

1. Breadth. The range of applicability of the data or methods must be sufficiently wide to allow rational evaluation of a variety of inlet designs.

2. Ease of application. The time and expense required must be consistent with consideration of a large number of candidates.

3. Accuracy. The methods or data used must accurately reflect differences in the candidates considered.

In the following sections, the experimental data and prediction methods that exist for inlet external drag are evaluated with respect to these requirements.* First, however, a few additional remarks are in order about the level of accuracy required. Even at the preliminary design stage, exact solutions for the various effects would be useful if they were compatible with allowable levels of cost and effort. This is essentially never possible, however. The usual situation is that these constraints force the designer towards the use of methods on the other end of the scale; that is, methods that barely possess the required accuracy. It is also true that no simple, general statement of the accuracy required can be made. That is, it is not possible to say that the methods or data must be accurate to within + X percent or + Y drag counts. This is so because the minimum drag increment that causes a

*An exception: the accuracy of all the assembled experimental data is not evaluated.
meaningful change in airplane performance is very much a function of the
detailed nature of the application (mission) considered.

Some calculations of the sensitivity of airplane performance to inlet
external drag for specific applications exist in the assembled litera-
ture. References 29, 68, 94, 103, 114, and 115 address this question in
different ways for various configurations of a supersonic transport. For
example, in reference 114, for a supersonic transport powered by after-
burning turbojet engines, a change in inlet-external-drag coefficient of
0.01 (-10 percent) results in a change in range of about 41 n.mi. (-1
percent). References 115 assesses this sensitivity for a similar
application in a different way. At Mach 1.2, an increase of 0.06 (33
percent) in $C_{D,ext}$ decreased the payload capacity of an SST with 3,500
n.mi. range by two passengers (-1 percent). This leverage can be
considerably greater for supersonic tactical aircraft flying so-called
"mixed missions" (combinations of supersonic and subsonic portions).
References 2, 8, 10, 27, 29, 365, and 380 address this point. In
reference 8, for example, a 30 percent increase in $C_{D,ext}$ is shown to
reduce mission range by about 10 percent if the drag increase occurs in a
transonic portion of the mission. In reference 2, differences in $C_{D,ext}$
for candidate inlet designs for such an aircraft are shown to result in
differences in range of up to 200 n.mi. This magnified sensitivity is
caused by the fact that the inlet drag is a considerable portion of the
total airplane drag under some situations (up to 30 percent at cruise;
reference 27). A similar strong dependence is demonstrated for the B-1
in reference 380. Thus, while no general statement can be made for the
required accuracy of a preliminary design method, it is clear that inlet
external drag exerts a powerful influence on airplane performance. If
differences caused by different inlet configurations are not adequately
represented, proper decisions for further development effort cannot be
made.
5.2.2 Experimental data

Given the complicated functional dependence of inlet external drag on the large number of independent variables and parameters represented in Eq. (4), it is obvious that experimental treatment can never be "complete" or adequate for design in itself. All of the possible combinations of the controlling quantities cannot possibly be experimentally studied. What is required to fill in the gaps is an interpolative and extrapolative procedure. This procedure can be a simple curve-fit to data or it can consist of an assumed Taylor's series representation with the "partial derivatives" in this series being evaluated from the data. The notion of such a series expansion is simple enough to be useful only if the mixed and higher-order derivatives are negligible.

Empirically based Partial Prediction Methods are a means of interpolating in and extrapolating the existing experimental data base and are discussed in the next section. In this section, the adequacy for preliminary design of the "Taylor's series" representation is briefly reviewed. It is shown that external drag is sufficiently configuration dependent (that is, it depends on geometric detail to such a degree) that the Taylor's series representation is useful only for very limited excursions from the original data.

The experimental data base (Table IV) represents a formidable amount of information, even if those investigations where external drag is not explicitly available are excluded. However, most of the remaining investigations represent at most small perturbations around a specific design point, since they were a part of a particular airplane development program or were aimed at the optimization of a specific inlet. That is, the great majority of the references listed in Table IV report studies that are too specialized and too unsystematic to be useful as the basis for a design methodology of any generality. There do exist, however, a few serious attempts at systematic variations of the controlling quantities of sufficient scope to allow evaluation of the practical-
ity of the "Taylor's series" approach. The investigations of references 33, 107 and the combined investigation of 52 and 108 as reviewed in 27 are useful in this regard.

The change in $C_{D_{EXT}}$ as the cowl contour is changed from some reference shape (with all other variables fixed) is one of the many "partial derivatives" that would have to be ascertained in the Taylor's series approach. Figures 2 through 4 are extracted from those references and show this dependence for the extremely simple case of a one-external-compression-surface inlet in a uniform free stream at zero angle of attack and sideslip.

Figure 2 shows the variation of $C_{D_{EXT}}$ with mass flow ratio for a two-dimensional inlet with six different sharp-leading-edge cowls at two Mach numbers. Similarly, each part of Figure 3 compares external-drag variations for two members of a family of elliptical cowls with constant leading-edge radius. These data are from reference 107 and are also for a single ramp, two-dimensional external-compression inlet. Finally, Figure 4 shows $C_{D_{EXT}}$ versus MFR for a conical and an elliptical cowl of the same leading-edge radius at three different transonic Mach numbers from reference 27 for an axisymmetric, single-cone external-compression inlet.

Examination of these figures reveals that the affect of changing cowl shape on $C_{D_{EXT}}$ is substantial and that the quantitative effect of going from one cowl to another is different in magnitude (and in some cases in sign) depending on the value of the Mach number and the mass flow ratio. Effects of other variables (e.g., angle of attack, ramp angle) on the dependence of $C_{D_{EXT}}$ on cowl shape are not shown but cannot be assumed negligible. In short, this dependence is sufficiently complicated that the simple Taylor's series approach is not useful except for very small excursions from a known data point. It follows then, that the experimental portion of the data base, taken by itself, is adequate for preliminary design purposes only if the inlets under consideration are
substantially the same and are to operate in essentially the same conditions as those for which data exist.

5.2.3 Partial prediction methods

To synthesize external drag with acceptable accuracy from the parts utilized in these methods (see Eqs. (1)-(3)), each of the parts must obviously be predicted accurately (omitting the possibility of compensating errors from further consideration). While some of the elements of these methods are straightforward, the critical one is that which deals with the change in cowl drag with reduced mass flow. This element is most often treated using $C_{LS}$ or $K_{add}$ factors, or by means of the "pitot inlet analogy" (see Table VII). The success of these approaches will be ascertained by examination of comparison of predictions made using these methods to experimental data. (Some of these comparisons are presented here for the first time, but for the most part, we must rely on comparisons found in the assembled literature.) Additionally, the prediction of additive drag will be briefly examined.

5.2.3.1 Cowl-lip suction ($C_{LS}$ factors)

Five of the references accumulated propose empirical correlations for this effect: 10, 11, 32, 39, and 86. The treatment in reference 86 is one-dimensional and because it is specified by the author to be inapplicable to realistic lip shapes, it is not considered further. Three of the remaining methods, 10, 11, and 39, have a reference mass flow ratio of unity. This presents special problems and these methods will be discussed together.

Methods with $MFR_{ref} = 1$. Cowl lip suction in these methods is defined as

$$C_{LS} = C_{DC} (MFR = 1) - C_{DC}$$  (5)
In reference 10, correlations for the quantity $C_{LS}/C_{DA}$ are presented as a function of (local) free-stream Mach number, with $C_{DA}$ as a parameter. Two correlations are given, one for two-dimensional inlets, the other for axisymmetric inlets. The two-dimensional correlation is assumed to include the effects of suction on the lips of the sideplates. The data used to generate these correlations are neither identified nor shown. Implicit in this approach is the idea that inlets exhibiting the same levels of additive drag at a given Mach number will have the same $C_{LS}$. That is, all the dependence of $C_{LS}$ on detailed geometry is in this method contained in $C_{DA}$. A number of investigators, however, have shown that additive drag is in fact relatively insensitive to variations in cowl geometry, while the variation in cowl drag with mass flow ratio is not. Based on these considerations, one would not expect particularly good agreement of this method with data. Nonetheless, it is very convenient to apply.

This point is investigated in figure 5 by comparing $C_{LS}/C_{DA}$ as a function of $C_{DA}$ for two different Mach numbers from the correlations for two-dimensional inlets in reference 10 with data for two different cowl designs from reference 33 (uniform free stream, angle of attack = angle of sideslip = 0). These data were derived from plots of $C_{D_{EXT}}$ and $C_{DC} + C_{DSP}$ using Eqs. (1a) and (5). Agreement of the correlation with the data is seen to be poor in both level and trend. As expected, the effect of cowl geometry is clearly not adequately represented by its effect on additive drag.

In the process of reducing the data in reference 33 to the form shown in figure 5, a major disadvantage of the representation of Eq. (5) became apparent. That disadvantage is the choice of unity as the reference mass flow ratio. Because inlets with external compression surfaces cannot in general achieve $MFR = 1$, extrapolations of data for $C_{DC}$ ($C_{DC} + C_{DSP}$ for two-dimensional inlets) are required to allow evaluation of $C_{LS}$. The extrapolations required for one of the Mach numbers in figure 5 are shown in figure 6. The uncertainties introduced into $C_{LS}$ by
this process can be substantial, making comparisons with data extremely difficult to interpret, or to put it another way, making correlations formulated in this way inaccurate in principle.

The correlation method of reference 11 is an improvement on that of reference 10 in that the effects of two geometric quantities are explicitly included. These quantities (defined in detail in reference 11) are effective cowl lip slope and capture-area ratio. \( \frac{C_{LS}}{C_{DA}} \) is given as a function of these quantities and the (local) free-stream Mach number. However, because the reference mass flow ratio is unity for this method, the preceding remarks apply here as well and comparisons with data become moot. Similar comments apply to the method of reference 39, where \( \frac{C_{LS}}{C_{DA}} \) is a function of cowl lip angle, camber, leading-edge radius, local free-stream Mach number and \( C_{DA} \).

A Method with MFR \( \neq 1 \).- The difficulties resulting from the use of a reference mass flow ratio of unity are avoided by Osmon (ref. 32). This method is applicable to two-dimensional inlets and employs curve-fits of the lip-suction data from reference 33. The independent variable is a parameter depending on mass flow ratio, throat Mach number, capture-to-throat area ratio and the contraction ratio required to decelerate a reference flow to sonic velocity isentropically \( (A/A^*) \). For subsonic flows, the reference flow is at the free-stream Mach number; for supersonic flows, \( A/A^* \) is evaluated at the Mach number to which the flow is assumed to expand at the end of the cowl area change. The reference value of the independent variable (where lip suction is zero) is taken to be 0.8 for all Mach numbers. For subsonic, isentropic inlet flows, this implies that the reference state is that for which the throat Mach number is approximately 0.82. No such simple statement of the reference condition can be made for supersonic flows.

The dependent variable consists of the cowl lip-suction coefficient times effective thickness-to-chord ratios for the cowl and the ramp (each ratio raised to an exponent which is a function of free-stream Mach
number), times the ratio of capture area to maximum projected inlet area, also raised to a Mach-number-dependent exponent. The curves relating the dependent and independent variables are also parametrized on Mach number. Bands are displayed on these mean curves to show the limits within which about 95 percent of the data from reference 33 fall. This scatter represents, among other things, the effects of different sideplate geometry which is not explicitly treated in this approach.

This method is applied to a two-dimensional inlet with three different cowl shapes and two different sideplate shapes (ref. 107) in figure 7. The comparisons are for a single external ramp (angle = 5°) at zero angle of attack for a free-stream Mach number of 0.8. The cowl drag measurements shown were derived in a uniform free stream and are normalized on maximum projected inlet area. The data in figure 7(a) are for an elliptical cowl with a leading-edge radius which is ~1 percent of the capture height and with full rectangular sideplates (see ref. 107 for details). The reference mass flow ratio shown is calculated using the criterion in reference 32 and is in excellent agreement with the maximum mass flow ratio from the data. Cowl-drag coefficients were calculated at two mass flow ratios by the method of reference 32 by applying the lip suction from that method to the extrapolated experimental cowl-drag coefficient at the reference mass flow ratio. Note that the extrapolation required by the reference mass flow ratio of this method is of a much more reasonable extent than for the methods previously mentioned with MFR_ref = 1. The predicted cowl drag at the high mass flow ratio is in excellent agreement with the data, but this is mostly due to the fact that a very small amount of lip suction is involved and the experimental reference cowl drag was used. At the lower mass flow ratio, where there is more lip suction, the agreement is not as good. A band is shown on the predicted value, reflecting the bands on the correlation bounding ~95 percent of the data from which it was derived. At this low mass flow ratio, the correlation considerably underpredicts the lip suction.
A possible contributing factor to this underprediction is the fact that the data from reference 107 came from a model with full rectangular sideplates which would essentially eliminate sidespill, enhancing the suction effects on the cowl. Although reference 33 includes some data for full rectangular sideplates, it is not clear if those geometries are included in the correlation of reference 32 (indeed it is not clear if the correlated data is for suction on the lips of the cowl or on the lips of the cowl and sideplates). If the correlation only includes the triangular sideplate geometries of reference 33, and if it is for lip suction on the cowl only, an underprediction for lip suction would be expected for the case of figure 7(a).

A portion of this uncertainty is eliminated for the case of figure 7(b), where the data are for the same cowl as figure 7(a), but where triangular sideplates were used. Note that the lip suction prediction, which differs from that for figure 7(a) only by being applied to a different reference cowl drag, is still less than that measured, although at the low mass flow ratio the agreement is improved.

Figure 7(c) shows the agreement for a cowl of the same shape as in figure 7(a), but for an increased leading-edge radius (2 percent of the capture height). The comparison is again for the full rectangular sideplates because that is the only case for which data exist in reference 107. The agreement is essentially the same as in figure 7(a) and the same comments apply. Figure 7(d) is for a cowl with the same leading-edge radius as figure 7(c), but of different shape (although it is still elliptical, the point of tangency with the leading-edge circle is different). Agreement is improved in figure 7(d). Although the data show a considerable affect of the detailed cowl shape, the correlation does not: the lip-suction predictions for the cases of figures 7(c) and (d) are essentially the same.
5.2.3.2 The $K_{\text{add}}$ factor

Another means of accounting for the change in cowl drag with varying MFR is the $K_{\text{add}}$ factor. Use of this factor is advantageous if some freedom from geometric detail is gained by presentation of spillage drag data in this form; that is, if $K_{\text{add}}$ is a correlating parameter which achieves some "collapse" of the original data into a more generalized form. Note that such generality of $K_{\text{add}}$ would require differences in spillage drag to be reflected in additive drag, just as in the lip suction methods discussed in the previous section. If, on the other hand, a different $K_{\text{add}}$ is required for each inlet geometry, no real advantage exists for this approach as compared to using the original spillage drag data. This point is examined using the primary source for $K_{\text{add}}$ factors, the previously mentioned systematic investigation of reference 33.

In this investigation, $K_{\text{add}}$ was evaluated as in Eq. (3), using experimentally determined spillage drag and the additive drag increment as calculated using the theory presented in reference 33. Data are shown at six Mach numbers for external-compression, two-dimensional, two-ramp inlets with various combinations of ramp angles, four different sideplate shapes and six cowl designs, all measured in a uniform free stream at zero angles of attack and sideslip. Some of the $K_{\text{add}}$ factors determined are shown as functions of mass flow ratio in figures 8(a) and (b). Figure 8(a) is for an inlet with a design Mach number of 3 tested at a Mach number of 0.7. The band identified as being due to variations in cowl shapes is for identical inlet geometry except for variations in the cowl external profile. The other $K_{\text{add}}$ curves reflect changes in second-ramp angle and sideplate shape. Figure 8(b) shows some of the same effects at Mach number of 1.3. It is clear from these figures that independence from geometric detail is not achieved. If a correlation were to be forced onto these data (e.g., by using some average $K_{\text{add}}$ to represent all the geometries), the deviations from this mean must clearly represent sizable uncertainty in evaluating the effects of inlet drag on
airplane performance for applications where spillage is at all significant in comparison to total airframe drag.

5.2.3.3 The "Pitot Inlet Analogy"

Still another way of accounting for the change in cowl drag with MFR is by means of the "pitot inlet analogy". In this analogy, inlet external drag under spilling conditions for inlets with external compression surfaces is calculated in a manner similar to the simplified treatment for spilling pitot inlets generally attributed to Fraenkel (ref. 59). Fraenkel calculated the subcritical external drag of a pitot inlet in a supersonic free stream by simply multiplying the spillage area by the pressure rise across the normal shock. This approximation underestimates the force on the subsonic deflected streamtube (the additive drag), but in this way approximates the effect of cowl lip suction. No allowance is made for details of cowl geometry.

The application of the method of reference 59 to pitot inlets can be quite successful, as is illustrated in figure 10. This figure is taken from reference 21 and shows predictions and data at three Mach numbers from reference 59 for two pitot inlets shown in figure 9. However, examples can be found where the assumption of independence from cowl geometrical detail is inadequate. An unfavorable comparison is contained in reference 376. In this reference, substantial differences in pitot-inlet cowl lip suction at a given free stream Mach number and mass flow ratio are shown to exist which are due to differences in detailed cowl shape but which obviously cannot be reflected in the analysis.

Fraenkel's analysis has been extended to axisymmetric inlets with centerbodies in reference 71, and to two-dimensional inlets with external compression surfaces in references 21, 102, and 106. These extensions require a means of defining the terminal normal shock location as a function of the inlet mass flow ratio. This has usually been done in the manner suggested by Moeckel (ref. 72). Moeckel's method has been applied
to axisymmetric inlets in references 71 and 364 and to two-dimensional inlets in references 21, 32, 81, 106, and 364 (see Table VI). In reference 81, the adequacy of several of these approaches for the estimation of normal shock locations in two-dimensional inlets is examined, and for successful prediction the need to account for side spillage is emphasized. The effect of side spillage on terminal normal shock location is included in the empirical treatment of reference 32; an approximate analytical treatment is contained in reference 81.

Results of application of the pitot inlet analogy to two-dimensional inlets are shown in figures 11-15 and to axisymmetric inlets in figure 16-18. The comparisons shown encompass varying free stream Mach number, compression-surface geometry and details of cowl and sideplate shape. The level of agreement with experimental data illustrated is variable, but in general it is surprisingly good considering the simplicity of the approach. No dependence of the level of agreement on Mach number is evident, although the poorest agreement shown [fig. 12(b)] is for the case where the Mach number is so low that the wedge shock is detached.

5.2.3.4 Additive drag

As just explained, the pitot inlet analogy deals with inlet external drag as an entity. Returning to the approach wherein the cowl and additive drag components of external drag are predicted separately, the approximate methods for additive drag listed in Table VI will now be discussed. The calculation of additive drag by integration of the pressure along the stagnation streamline is treated in the section on Flow-field Methods; however, because it is a trivial matter in a supersonic free stream for geometries that allow all shocks to be attached, this is the procedure used to calculate $C_{DA_{crit}}$ in the "approximate" methods of this section. Calculation of critical additive drag in other speed ranges or in geometries with detached shocks, and calculation of subcritical $C_{DA}$ under all circumstances is typically done by means of a one-dimensional momentum analysis on a control volume.
encompassing the captured airflow extending from free-stream conditions to the cowl lip plane. Such a control volume is illustrated in the sketch below for a two-dimensional inlet operating subcritically in a supersonic free stream. For simplicity, viscous forces have been neglected and the inlet is shown at zero angle of attack.

One-dimensional momentum analysis for this control volume gives

\[ D_{\text{add}} = \int_{\infty}^{\ell} (P-P_\infty) dA_p + (P_{\ell}-P_\infty)A_\ell \cos \lambda + \int_{\text{ramp}} (P-P_\infty) dA_p \]

\[ + \rho_\infty A_\infty V_\infty (V_\ell \cos \lambda - V_\infty) \]  \hspace{1cm} (6)

or in coefficient form,

\[ C_{D_A} = \frac{(P_{\ell}-P_\infty)A_\ell \cos \lambda}{q_\infty A_c} + \int_{\text{ramp}} \frac{(P-P_\infty)}{q_\infty A_c} dA_p + 2(MFR) \left( \frac{V_\ell}{V_\infty} \cos \lambda - 1 \right) \]

\hspace{1cm} (7)

Calculation of \( C_{D_A} \) using this approach thus requires estimation of conditions in the cowl lip plane as well as the pressure force on the external compression surface. As is shown in this sketch, for subcritical operation of inlets with external compression surfaces in a supersonic free stream, the contribution of the integrated pressure drag of the external compression surface depends on the position of the terminal normal shock. Estimation of this position then becomes the
critical item, and the methods used are those discussed previously in connection with the pitot inlet analogy.

The reported comparisons with data of additive drag predicted using the methods listed in Table VI (all for isolated inlets at zero angles of attack and sideslip, e.g., references 32, 47, 49, 90, 111, 372) tend to lead to contradictory conclusions. For example, for pitot inlets*, in references 49 and 111 it is shown that additive drag is accurately predicted using the one-dimensional momentum analysis. In reference 372, however, values calculated in this way are generally lower than the presumably more accurate values calculated using two-dimensional incompressible potential flow theory and the Lieblein-Stockman compressibility correction. Comparisons of predictions from these two theoretical approaches are shown for an axisymmetric pitot inlet over a range of subsonic free-stream Mach numbers (fig. 19). However, because no comparison with data for $C_{DA}$ was made in this reference, no definitive conclusions relative to the accuracy of either prediction method can be drawn.

For axisymmetric inlets with external compression surfaces, figure 20 from reference 49 shows comparisons with data of predictions for several inlets at two Mach numbers. The agreement is quite variable. These inlets had the same cowl and cone half angle, but the cone was extended by different amounts at each Mach number corresponding to three values of supercritical mass flow ratio ($\beta$ in this figure). For the two-dimensional inlet shown in figure 21 the version of the momentum analysis expounded in reference 33 is shown in reference 47 to give very poor results (fig. 22). Figure 22(a) is for a case in which the ramp shock is detached; in reference 47 most of the disagreement is shown to be due to the ramp drag term. Measured ramp pressures are presented in reference 47 indicating a complicated viscous flow which is obviously not well

*In reference 90, an empirically derived curve is shown to adequately predict additive drag for some pitot inlets. However, no evaluation of the bounds of applicability of this method was conducted.
represented by inviscid one-dimensional theory. On the other hand, in figure 23, taken from reference 32, a momentum analysis (solid line), incorporating the normal-shock estimation procedure of reference 32 which has empirical allowance for the effects of sidespill is shown to give reasonably accurate predictions for the data from reference 33. The dashed line in that figure shows additive drag predicted using normal shock position estimated without allowance for sidespill (ref. 72) and agreement is degraded. Figure 23 and a comparison presented in reference 32 of measured normal shock position with predictions from the two methods used in figure 23 show that the sensitivity of additive drag prediction to normal shock position decreases with increasing Mach number. In spite of the success shown in figure 23 of the empirically based shock-location procedure, its applicability to geometries other than those of reference 33 (from which it was derived) is unknown.

The comparisons with data just discussed are inconclusive relative to the accuracy of the one-dimensional momentum approach to predicting $C_{DA}$. There is, however, an additional consideration which reflects adversely on the potential accuracy of the method. At a given $M_\infty$ and MFR, this method would predict the same additive drag for an axisymmetric inlet with conical centerbody of cone half-angle $\theta$ as for a two-dimensional inlet with the same throat-capture area ratio and a wedge angle of $\theta$ (excluding allowance for sidespill). However, it is well known (e.g., refs. 27 or 30) that the additive drag for two-dimensional inlets is higher than that for axisymmetric ones in this circumstance. This is due to the higher pressure force on the external compression surface caused by sharper curvature of the flow in the two-dimensional case. Inclusion of an allowance for sidespill in the prediction for a two-dimensional inlet (which lowers the predicted $C_{DA}$ as is shown in figure 23) does not help the situation.
5.2.4 Flow-field methods

In Table VIII, these methods have been divided into four classes: numerical solution of the unsteady Euler equations, finite-difference solution of the full potential equation, finite-difference solution of the incompressible potential equation with a compressibility correction, and the streamtube curvature approach.

By definition these methods provide a detailed description of the flow in the vicinity of the inlet, thereby allowing evaluation of all of the inlet drag terms. The price for this level of information is, of course, a computational cost that is increased substantially over that for the simple "partial" methods described earlier; in fact, in most implementations of these methods, the computational requirements are clearly in excess of what is reasonable for preliminary design purposes. In this section, a selection of the available comparisons with experimental data is presented to allow evaluation of the accuracy of the implementations of the basic approaches we have acquired. Also, to take account of the trade-off with computational costs, computer run times are given where they were reported. Because the numerical analysis and computer programming techniques used play a very important role in the implementation of flow-field methods, the accuracy and the computer cost demonstrated for a particular implementation may not apply to others of the same general approach. Thus, the conclusions drawn relative to speed and accuracy are usually applicable only to the particular implementation under discussion.

5.2.4.1 Unsteady Euler equations

In the class of flow-field methods consisting of numerical solution of the unsteady Euler equations, because of the details of the specific implementation, two of the methods are applicable only to pitot inlets in a supersonic free-stream (refs. 154, 354). In one of these, reference 154, the particular formulation of the boundary condition used resulted
in numerical instabilities and insufficient accuracy to calculate pressure distributions on "thin cowls," although bow shock geometry could be predicted fairly well. Typical central processor unit (CPU) time for a converged solution is ~4 minutes on a CDC 7600 machine. In the other of these pitot-inlet methods, reference 354, fairly good agreement with an experimental cowl surface pressure distribution is demonstrated for \( M_{\infty} = 1.14 \), \( M_{FR} = 0.98 \), and for bow-shock standoff distance for range of mass flow ratios at that Mach number using a relatively coarse computational grid. No integrated drag data are presented. For this method and grid, about 8 minutes on an IBM 370/158 is required for a converged solution.

Reference 19 presents an explicit finite-difference method of solution of the unsteady Euler equations which is valid for two-dimensional flow fields for two-dimensional or axisymmetric geometries in any speed range. The solution algorithm is briefly described, as is the formulation of the required boundary conditions and methods used to speed convergence of the solution. Interior points in the required computational mesh are automatically generated by the program from mesh coordinates specified on the boundaries. Solutions are presented and compared with data at two values of MFR for a two-dimensional three-ramp external-compression inlet at \( M_{\infty} = 0.7 \). Because the essential features of the comparison with data in both cases were the same, only one value of MFR (MFR = 0.5) will be discussed here. The geometry considered is shown in figure 24. Comparisons of calculated values with measured pressures on the ramp (fig. 25(a)), on the cowl external surface (fig. 25(b)), and on the cowl-lip internal surface (fig. 25(c)) are shown.

In these figures, good qualitative agreement is evident, but in reference 19 it is concluded that pressure level is not satisfactorily predicted. Drag computed from these pressure distributions would be considerably in error, particularly on the cowl external surface; the authors feel that improvement in this area would require a refined mesh structure. However, for the solutions shown, the full storage capacity of the available machine (a CDC 6600) was used, and approximately 5.6
hours of computer time were spent in the evolution of the flow field from
the prescribed initial free-stream conditions. Although the solutions
shown are not considered to be steady state (an additional 1.5 hours of
computer time was estimated for full convergence), the calculated
pressures are estimated to be near their final values; major improvement
is felt to depend on mesh refinement.

5.2.4.2 Full potential equation

Finite-difference solution of the full potential equation is the
second of the general classes of flow-field methods. These methods are
valid up to free-stream speeds which result in local supersonic regions
terminating in strong shocks. Various implementations of the basic
approach have been made for pitot inlets and axisymmetric and two-dimen-
sional inlets with external compression surfaces. Reference 215 is an
example of an implementation of the general scheme valid for axisymmetric
pitot inlets at angle of attack in a uniform subsonic free stream; note
that this is one of the two flow-field methods we have acquired which can
deal with a three-dimensional flow field. In reference 215, the solution
method is described (including special handling of the circumferential
derivatives), as are the boundary conditions and the initial field.
Comparisons with measured surface Mach number are given for five pitot
inlets operating in low-speed free streams ($M_\infty \leq .2$) at a variety of
angles of attack and mass flow ratios (throat Mach numbers $\leq 0.86$: at
0.86, the inlet was observed to be choked). Very good agreement with the
data is demonstrated. No integrated drags were calculated. Typical run
times for a converged solution are from 12 to 15 minutes on a CDC 6600
computer.

Another implementation of this class of methods is that of reference
37. The same implementation is also described in reference 52. This
version allows solution of two-dimensional flow fields for inlets of
either two-dimensional or axisymmetric geometry with external compression
surfaces. The method is applied to two-dimensional inlets and is com-
pared to data in references 63 and 224. Examples of these comparisons are given in figure 26 which is reproduced from reference 63. In this figure, fair agreement is shown for the drag slope but poor agreement is achieved for drag level. In reference 63, the disagreement is attributed to the existence of three-dimensional effects and shocks, and to inadequate grid resolution in the computation in regions of high gradients. Application of this technique to an axisymmetric inlet is shown in figure 27, taken from reference 37. Here, the level of $C_{D_{\text{EXT}}}$ has been adjusted by calculating additive drag using momentum analysis on the entering streamtube and replacing the stream thrusts generated by the method with one-dimensional values. The approach more typical in flow-field methods of integrating the pressure distribution on the stagnation streamline leads to erroneous $C_{D_{A}}$ in this implementation. The discrepancy in level of additive drag observed here is claimed not to exist in the calculation for two-dimensional geometries. However, in the analysis of two-dimensional inlets, a problem with mass conservation in the calculation is indicated. This problem is obviously attributable to this specific implementation of the potential-equation method. That is, it is not a general characteristic of this class. However, because the implementation described in references 37 and 52 is the only application of the full potential equation to inlets with external compression surfaces we have been able to discover in the published literature, it is not clear whether the problem with additive drag level described above is a general one. Computational run times on the order of 15 seconds on a CDC 6400 computer are claimed.

5.2.4.3 Incompressible potential equation with compressibility correction

The third general class of method appearing in Table VIII is the solution of the incompressible potential equation with a compressibility correction. One implementation of this method we have acquired is described in reference 52, where the Douglas Neumann incompressible flow program (refs. 8.4-8.6) is coupled with either the
Prandtl-Glauert, Kármán-Tseiu, Laitone, or Krahn compressibility correction. The method is applied to two-dimensional and axisymmetric inlets with centerbodies; in ref. 52, surface pressures on the cowl and centerbody are compared to data at $M_\infty = 0.7$, 0.9 and several mass flow ratios for the axisymmetric inlet shown schematically in figure 27. No integrated values of drag are given. The calculated pressure coefficients agree best using the Laitone correction.

The other implementation of the class of methods shown in Table VIII is that of reference 405. This is the second of the two Flow-field Methods we have acquired which can deal with a three-dimensional flow field. In this approach, the incompressible flow about a pitot inlet with or without a centerbody is solved using a panel method. Solutions for unit onset flows parallel to each of the coordinate axes are combined with a solution for static operation to result in a rigorous incompressible solution for an inlet at arbitrary angles of attack and yaw with arbitrary mass flow rate. Compressibility is accounted for by applying the Lieblein-Stockman correction (ref. 8.7). The method is applicable for subsonic free streams, but local regions of the flow may be supersonic although they should be shock-free. The geometry may be three-dimensional but it must exhibit a plane of symmetry. In reference 405, the method is applied to a subsonic-design-point axisymmetric pitot inlet at 75° angle of attack in a situation where the throat Mach number is 0.603 (the free stream velocity is not reported). Excellent agreement with data is shown for pressures on the internal and external cowl surfaces. Drag is not calculated. It is observed in reference 405 that panel methods usually require considerably less computer time than finite-difference methods; the superposition technique and compressibility correction used also lead to computational efficiency. Thus, it is claimed that the method of reference 405 is faster than a finite-difference method by two orders of magnitude.
5.2.4.4 Streamtube curvature analysis

The final general class of methods uses streamtube curvature (STC) analysis. This approach, as explained in reference 274, basically uses one-dimensional compressible flow analysis in a number of adjacent streamtubes; when taken together the entire flow field is simulated. Streamline positions are refined iteratively; in each iteration the momentum equation normal to the streamlines is integrated using calculated values of streamline curvature to obtain velocity, and the continuity equation is used to define a new streamline position. The iterative process is continued until streamline movement is less than a specified amount.

A characteristic of streamtube curvature methods, as discussed in reference 239, is that the method is extremely sensitive to input geometry. As stated therein, "The difference between an aborted run and a successful run is usually a minute change in the geometric data." This extreme sensitivity is due to the need to calculate surface curvatures from input geometry and can make the method very troublesome to use. In reference 274, the method is applied to the pitot inlet shown in figure 28 for several Mach numbers and mass flow ratios. Calculated results accounting for viscous effects by means of the integral boundary layer method of Stratford and Beavers (ref. 8.8) are included. Sample comparisons with data from three circumferential positions identified in ref. 274 as "NASA-Langley Data, ATT Nacelle Inlet Test, 16 Foot Transonic Tunnel" are shown in figure 29. In figure 29(a), a fully subsonic case is shown and excellent agreement is obtained. At a higher value of free-stream Mach number, figure 29(b), increasing the number of grid points in the STC solution led to local oscillations in the inviscid solution which were claimed to be eliminated in the physical situation by "viscous effects". At a still higher Mach number, figure 29(c), predictions are included done with the boundary layer analysis (labeled STC-SAB) and without it (labeled STC). It is seen that neither one is in agreement with the data; separation over the initial portion of the cowl
lip was observed experimentally at this condition, and the integral boundary layer method is inadequate in this situation. Cowl drag forces integrated from the predictions and data of figures 29(a), (b), and (c) are in reasonably good agreement (within .01 for $C_{DC}$), but at least in the case of figure 29(c), this must be attributable to compensating errors.

In reference 239, application of the streamtube curvature method to the B-1 external-compression inlet shown in figure 30 is made for free-stream Mach numbers of 0.7 and 0.85 for values of MFR of 0.55 and 0.75 and several positions of the variable ramps. Sample comparisons with data are shown in figures 31-33. All of these figures are for the ramp positions $R_B = 7.2^\circ$, $H_L = 27.3$ and are for $M_\infty = 0.85$. Figure 31 shows the ramp and cowl pressures for MFR = 0.55, while figure 32 is for MFR = 0.75. Additive drag over a range of mass flow ratios is shown in figure 33 for this Mach number and geometry. The experimental data were obtained for an inlet mounted on a fuselage and stub wing assembly. Allowance for the angularity of the local inlet flow field was made by running the STC program using various incidence angles and selecting the incidence at each value of $M_\infty$ that gave best agreement with the measured ramp pressures. Reasonable values of flow incidence resulted from this procedure and the agreement for ramp pressures in figures 31(a) and 32(a) is fairly good. Calculated cowl pressures agree poorly with the measurements, as shown in figures 31(b) and 32(b). This poor agreement is likely due to some combination of inadequate grid resolution and viscous effects, but further work would be required to ascertain this. However, additive drag is predicted with reasonable accuracy, as seen in figure 33; this positive result is possible because of the high degree of two-dimensionality of the inlet flow field indicated by the transverse taps in figures 31 and 32.
5.2.5 Concluding remarks

An evaluation of the adequacy for preliminary design (based on completeness, ease of application, and accuracy) of the data base for inlet external drag has revealed:

1. The sensitivity of airplane performance to inlet external drag is a strong function of the airplane's mission. Therefore, the required accuracy for preliminary design methods cannot be stated in general. However, for aircraft with mixed missions, inlet external drag exerts a powerful influence on the aircraft's performance and accurate predictions are required to allow for rational configuration definition.

2. Although a few systematic studies exist, the experimental data portion of the data base consists mostly of specialized studies of particular inlets. Because inlet external drag depends in an important way on geometrical details, this portion of the data base is adequate for prediction only for inlets not substantially different than those previously tested.

3. Calculation of cowl lip suction by methods with a reference mass flow ratio of unity are inaccurate because of the large extrapolation of data usually entailed.

4. Values of lip suction as calculated from the correlation of reference 32 for inlets reasonably similar to those from which the correlation was derived are in only fair agreement with data at low mass flow ratios. Because of its formulation, this method cannot predict experimentally demonstrated differences in lip suction resulting from changes in detailed cowl shape.

5. The $K_{add}$ factor does not exhibit freedom from geometric detail. Its use is equivalent to using the original experimental data.
6. Use of the "pitot inlet analogy" to predict $C_{D_{EXT}}$ in a supersonic free stream leads to variable agreement with measurements (figs. 11-18). In this method, $C_{D_{EXT}}$ is again assumed to be independent of detailed cowl lip shape.

7. Prediction of additive drag using a one-dimensional momentum analysis leads to acceptable agreement with data only for the simplest possible configuration, i.e., a pitot inlet. Predictions in other situations are in general poor, although methods containing additional empiricism (e.g., for normal shock location) can give reasonable agreement in limited classes of geometries.

8. Some of the Flow-field Methods discussed herein can yield adequate results although the comparisons with data available do not allow for comprehensive evaluation of their limits of applicability. For those methods applicable to supersonic design-point inlets, the large computers and long run times required to achieve satisfactory agreement result in their being of limited usefulness for preliminary design.

6. DATA BASE FOR AFTERBODY DRAG AND THE EVALUATION OF ITS ADEQUACY FOR PRELIMINARY DESIGN

In this section, the part of the airframe/propulsion system interaction data base that is associated with afterbody drag is evaluated. Other aerodynamic interference effects associated with the afterbody will not be considered in detail in this report.

6.1 Requirements for Drag Prediction

The functional dependence of afterbody drag can be represented as

$$C_{D_{AB}} = f(\text{Afterbody Geometry, } M_{\infty}, \text{NPR, Re, } \alpha)$$ (8)
where the specification of Afterbody Geometry includes such features as number of engines, interfaireing type, engine nacelle spacing, empennage type and nozzle type. This relationship includes complex interaction effects due to the interference flow fields of the various constituents of the afterbody/airframe system as well as exhaust plume shape and entrainment effects. The total afterbody drag of a twin-engine fighter model can amount to 40 to 45 percent of overall zero-lift vehicle drag even though the afterbody portion of the vehicle may comprise only one-third of the total wetted area.

In the preliminary design process, this relationship must be evaluated for various candidate configurations to allow the most promising of them to be selected for development. To be useful for preliminary design, the afterbody/airframe data base must satisfy the same three requirements as for inlets, namely:

1. Breadth
2. Ease of application
3. Accuracy

In the following sections, the experimental data and prediction methods that exist for afterbody/drag are evaluated with respect to these requirements. Empirical prediction methods are generally found to be the easiest to apply but may be quite limited in the requirements of breadth and accuracy. Theoretical methods are more difficult to apply, may require costly computer time, and generally also suffer from lack of breadth and accuracy.

With regard to the accuracy required, it is clear that in a complex system such as a high performance fighter, the errors associated with each constituent of the overall system must be kept as small as possible. The general remarks made previously with regard to inlet external drag apply to the afterbody drag problem as well. That is, no simple general statement of the accuracy required can be made. The
minimum drag increment that can be allowed is a function of the detailed nature of the mission considered.

Sensitivity of aircraft range to errors in predicted drag has already been discussed in section 5. The same considerations apply with regard to afterbodies since the afterbody drag can be a significant percentage of the total vehicle drag.

6.2 Empirical Prediction Methods

6.2.1 Single-engine configurations

The only comprehensive empirical method for single engine configurations is the Afterbody Drag Approximation Procedure (ADAP) of Bergman, et al., described in references 6, and 254-256. The method was adapted for use on a CDC 7600 computer by Kuhn (ref. 307). The ADAP calculates pressure drag for axisymmetric or non-axisymmetric (2-D) afterbody/nozzle configurations. The program is based on data correlations presented in references 254 and 255 for isolated bodies (no wing, tail, or 3-D fuselage effects) at subsonic speeds (Mach number less than approximately 0.9) and, for purely axisymmetric configurations, at supersonic speeds (Mach numbers from 1.0 to 1.8). Base drag and friction drag are not included in the correlations.

6.2.1.1 Input parameters

The process of selecting geometric and flow-field parameters was described in reference 254. Because the intended use of the parameters was to relate experimental data, rather than for generating analytical solutions, they were selected to be somewhat generally descriptive instead of being extremely precise. Also, the input parameters were derived to be easily determined for a practical drag prediction procedure for preliminary design. The procedure applies to pressure drag only. Friction drag can be predicted separately and added to the pressure drag.
Geometric parameters.—Fuselage/nozzle afterbody geometry is described by non-dimensional parameters. The forebody is assumed to be a constant-area section extending upstream from the afterbody's maximum area. Afterbody geometry is simplified to the extent that protuberances such as horizontal and vertical tails are not included. Specifically, the type of afterbodies considered are those having smoothly contoured surfaces (no sharp corners) and negligible base areas (nozzle with thin trailing edges). The geometric parameters used in the ADAP are illustrated in figure 34. Note that fineness ratio and external area ratio (closure ratio) are used to define the afterbody rather than a detailed description of the surface contours. This approach was taken because it was found that smoothly contoured boattails such as those with circular arc or parabolic shapes exhibit roughly identical drag characteristics at subsonic Mach numbers. In contrast, contours having high slopes at the forebody/afterbody junction, such as conical afterbodies, usually create abnormally high drag levels and early drag rise. Such contours are excluded from the afterbodies used in the ADAP correlations.

Another geometric parameter that is related to afterbody drag is the nozzle expansion ratio. That parameter determines the exit flow Mach number and has a significant influence on the shape of the exhaust plume.

In summary, the geometric parameters required as input to the ADAP are as follows:

1. Fineness ratio, $L/Am = FR$
2. Nozzle expansion ratio, $Ae/At = ER$
3. Boattail closure ratio, $Ae/Am$

Flow-field parameters.—In addition to free-stream Mach number and Reynolds number, a parameter that has been found to be important in describing nozzle flow fields is the nozzle pressure ratio, defined as the exhaust-stream total pressure divided by the free stream static
pressure. Another parameter that is useful in describing plume conditions is the ratio of exhaust-stream static pressure, at the nozzle exit, to the free-stream static pressure. This parameter can be derived from the nozzle pressure ratio, nozzle geometry and free-stream Mach number, and so is not an independent parameter.

The influence of Reynolds number on boattail drag has not been completely determined for bodies of the type used in the ADAP correlations. For the data used in the derivation of the ADAP, Reynolds number was not a significant parameter. Consequently, Reynolds number is not included in the input.

The flow field parameters required for input are:

1. Free-stream Mach number, $M_\infty$
2. Nozzle pressure ratio, NPR, $p_{ej}/p_\infty$
3. Ratio of specific heats of the exhaust, $\gamma$

Additional parameters for two-dimensional nozzles.—The effect of non-axisymmetric nozzles was accounted for by assuming the maximum cross-sectional area is square and that two-dimensional effects are important only when the exit plane geometry is not square. The important parameters for two-dimensional nozzles are then:

1. Height ratio of nozzle, $H_e/H_m$
2. Width ratio of nozzle, $W_e/W_m$

6.2.1.2 Data correlation technique

Afterbody drag can be represented as a basic drag characteristic of the afterbody itself, plus the effects of exhaust plume interference. A jet-effects analysis conducted by General Dynamics resulted in the development of a methodology of dividing jet effects into components of plume-shape effects and entrainment effects. A more detailed account of
the jet-effects analysis is found in reference 6. A pictorial description of the drag components is presented in figure 35. Notice that basic afterbody drag is defined as the drag which occurs if a semi-infinite cylinder extends downstream of the nozzle in place of a true plume.

For subsonic free-stream flow, jet-on afterbody drag was found to approximate basic drag (drag with a cylinder in place of a jet) when the plume's theoretical (1-D) fully-expanded area is roughly 9% larger (averaged value) than the nozzle exit area; that is, when nozzle flow is underexpanded by this amount. Examples of this phenomenon from references 6 and 46 are shown in figure 36. Therefore, at NPR conditions producing this amount of underexpansion, plume entrainment effect is equal in magnitude but opposite in sense to plume shape effect. Also, it can be assumed that entrainment effects are negligible when the plume momentum per unit area equals that of the free stream. For example, for an unheated plume, zero entrainment effect occurs when the NPR equals the free-stream total-to-static pressure ratio, that is when the Mach numbers of the jet and the free stream are equal. Although this low-NPR situation is unlike nozzle operation in flight, nozzle test data are sometimes taken at this condition.

For supersonic free-stream conditions, reference 256 points out that the plume underexpansion percentage is a function of afterbody/nozzle geometry. As a result, basic drag curves for supersonic flows were derived from a schedule defining the necessary plume expansion (in terms of nozzle pressure ratio, NPR) to approximate basic drag for different boattail geometries (fig. 37).

Basic afterbody drag can be predicted via configurations actually having long cylinders in place of plumes or by using the average 9 percent expanded-area criteria for subsonic flow or the appropriate expansion for supersonic flow. Plume entrainment effects were determined by measuring boattail drag with a solid, plume-shaped extension attached and then comparing the drag to the drag without the extension, but with
the actual pressurized exhaust in its place. For supersonic flow, it was assumed that entrainment occurred at choked flow \((\text{NPR} = 2, M_e = 1\) for a convergent nozzle) because of subsonic flow in the boundary layer being accelerated by the supersonic flow in the exhaust jet. Shape effects were assumed to occur when plume billowing began, that is, at \(\text{NPR}'s\) greater than that required for fully expanded nozzle flow.

Therefore, given afterbody drag values for various geometry/flow field conditions, basic drags \((C_{D_B})\) and plume entrainment effects \((\Delta C_{D_E})\) were determined (refs. 255 and 256). Accordingly, the equation

\[
C_{D_{AB}} = C_{D_B} + \Delta C_{D_S} + \Delta C_{D_E}
\]

presented in figure 35, was solved for \(\Delta C_{D_S}\), having determined the remaining terms. These drag components were then correlated such that their respective values for fuselage/nozzle geometry in general could be systematically predicted.

Basic afterbody drag.- The drag of smoothly contoured afterbodies (e.g., circular arc, parabolic) with cylinders in place of exhaust plumes was correlated by the use of fineness ratio and boattail closure ratio parameters. Typical correlations of basic afterbody drag are presented in figure 38. Reynolds-number effects on pressure drag coefficient are normally negligible at free-stream Mach numbers below drag rise (Mach numbers less than approximately 0.85) for turbulent boundary layers and smoothly contoured afterbodies—the situation being analyzed. Thus the drag coefficients presented are applicable to Reynolds numbers associated with turbulent boundary layer conditions.

Plume shape effect.- The maximum cross-sectional area to which an exhaust plume expands is a predominant factor influencing the plume interference effects on afterbody drag. External plume expansion will vary not only with nozzle pressure ratio, but also with nozzle internal
area ratio. Unfortunately, the majority of available test data at subsonic Mach numbers are for afterbodies with convergent nozzles or only slightly convergent-divergent nozzles (i.e., $A_e/A_t \approx 1$). Therefore, generalized drag predictions for afterbodies with convergent-divergent nozzles were handled in a unique manner; in particular, the method used in the ADAP analysis was to transform the afterbodies with convergent-divergent nozzles into equivalent-drag afterbodies with convergent nozzles so that relatively abundant convergent nozzle data could be used. Specifically, an afterbody with a convergent-divergent nozzle is converted into an afterbody with a convergent nozzle which has an identical afterbody external shape and an identical plume external expansion. This approach is illustrated in figure 39. This necessitates that a "corrected", rather than true, nozzle static pressure ratio, $p_j/p_\infty$ (NSPR) be used when analyzing the afterbody drag with a convergent-divergent nozzle. Corrected NSPR is that NSPR at which a convergent nozzle must operate to produce a plume with a maximum free-expansion area identical to that of the convergent-divergent nozzle in question. Note that the theoretical full-expansion area (from isentropic flow tables) is used as a quasi-average representation of the periodic behavior of the plume. Examples of the transformation from actual NSPR to corrected NSPR are given in figure 40.

The correlation of plume shape effect is presented in figure 41. As shown in these curves, a corrected NSPR equal to unity (i.e., no external expansion of the plume) produces no effect on basic drag. Basic drag can be described as the drag of an afterbody having a pseudo, cylindrically shaped nonentraining plume. The plume billows beyond the diameter of the nozzle exit at values of corrected NSPR greater than 1 and creates a drag-reducing effect.

Plume entrainment effect. - Plume entrainment, like flow separation, is a result of viscosity. Entrainment occurs because of the momentum differential between the plume and the external flow. The exhaust plume,
having more momentum per unit area than the free stream entrains external flow.

The majority of subsonic test data used in the correlations pertain to free-stream Mach number between 0.4 and 0.85. However, more emphasis was placed on correlating the Mach 0.85 data since this is more representative of typical subsonic cruise conditions. The entrainment correlation is based on a jet momentum ratio given by

\[ \theta = \frac{(\rho v^2)_e}{(\rho v^2)_\infty} = \frac{(\gamma p M^2)_e}{(\gamma p M^2)_\infty} \]  \hspace{1cm} (10)

from which

\[ \theta_{\text{corr}} = \text{NSPR}_{\text{corr}} \left( \frac{\gamma_e}{\gamma_\infty} \right) \left( \frac{M e_{\text{ref}}}{M e_{\text{ref}}} \right)^2 \left[ \frac{1 + \frac{M - M_{\text{ref}}}{M_{\text{ref}}}}{1 + \frac{M - M_{\text{ref}}}{M_{\text{ref}}} \left( \frac{M - M_{\text{ref}}}{M_{\text{ref}}} \right)^2} \right] \]  \hspace{1cm} (11)

where \( M e_{\text{ref}} = 1 \) and \( M e_{\text{ref}} = 0.85 \)

Thus,

\[ \theta_{\text{corr}} = 1.384 \text{NSPR}_{\text{corr}} \left( \frac{\gamma_e}{\gamma_\infty} \right) \left[ \frac{1 + (M - 1)^2}{1 - (1.176 M_\infty - 1)^2} \right] \]  \hspace{1cm} (12)

The correlation of jet entrainment effects via the \( \theta_{\text{corr}} \) momentum parameter is shown in figure 42. As expected, momentum ratios greater than unity produce a drag-increasing effect due to viscous pumping.
6.2.1.3 Calculation procedure for pressure drag on axisymmetric bodies

Equations used.- From the input quantities discussed in the previous section, the computer program calculates two parameters to obtain the plume shape and entrainment effects. These parameters are NSPR\textsuperscript{corr}, an effective nozzle static pressure ratio, and $\textit{\theta}$, the jet momentum ratio. NSPR is the nozzle static pressure ratio, as defined previously. NSPR\textsuperscript{corr} is that NSPR at which a convergent nozzle must operate to produce a plume with a theoretical, maximum-free-expansion area, $(A_j)_{f.e}$, identical to that of the convergent-divergent nozzle in question. The momentum ratio, $\textit{\theta}$, is the jet-momentum at the exit divided by the free stream momentum.

The calculation of NSPR\textsuperscript{corr} is done in four steps by the ADAP procedure:

1. Knowing NPR, calculate corresponding $(A_j)_{f.e.}/A_t$.
2. Divide $(A_j)_{f.e.}/A_t$ by the expansion ratio, $A_e/A_t$.
3. Knowing $(A_j)_{f.e.}/A_e$, which is an area ratio, calculate corresponding $p/p_t$, the reciprocal of which is NPR\textsuperscript{corr}.
4. Multiply NPR\textsuperscript{corr} by $p/p_t$ at $M_e = 1$ (convergent nozzle case) to obtain NSPR\textsuperscript{corr}.

The user inputs the expansion ratio, the nozzle pressure ratio, and $\gamma$ into the ADAP, and the program internally performs these steps. The first step, from the Mach functions, is

$$
\frac{(A_j)_{f.e.}}{A_t} = \frac{1}{\sqrt{\left[\frac{2}{\gamma-1}\right]^\frac{\gamma-1}{\gamma-1} \left[\frac{\gamma-1}{\text{NPR}}\right]^\left[\frac{\gamma+1}{2(\gamma-1)}\right]}}
$$

(13)
When considered relative to the nozzle exit station, this potential expansion is combined with the internal expansion ratio, thus yielding a relative expansion ratio defined by

\[
\frac{(A_j)_{f.e.}}{A_e} = \frac{(A_j)_{f.e.}}{A_t} \cdot \frac{A_e}{A_t}
\] (14)

This parameter implies that a relative or corrected nozzle pressure ratio is needed to describe external plume expansion. This NPR\textsuperscript{corr} is the $p_t/p$ that corresponds to the $(A_j)_{f.e.}/A_e$.

Mathematically, no closed-form solution exists for calculating a pressure ratio when given an area ratio. ADAP uses the Newton-Raphson method to find NPR\textsuperscript{corr} from the relations

\[
f = \left( \frac{2}{\gamma+1} \right)^{-1} \left( \frac{p_t}{p} \right)^{\gamma+1} - \left( \frac{2}{\gamma-1} \right)^{\gamma} \left( \frac{A_j}{A_e} \right)^2 \left( \frac{p_t}{p} \right)^{\gamma-1} + \left( \frac{2}{\gamma-1} \right)^{\gamma} \left( \frac{A_j}{A_e} \right)^2
\] (15)

\[
\frac{df}{dp_t} = f' = \left( \frac{2}{\gamma+1} \right)^{-1} \left( \frac{p_t}{p} \right)^{\gamma+1} \left( \frac{1}{\gamma} \right) \left( \frac{p_t}{p} \right)^{\gamma} - \left( \frac{2}{\gamma-1} \right)^{\gamma} \left( \frac{A_j}{A_e} \right)^2 \left( \frac{p_t}{p} \right)^{\gamma-1} - \frac{1}{\gamma}
\] (16)

and the proper value of $p_t/p$ is that value which causes $f(p_t/p) = 0$. NSPR\textsuperscript{corr} is then found from

\[
\text{NSPR}_{\text{corr}} = \frac{p_t}{p} \left[ 1 + \frac{\gamma-1}{\gamma} \right]^{\frac{\gamma}{1-\gamma}}
\] (17)
The momentum ratio, $\theta$, is then calculated from

$$\theta = 1.384 \left(\text{NSPR cor} \right) \frac{\gamma}{1.4} \left[ \frac{1 + (M_e - 1)^2}{1 - (1.176 M_e - 1)^2} \right]$$  \hspace{1cm} (18)

In the above equation, all of the variables, except $M_e$, are known. For ADAP, the exit Mach number is a function of expansion ratio and $\gamma$. Since there is no closed-form solution for Mach number given an expansion (area) ratio, the Newton-Raphson iterative technique is used to find $M_e$, where:

$$f = \left[ \frac{2}{\gamma+1} + \frac{\gamma-1}{\gamma+1} M_e^2 \right]^{\frac{\gamma-1}{2(\gamma-1)}} - \left( \frac{A_e}{A_t} \right) M_e$$  \hspace{1cm} (19)

$$\frac{df}{d(M_e)} = f' = M_e \left[ \frac{2}{\gamma+1} + \frac{\gamma-1}{\gamma+1} M_e^2 \right]^{\frac{3-\gamma}{2(\gamma-1)}} - \frac{A_e}{A_t}$$  \hspace{1cm} (20)

When $f$ converges to 0, $M_e$ is found. The momentum ratio, $\theta$, is then calculated from Eq. (18).

Method of interpolation.- In the computational technique described in references 254, 255, 256, and 307, there are 174 parametric curves that define values necessary for computing the three drag components (basic pressure drag and the two corrections for plume shape and entrainment) of the axisymmetric afterbody/nozzle pressure drag. For the basic afterbody drag curves, for subsonic flows, 20 control points (points from ref. 354 curves, fig. 38) are used to define each curve. For the supersonic basic drag curves, the plume-shape-effect, and the plume-entrainment-effect curves, 6 control points are used for each curve (figs. 38(b), 41, and 42).
The appropriate data arrays containing control points necessary to describe specific curves for corresponding NSPR\text{corr} and \( \theta \) at given FR are used during program execution. The 20 or 6 control points (depending on which curve type is called) are mathematically interpolated (curve-fit) with a 3rd-order, continuous-1st-and-2nd-derivative, natural-spline-function subroutine. When the control points are curve-fit, evaluation at any coordinate on the curve is available. For the interpolation between NSPR\text{corr}, FR, and \( \theta \) curves, a linear interpolation is used.

An example is shown for illustration. For the case of FR = 2.3, \( A e / A m = 0.53 \), NPR = 6, ER = 1.1, and \( \gamma = 1.4 \), the plume shape effect is to be determined. The program would calculate NSPR\text{corr} = 2.6 and obtain the data arrays containing control points describing NSPR\text{corr} for NSPR\text{corr} = 2 and 3 at FR = 2 (see fig. 43). Then the spline-functions subroutine would evaluate \( \Delta C_{DS} \) at NSPR\text{corr} = 2 and 3 at \( A e / A m = 0.53 \). The program would then interpolate the plume-shape drag increment for NSPR\text{corr} = 2.6 from these two values. The above process would be repeated for the FR = 2.5 points. Once values at both FR's (FR = 2 and FR = 2.5) are calculated, linear interpolation can be used to produce the plume shape increment, \( \Delta C_{DS} \) at FR = 2.3.

To calculate the plume entrainment effect, the program would use the momentum ratio, \( \theta \) as the correlating parameter (instead of NSPR\text{corr}) and follow the same procedure.

6.2.1.4 Calculation procedure for subsonic pressure drag on non-axisymmetric bodies

Only a limited amount of data is presently available with regard to drag on non-axisymmetric nozzles. Consequently, there is great difficulty in defining an accurate generalized correlation of 2-D effects.
Published and unpublished test data from Pratt and Whitney Aircraft's isolated parametric 2-D nozzle tests (ref. 249) were incorporated by General Dynamics as the primary data source. These data, to be discussed in the following section, were employed in reference 255 to define relationships between axisymmetric and nonaxisymmetric nozzles that are used to calculate the 2-D nozzle pressure drag.

Experimental data.—Experimental data used to develop the correlation are shown in figure 44. The geometry is representative of single-engine installation, i.e., square at the forebody's maximum cross-sectional area. At the exit plane, the width is the same as the width at the afterbody maximum cross section (no side boattailing). Height at the exit plane is one-fourth the height at the maximum cross section. Therefore, the nozzle exit plane forms a 4:1 rectangle. The P&WA data was originally in the form of drag coefficients referenced to aft-facing projected area vs terminal boattail angles. The ADAP program calculates drag coefficients referenced to maximum cross-sectional area and calls for fineness and area ratios rather than boattail angles as the input quantities. Therefore, a conversion of $C_{D_Ap}$ to $C_{DAm}$, as well as $D_e/D_m$ and boattail trailing edge angle, $\beta$ to $A_e/A_m$ and FR was performed using the following equations

1. $A_p = A_m - A_e$
2. $A_e = .25 A_m$ since $H_e/H_m = .25$ and $W_e/W_m = 1$
3. $A_p = .75 A_m$

Therefore

$$C_{DAm} = .75 C_{DAp}$$
Correlation parameter.- A comparison, shown in figure 45, relates the 2-D drags to ADAP-predicted axisymmetric-nozzle drags based on the same nozzle-pressure and area ratios. The difference between the two drag levels is an incremental effect which is correlated and manipulated to convert axisymmetric nozzle predictions into those for 2-D nozzles. It was also reasoned in reference 255 that 2-D geometry predominantly affects the basic pressure drag (because of altered flow recompression) and entrainment drag increments (because of increased plume surface area) as opposed to affecting plume billowing. Consequently, this relationship was defined as

\[ \Delta C_{D_{2-D}} = \alpha \left[ \frac{\Delta C_{D_{2-D}}}{C_{D_{B}} + \Delta C_{D_{E}}} \right] \left( C_{D_{B}} + \Delta C_{D_{E}} \right)_{\text{ADAP}} \]

where the quantity \( \Delta C_{D_{2-D}}/(C_{D_{B}} + \Delta C_{D_{E}})_{\text{axi}} \) is a function of fineness ratio and, as shown in figure 46, is stored via 10 control points in the ADAP program data files.

While the above relationship accounts for the geometric parameters of fineness ratio and \( A_e/A_m \), it does not account for variations in sidewall boattailing; thus, the proportionality sign is employed. The P\&WA data involves a 2-D nozzle having no sidewall boattailing. In order to investigate 2-D nozzles having some sidewall boattailing, a constant, \( \phi \), is used to include the effects of sidewall boattailing, so that
\[ \Delta C_{D_{2-D}} = \phi \left[ \frac{\Delta C_{D_{2-D}}}{C_{D_B} + \Delta C_{D_E}} \right]_{\text{adi}} \left( C_{D_B} + \Delta C_{D_E} \right) \]  

The parameter \( \phi \) must be a function of the input 2-D geometry; ADAP uses the parameters \( \frac{W_e}{W_m} \) (exit width divided by max. afterbody width) and \( \frac{H_e}{H_m} \) (exit height divided by max. afterbody height). For the PWA test, \( \frac{W_e}{W_m} = 1 \) and \( \frac{H_e}{H_m} = 0.25 \); however, one must recognize that for an isolated 2-D nozzle the flow field cannot distinguish the "side" of the nozzle from the "top"; hence, the data would be the same if \( \frac{W_e}{W_m} = 0.25 \) and \( \frac{H_e}{H_m} = 1.0 \). This evaluation is important because it restricts the possible relationships between \( \phi \) and the 2-D geometric parameters. By assuming that for an isolated 2-D afterbody, the maximum cross-sectional area is square \( \left( \frac{W_m}{H_m} = 1 \right) \), and then assuming that 2-D effects are significant only when the exit plane 2-D geometry is not square, the following equation was derived.

\[ \phi = \left| \frac{(1 - W_e/W_m)(1 + H_e/H_m) - (1 + W_e/W_m)(1 - H_e/H_m)}{2 \left( 1 - \frac{W_e}{W_m} \frac{H_e}{H_m} \right)} \right| \]  

The final equation, therefore, is a function of sidewall boattailing as well as upper- and lower-surface boattailing, fineness ratio, nozzle pressure ratio, and Mach number and is defined by the following relationship:
6.2.1.5 Comparisons with experimental data

Several example cases were calculated to evaluate the prediction method. The first two examples, shown in figures 47 and 48, compare the predictions with data from references 249 and 36. The data were contained in the data base used by General Dynamics to derive the ADAP correlations. However, the calculations shown in figures 47 and 48 were performed using the FORTRAN version of the program developed by Kuhn (ref. 307). The agreement is excellent.

Other examples are shown in figures 49 and 50 which indicate more clearly the limitations of the predictive method. In figure 49 is shown the comparison between predicted drag coefficients and experimental values as functions of nozzle pressure ratio for various Mach numbers on several of the circular-arc boattails studied in reference 92. The values of the parameters covered in these figures are listed in the following table.
The data from reference 92 for supersonic Mach numbers were included in the ADAP data base. The subsonic data from reference 92 were not. Therefore, the subsonic data constitute an independent check of the prediction method. Also, it is noted that the ADAP correlations include fineness ratios from 1.0 to 4.0 for subsonic flows and 1.0 to 2.0 for supersonic flows. Thus, the body with fineness ratio of 0.8 requires extrapolation from the main data base in order to predict the drag. Another feature of the data is the existence of an extensive region of separated flow on the FR = 0.8 and 1.0 afterbodies as indicated by oil-flow studies discussed in reference 275. The agreement between the prediction and the data shown in figure 50 is good for the supersonic case for fineness ratios above 1.0. The predicted drag is generally somewhat lower than the data for all fineness ratios for the subsonic cases.

A better evaluation of the quality of the prediction can be seen in figure 50, where the drag coefficients are plotted as functions of Mach number for a single nozzle pressure ratio of 3.0. The first notable fact that can be observed from this comparison is that the ADAP does not predict the transonic drag rise. For subsonic flow best agreement for all the bodies occurs around a Mach number of 0.8. This is to be expected since the ADAP data correlation emphasized $M_{\infty} = 0.85$ data with
some data between $M_\infty = 0.4$ and 0.85. It is not clear whether the lack of agreement shown in figure 50(a) for the $FR = 0.8$ case can be attributed to the separation observed on that body or simply to the fact that such data were not included in the data base. Some improvement could probably be achieved in these predictions if the subsonic and transonic data from reference 92 were incorporated into the correlations.

6.2.2 Twin-engine configurations

All the remaining methods to be examined employ as a basic correlating parameter a projected-area-weighted average slope of the afterbody area distribution known as the Integral Mean Slope (IMS). This parameter has been found to correlate both twin-jet and single-jet parametric drag data (ref. 118).

The first method to be examined was developed by the Boeing Company and Pratt & Whitney Aircraft. The method is based on correlation of a large amount of data. However, it is limited to Mach numbers below 0.95. The second prediction method was developed by the Lockheed California Company (Calac) under an extensive program in which 92 aircraft configurations were tested over a wide range of Mach numbers, nozzle pressure ratios, and configuration variables. The third method, developed by the Naval Air Development Center, extends the Calac method to a broader base of experimental data by including all available afterbody drag data for twin-engine "fighter-type" configurations in a data base. A statistical method is then applied to develop a correction for the Calac correlation.

6.2.2.1 ESIP twin-afterbody drag prediction method

The ESIP prediction method (ref. 24) deals with the pressure drag of the aft fuselage downstream from the maximum cross-sectional area for subsonic Mach numbers up to 0.95. The correlation covers configurations with horizontal interfairings. There is reason to believe it is also
applicable to other interfairing concepts or even to single-engine configurations but data are not now available to verify this. The correlation accounts for the effects of either a single vertical stabilizer or twin vertical stabilizers extending radially from nacelle centerlines.

The correlation treats only the design-pressure-ratio case for which the nozzle-exit static pressure equals free-stream ambient pressure. Pratt & Whitney has developed a jet-plume parameter to help handle off-design conditions (ref. 113).

The ESIP correlation is based on the integral mean slope of an equivalent body of revolution representing the afterbody. However, in this correlation, an attempt has been made to account for the effects of separated flow. In the development of the method, studies indicated that the aft-end drag coefficient should correlate well with the IMS parameter. Other data, however, showed that the correlation broke down for afterbodies whose area plots involved regions of steep slopes. A modified IMS parameter called IMS was then developed to avoid a sensitivity to afterbody contours in regions which were likely to have separated flow. The IMS approach is based upon specifying a maximum slope of the nondimensional area distribution which can be used in the IMS calculation. The maximum slope is substituted for the real slope at each step of the IMS calculation for which the real slope exceeds the maximum slope. The best correlation was obtained by making the maximum slope a function of Mach number. The resulting correlation is shown in figure 51. Here \( \hat{C}_D \) indicates the pressure drag coefficient based on projected frontal area of the afterbody. The data represent a wide range of aft-end geometries and nozzle types and both single- and twin-vertical-stabilizer configurations. The correlations for the two tail types were found to be nearly identical except that the single-vertical-tail data indicated a lower drag with a level shift of 0.006 in \( \hat{C}_D \). Thus, the two correlations were combined into a single correlation in terms of \( \hat{C}_D \).
where $\Delta C_D = 0$ for centered-twin-vertical stabilizer configurations and 0.006 for single-vertical-stabilizer configurations.

Further studies of the correlation indicated that all of the data varied almost precisely as the $IMS_T$ parameter raised to the 2.77 power. The $IMS_T$ dependence was divided out to obtain a drag parameter which was a function of Mach number only as shown in figure 52.

The overall procedure for using the correlation in a prediction method for subsonic afterbody drag of twin-engine configurations is illustrated schematically in figure 53. The first step is to input the nondimensional afterbody area distribution. The area distribution together with the maximum slopes are then used to calculate the IMS values as a function of Mach numbers. The remaining calculations are then the determination of the drag parameter from the basic correlation curve and the inclusion of the appropriate drag increment for the tail type.

The limits of this correlation are as follows: The correlation does not extend below Mach numbers 0.7 because no data were obtained there in the test program. However, extrapolation to Mach zero along a constant equal to the value of Mach 0.7 appears to be a reasonable approach due to the lack of a compressibility effect at Mach 0.7. Extrapolation in the other direction to Mach numbers greater than 0.95 would not be possible. Similarly, the data base for the correlation includes models with $IMS_T$ values as large as about 1.1. The method should not be considered applicable to models with larger values of $IMS_T$. Finally, the correlations are based on a narrow range of afterbody shapes, with all data obtained with a wingless forebody. Since data presented in reference 45 indicate a rather significant effect of a wing on total afterbody drag, application of the ESIP correlation to complete configurations should be done with caution. While the method may not predict the absolute level of drag accurately for a given configuration, it may be useful for a preliminary design analysis in comparing the relative merits of several candidate configurations.
6.2.2.2 Calac twin-nozzle/afterbody drag performance method

The twin-nozzle/afterbody drag performance method is described in reference 50, which is based on work described in references 44 and 46. It consists of a computer program for predicting twin-nozzle/afterbody drag and internal nozzle performance for fighter-type aircraft having twin buried engines and dual nozzles. The program is a revised version of that described in reference 45 and is capable of generating the installed thrust-minus-drag data required for conducting mission analysis studies of aircraft of this type. The configuration variables which can be analyzed include:

1. Nozzle type - convergent flap and iris
   convergent-divergent with and without secondary flow
   plug, both shrouded and unshrouded

2. Nozzle lateral spacing - narrow (S/D = 1.25)
   intermediate (S/D = 1.625)
   wide (S/D = 2.0)

3. Interfairing type - horizontal wedge
   vertical wedge

4. Vertical stabilizer type - single vertical for narrow, intermediate, and wide spacings
   twin vertical for wide spacing

The performance prediction methods are based almost entirely on empirical correlations. Correlations used in conjunction with one-dimensional flow relationships are employed for the prediction of the nozzle thrust and discharge coefficients, and correlations of wind-tunnel test data are employed for the effect of nozzle pressure ratio and flow separation on both internal and external nozzle surfaces. A schematic diagram of the method is presented in figure 54.
Boattail pressure drag. Subsonic flow: For subsonic external flow, the prediction method begins with a basic correlation of the pressure drag on a twin afterbody aft of a certain reference station. For the wind-tunnel models used in developing the correlation, that station was the metric break station separating the part of the model on which aerodynamic loads were measured from the rest of the model. It is suggested that the appropriate station to be used for analysis of a new configuration is the axial location of the wing trailing edge. Input data for the computer program include quantities to account for the drag on the portion of the afterbody between this reference station and the maximum cross-sectional area station.

The boattail-drag coefficient referenced to the cross-sectional area at the metric break station is computed from the following empirical correlation of data.

\[ C_{D_{MB}} = K_1 \left( \frac{\text{INSA}}{M_\infty} \right)^{2/3} \frac{\Delta p}{A_{mb}} + K_2 \frac{F_{ID}}{q_\infty A_{mb}} \]

\[ K_1 = \frac{C_{D_{MB}}}{\text{INSA}} \left( \frac{M_\infty}{\text{INSA}} \right)^{2/3} \]

\[ K_2 = \frac{\Delta D_{F_{ID}}}{} \]

The breakdown of Eq. (26) is as follows: The first term is the drag for the design pressure ratio, where the nozzle-exit static pressure equals the free-stream static pressure and a cylindrical exhaust plume is produced. The second term is the drag increment from the design pressure ratio to operation at a higher ratio.
A typical correlation for the design-pressure-ratio drag coefficient parameter, \( K_1 \), is shown in figure 55. Note that the equivalent base-to-metric-break area ratio is also a parameter in the correlation. The basic correlation relates the drag coefficient, the Mach number, and the integral mean slope IMSA of an equivalent body of revolution with the same area distribution as the configuration of interest downstream of the metric break station using a relationship based on transonic similarity theory. Other parameters which affect the correlation are the interfairing type, the tail type, and the nozzle type. The correlation shown in figure 55 applies for configurations with horizontal interfairings and a single vertical tail and is independent of nozzle spacing. A linear interpolation and extrapolation for equivalent base area ratios other than those presented in the figure is employed.

For convergent and convergent-divergent nozzle installations the drag parameter, \( K_2 \), which is the increment in drag from design-pressure-ratio operation to operation at a higher pressure ratio, is shown in figure 56 as a function of nozzle underexpansion loss. The drag increment, which is normalized by the ideal thrust, is dependent upon both the Mach number and the shroud-exit to metric-break area ratio. This correlation is a function of nozzle type and nozzle power setting.

Jet-off boattail drags can be computed using the correlation results typical of figure 57. In this case, the increment in drag from jet-off operation to operation at the nozzle design pressure ratio for various nozzle lateral spacings and for various Mach numbers are presented. The drag increment is presented in terms of an increment in drag coefficient referenced to the twin-nozzle shroud exit area and is correlated as a function of boattail trailing-edge angle \( \beta \) at the nozzle exit.

Supersonic flow: Boattail-drag coefficients based on maximum area for a supersonic external flow are computed from the following equation.
\[
C_D = C_{D_{EB}} \frac{\hat{C}_D}{C_{D_{EB}}} \frac{A_P}{A_m} + K_3 \frac{p_e - p_L}{A_{so}} \frac{A_{so}}{\rho \infty} \frac{p_{\infty}}{A_m} \frac{q_{\infty}}{A_m}
\]

where the first term is the jet-off drag and the second term is the increment in drag from jet-off to jet-on operations. The equivalent-body drag is obtained by entering the method-of-characteristics boattail-drag correlation curves presented in figure 58 with a Mach number and IMS. A subsequent correlation such as figure 59 is used to calculate the ratio of jet-off drag to equivalent body drag as a function of Mach number and vertical stabilizer type. For jet-on operation, \( K_3 \), which is the increment in drag from jet-off operation normalized by the product of the nozzle shroud external cross-sectional area at the nozzle exit and the difference between nozzle internal exit pressure and the local boattail surface pressure (assuming no flow separation) is obtained from another correlation shown in figure 60 as a function of nozzle mean-boattail angle. The mean boattail angle used is the mean angle over a distance equal to one-third the nozzle exit radius measured from the nozzle exit. This length was selected as being representative of the flow separation length. The local boattail flow properties are obtained from a method-of-characteristics solution. Thus, \( K_3 \) is the increment in drag from jet-off operation to operational conditions at which separated flow occurs due to plume effects. It has been observed that little or no separation occurs for \( (p_e - p_L)/q_L < 1.4 \). Therefore, the correlation results presented in figure 60 are restricted to pressure coefficients greater than 1.4.

Boattail friction drag.- The required input for computation of the boattail friction drag is the boattail length \( L_{BT} \), the wetted surface area \( A_{\text{wBT}} \), and either the momentum thickness \( \theta \) at the start of the
boattail or an effective flat plate length \( L_{\text{eff}} \) upstream of the start of the boattail. With these inputs, an average boattail skin friction coefficient is computed by use of Sivells-Payne correlation (ref. 161) which, when combined with the wetted area, yields the friction drag as discussed below.

With an input momentum thickness at the start of the boattail the reference-length Reynolds number, \( R' \), is obtained by iterative solution of the following equation

\[
R'_{e_1} = \frac{\mu'_{e_1}}{\mu_1} \frac{(0.044 R'_{e_1})}{(\log_{10} R'_{e_1} - 1.5)^2}
\]  

(28)

where the primed quantities denote values evaluated at the reference temperature, \( T' \), which is obtained from the following equation

\[
T' = T_1 + 0.035 M^2 + 0.45 \left( \frac{T_{aw}}{T_1} - 1 \right)
\]  

(29)

where

\[
T_{aw} = T_1 \left[ 1.0 + \left( \frac{y-1}{2} \right)(0.89)M^2 \right]
\]  

(30)

If an effective flat plate length upstream of the boattail is input, the reference Reynolds number is obtained from the following equation
The local skin friction correlation equation taken from reference 161 is

\[
R'_e = \frac{\rho'_1 U_\infty}{\mu'_1} L_{\text{eff}}
\]  

(31)

The local skin friction coefficient at the end of the boattail is computed in a manner similar to that described above except that the length employed in the computation of the reference-length Reynolds number is

\[
C_{fL} = \frac{0.088 (\log_{10} R'_e - 2.3686) T_1}{(\log_{10} R'_e - 1.5)^3 T_1}
\]  

(32)

If the momentum thickness Reynolds number is input, the effective flat plate length at the start of the boattail is computed as follows:

\[
L_2 = L_{\text{eff}} + L_{\text{BT}}
\]  

(33)

The skin friction drag coefficient based on maximum area is

\[
C_{Df} = \left( \frac{C_{fL} + C_{fT}}{2} \right) \frac{A_{WBT}}{A_m}
\]  

(35)
Annular base drag.- The annular base pressure for a subsonic external flow is computed from the following modification (developed in ref. 44) of the Brazzel-Henderson base pressure correlation (ref. 66):

\[
\frac{p_b}{p_\infty} = \frac{0.9 + 0.0167(R_{mf})}{0.94 + 0.06(A_{so}/A_m)}
\]

(36)

where \(R_{mf}\) is the nozzle-exit to free-stream momentum ratio, defined as

\[
R_{mf} = \frac{\left(\frac{\dot{m}V_e}{\dot{m}V}\right)}{\left(\frac{\dot{m}V_e}{\dot{m}V}\right)_\infty} = \frac{\gamma_e p_e A_e M_e^2}{\gamma_\infty p_\infty A_m M_\infty^2}
\]

(37)

For a supersonic external flow, the following base pressure correlation developed by Brazzel-Henderson is also employed.

\[
\frac{p_b}{p_\infty} = \left[\frac{T_e}{T_e^k}\right] \left[\begin{array}{c}
3.5 \\
0.5 + 3.0 A_{so}/A_m
\end{array}\right] \left[0.19 + 1.28 \left(1 + R_{mf} - 0.047(5-M_\infty) \right) \right] \left[2 \left(\frac{\Delta X_e}{D_m}\right) + \left(\frac{\Delta X_e}{D_m}\right)^2\right]
\]

(38)

The first term on the right side of Eq. (26) normalizes the jet temperature to the jet temperature of a sonic nozzle. The second term corrects for boattail effects, and the third term is a correlation based on the ratio of nozzle-exit momentum flux to free-stream momentum flux. A nozzle-position (relative to the end of the boattail) correction is obtained by the fourth term.

Comparison with data.- An independent investigation (ref. 41) indicates that the twin-nozzle afterbody-drag performance method reasonably predicts the trends and absolute levels of data for a con-
figuration similar to those used for the original Calac correlation. The evaluation model was a twin-jet air superiority fighter tested in the AEDC PWT 16T facility.

One significant difference exists between the model and the model as defined for the computer program. The model has intermediate spacing of the nozzles and twin vertical tails, whereas for intermediate nozzle spacing, the computer model is limited to a single vertical tail. No accurate assessment has been made of the effects of this difference. It is known, however, that tail surfaces do significantly affect the afterbody flow field.

Another difference between the model and the model defined for the computer program is that the model nozzle lateral spacing falls between the values for narrow and intermediate spacing. The empirical data chosen for the calculations were those closest to the model configuration. The empirical correlations are based on data that is limited in the 0.9 to 1.2 Mach number range. The predicted afterbody drags, therefore, for this Mach regime should be used with caution. Figure 61 provides a comparison between the measured and predicted values of drag coefficient for the cruise nozzle at nozzle pressure ratios of 3.3 and 4.9. Excluding the Mach 0.9 to 1.2 range, the predicted values of drag coefficient are in reasonable agreement with the measured values. A comparison is made of the measured and predicted drag for the reheat nozzle at nozzle pressure ratios 4.2, 5.2 and 7.8 in figure 62. The results are similar, but especially good at a nozzle pressure ratio of 4.2. This is not unexpected since the problem of aft-end closure has been lessened in the reheat mode. Reference 41 concludes that a need exists for more test data to fully evaluate this prediction technique and an extension of the empirical data base to include closely spaced nozzles with dual vertical tails. This comparison is discussed further in the next section where the Calac method is compared with an extended version of the method.
6.2.2.3 NADC prediction technique for twin jet fighter-type aircraft

Description.- The NADC prediction technique (ref. 143) is an extension of the Calac technique. The method was developed by expanding the Calac aft-end drag data base to include data from models of other existing twin-jet fighter-type aircraft. Parametric investigations were then conducted to determine if the existing Calac aft-end drag parameters were all inclusive and could account for the inherent drag variations of the added fighter-aircraft models. Finally, a new drag parameter dependency relationship was derived for the important parameters. The derived relationship is

\[ C_{D_{\text{NADC}}} = C_{D_{\text{CALAC}}} + \sum_{i=1}^{N} \alpha_i X_i \]  

where \( \alpha_i \) are coefficients determined by the statistical analysis and \( X_i \) are the various parameters including NPR, and various similarity relations, and combinations of area ratios, Mach number and Reynolds number listed in Table X. The extended correlation was found to provide a significant error reduction from the Calac correlation. An asset specifically designed into the method is that the data base can be continually expanded to include additional twin-jet aft-end drag data.

The limitations of the NADC prediction technique are that the method is dependent on wind-tunnel data quality and, since it is based on the Calac method it is applicable to the same limited class of configurations. Also, the method does not account for wind-tunnel to full-scale flight Reynolds number variations.

The Calac data base consisted of data from wind-tunnel models which had the horizontal and vertical stabilizers mounted separate from the
afterbody force balance so that only the tail interference effects on the afterbody drag were directly measured. Some of the data considered in the NADC correlation procedure had the tails mounted directly onto the afterbody so that the measured drag included the total drag of the tails. Thus, one of the steps in the NADC data-correlation procedure which may produce some error in the predictions results from the necessity of subtracting from the experimentally measured afterbody drag an estimated value of drag of the tails in order to include only the tail interference effect in the afterbody drag correlation.

Evaluation of the prediction method.- In order to evaluate the accuracy of the prediction method as a design tool, some afterbody drag predictions were compared with experimental results for two configurations that were not included in the correlation data base. One of the configurations (ref. 32) was that analyzed previously by Everling (ref. 41) who compared the data with predictions made by the Calac version of the program. That configuration had tails mounted on plates attached to the forebody, separate from the afterbody force balance. Thus, only the tail interference effects should be present in the data, and the predictions should be directly comparable. The other configuration (ref. 262) is more difficult to compare since data are available only for the cases of tails on or tails off. No data were obtained for afterbody drag in the presence of the tails but not including their drag for direct comparison with the prediction. The results of the tests described in reference 262 provide a qualitative evaluation of the prediction method.

The independent investigation by Everling (ref. 41) of the data of reference 32 indicated that the Calac twin-nozzle/afterbody drag performance method reasonably predicted the trends and absolute levels of data for a configuration similar to those used for the original Calac correlation. The evaluation model was a twin-jet air-superiority fighter tested in the AEDC PWT 16T facility.
As mentioned previously, some of the data used by the NADC in extending the experimental data base had the tails attached to the afterbody balance. The tail profile drag for such cases was estimated and subtracted from the measured drag in order to produce afterbody drag data consistent with that in the Calac data base. The accuracy of such adjustments is unknown.

Figure 63 shows a comparison between the measured and predicted values of drag coefficient. Figures 63(a) and (b) show the comparison for the cruise nozzle at nozzle pressure ratios of 3.3 and 4.9. Predicted values are shown for both the basic Calac program and the NADC modified program. The NADC adjustment gives a slightly improved prediction for the mid-transonic range of Mach numbers while the Calac prediction is better for low and high Mach numbers. Similar results are achieved for the reheat nozzle as shown in figures 63(c), (d), and (e).

Figure 64 shows the drag coefficients on a wind tunnel model of the F-18 aircraft, described in reference 262. Those data are similar to the previous data in the sense that the model has twin tails and narrow-to-wide nozzle spacing, whereas the Calac data base provides for twin tails only for the wide nozzle spacing.

One aspect of the tests reported in reference 262 was the investigation of the effect of removing the tails from the model. This effect is indicated in figure 64 where the drag with tails removed is seen to be significantly lower than with the tails attached to the afterbody (and to the afterbody balance) for both the cruise nozzle at $M_{\infty} = 0.6$ and 0.9 and the maximum reheat nozzle at $M_{\infty} = 1.6$ and 2.0. The prediction method, which is based on data that includes, presumably, only the tail interference effects (and not the drag of the tails themselves), would be expected to yield a drag value nearer the tails-off value than the tails-on value. As shown in figure 64 that expectation is met by the Calac prediction with the exception of the case for $M_{\infty} = 0.9$ for the cruise nozzle (fig. 64(b)), but the NADC prediction is not so consistent.
A more exact evaluation of the data to determine the actual interference effect of the tails on the afterbody drag requires an accurate evaluation of the tail profile drag and is beyond the scope of the present investigation.

The comparisons shown in figure 63 indicate that for configurations that closely resemble those in the Calac data base, the Calac prediction may be better than the NADC prediction except for low supersonic Mach numbers. The comparisons shown in figure 64 provide no quantitative basis for choosing either method. It would be expected that the drag for configurations that more closely resemble those that were added to the data base by the NADC would be more accurately predicted by the NADC version of the program.

6.3 Theoretical Prediction Methods

In general, the interference flow fields associated with installation of the propulsion system in the airplane are dominated by viscous effects such as boundary-layer separation and jet-exhaust plume entrainment. Existing analytical attempts to predict these interference effects are limited to specific effects for simple configurations such as isolated axisymmetric nozzles. Generalizations are often made from these simple shapes to more complex shapes using the concept of an equivalent body of revolution. This technique is found to be unsatisfactory except for small deviations from symmetry. In this section, the general problem of predicting afterbody flow fields will be described followed by a discussion of the status of the theoretical prediction technology.

6.3.1 Description of problem

The usual approach for calculating the flow over boattails is to use the technique of dividing the flow field into a number of analytical regions. A typical breakdown of these regions is shown in figure 65. Included are an inviscid external flow (region I), inviscid jet plume
(region II), and viscous region (region III), which includes boundary layers on the boattail and inside the nozzle and the mixing layer between the jet and the external flow. The viscous region may be further divided into attached boundary layer and separated boundary layer regions. Several calculative techniques have been developed which show promise as engineering-type prediction methods. At the present time, no generalization of analytical prediction methods to non-axisymmetric nozzles, or to full aircraft configurations has been made.

With regard to the exhaust plume, two physical effects have been identified by Bergman (ref. 6). The first is a displacement outward of the external flow streamlines caused by the shape of the plume. Outward displacement of the streamlines results in stronger flow recompression on the boattail and has a beneficial effect on the drag, as shown in Figure 66. The shape of the exhaust plume is a function of nozzle pressure ratio for fixed free-stream conditions. The second effect, entrainment, begins when the exhaust plume velocity is approximately equal to free-stream velocity. This effect increases boattail drag by increasing the velocity of the flow over the boattail and effectively lowering the boattail pressure. The net effect on boattail drag is the sum of the two effects.

Increasing exhaust jet temperature has been found to decrease the drag for fixed nozzle pressure ratio. This effect is caused by greater spreading of the hot plume than the cold plume. The effect of exhaust plume temperature on entrainment is not known.

It must be noted that many of the methods to be discussed in this section contain certain empirical features, and therefore might be considered semi-empirical rather than theoretical. The definition of a theoretical prediction method is considered to include methods wherein empirical techniques may be used for certain specific features such as prediction of the separation point location, but the basic calculative method is a solution of the theoretical governing equations of the flow.
field. The term empirical prediction methods is thus reserved for those methods based primarily on correlation of experimental data.

6.3.2 Status of theoretical prediction technology

A summary of the status of current technology for predicting drag on axisymmetric nozzle boattails is presented in Table XI. In that table, the factors that must be accounted for are listed in the left hand column, the pertinent reference numbers from the afterbody drag reference list are listed at the top and an X is placed in the appropriate box denoting whether the prediction method accounts for that factor. A blank box in the column for a certain reference means that reference does not treat the corresponding factor. The table reveals that several methods exist for accounting for boundary layers, including separated boundary layers, and several methods exist for accounting for plume shape. Exhaust-plume entrainment is not accounted for in an adequate manner by any existing techniques. Those methods listed as accounting for entrainment are first approximations and have not developed a viable entrainment model.

The theoretical prediction methods fall naturally into three basic categories depending on whether the flow conditions are subsonic, transonic, or supersonic. Within each of those categories another major subdivision is between those methods which treat only attached flow and those which also treat separated flows. The individual methods in each of these categories and subdivisions will be discussed in the following sections.

First, however, it is noted that a program was initiated by the NASA Langley Research Center in June 1976 to assess the state-of-the-art for predicting pressure distributions and drag of boattail nozzles. The objective of the program was to compare available analytical and empirical methods with one set of experimental data so the relative merits of the various methods could be easily assessed. The results were
subsequently distributed to interested parties (ref. 8.10). The prediction methods that were applied to the prescribed data set included those described in references 136, 200, 210, 211, 212, 217, 227, and 294. Other investigators have also recently used the same data for comparison (refs. 264, 300). All of the original groups of methods predicted the afterbody drag poorly for subsonic and transonic flows for both an afterbody with no separation and for one with extensive separated flow. The one method that was applicable to supersonic flows, that of Holst (ref. 210), was in good agreement with experimental integrated pressure drag on an afterbody with extensive separation. The prediction methods will now be discussed individually in the following sections.

6.3.2.1 Subsonic afterbody drag calculative methods

The most complete theoretical method for subsonic axisymmetric boattail drag with attached flow is a method produced by the Lockheed California Company (ref. 48).

The method predicts the drag of arbitrary axisymmetric boattail contours through use of the subsonic potential flow method of Smith and Pierce (ref. 164) combined with approximate exhaust-plume and integral-boundary-layer calculative methods. Mutual interaction effects between the external flow and the exhaust flow and between the external flow and the boundary layer are treated. The computer program is equipped to handle convergent, convergent-divergent, and plug-type nozzles. The method is limited to subsonic free-stream Mach numbers below that for which supercritical flow would occur over the boattail, and to configurations for which no boundary-layer separation occurs. Jet-plume mixing is not treated.

Comparisons with data shown in figures 67-69 indicate the method produces good agreement with the data on the conical afterbody of reference 134 when the boundary layer is accounted for (fig. 67). For convergent-divergent and plug-nozzle-boattail configurations (figs. 68 and 69), the pressure distribution trends are properly predicted, but the
absolute pressure levels are not predicted very well, especially in the recompression region near the nozzle-boattail trailing edge.

Another method for calculating the subsonic drag of axisymmetric boattails is that of Rom and Bober as described in reference 99. That reference accounts for the boundary layer on the boattail but does not account for jet plumes. Instead, an approximate method is used to account for the flow in the base region, corresponding to jet-off conditions. Separation of the boundary layer is assumed to occur only at the boattail base. Separation on the boattail is not treated.

A calculative method including provision for separation on the boattail was developed by Presz (ref. 253, 80). In that method, the separated region is modeled as a conical dividing streamline surface placed between the separation point on the boattail and the point of reattachment on a solid cylindrical sting. An empirical approach based on an integral boundary layer method is used to determine the location of the point of separation and the angle of divergence of the dividing streamline surface from the boattail surface. In Presz's method, the subsonic potential flow theory of Smith and Pierce is then used to calculate the pressure distribution on an equivalent body consisting of the real body and the conical separation surface. Comparison with data indicates the approach gives a reasonable approximation for the afterbody pressure distribution and drag. Typical comparisons between the theory and data are shown in figures 70 and 71.

An improved version of the method of reference 253 was developed by Presz (refs. 264, 265) by relaxing the requirement that the separation surface be conical and using the boundary-layer displacement thickness to modify the body shape. Before separation and after reattachment, the modified body shape is the actual body contour plus the displacement thickness. In the separated region, the shape is determined by adding the displacement thickness to the dividing streamline determined by a discrete control volume analysis. A jet-plume entrainment model is also
included to account for changes in the inviscid flow field due to effects of jet entrainment on the dividing streamline. Comparisons with experimental pressure data on several circular-arc boattails (figs. 72-76) indicate that the method accurately predicts the pressure distribution for bodies with stings on which no separation occurs (fig. 72) or for moderately separated flows (fig. 73). For sting-mounted models with more extensive separation, the comparison is not so good (fig. 74). For models with jets and separated flow, the calculative method of reference 264 appears to be unable to predict the constant pressure that occurs in the separated region (figs. 75 and 76). The errors in the pressure predictions result in fairly large errors in drag as shown by the drag-coefficient values listed in figures 72-74.

Presz's results from reference 253 were used by Reubush and Putnam in reference 136 as a basis for a correlation which gives the divergence angle of the dividing streamline surface as a function of the Mach number at the separation point. The conical separation surface is then used as the solid boundary and an attached boundary layer is calculated for the entire length of the augmented body plus the sting. The subsonic potential flow method of Hess and Smith (ref. 162) is used to calculate the external flow. Comparisons with experimental data indicate good results can be achieved with this technique if an accurate separation prediction is used.

Incorporation of exhaust-plume entrainment into subsonic afterbody flows is discussed in references 297, 298, and 299 where a modular approach is taken to couple the viscous and inviscid processes. Separation of the flow over the afterbody is not considered.

6.3.2.2 Transonic afterbody drag calculative methods

The theoretical methods available for transonic speeds fall naturally into the same two subcategories as for the subsonic methods. Namely, those in which boundary-layer separation on the boattail is treated and
those in which it is not. Furthermore, the lack of accurate shock-wave-boundary-layer interaction theories effectively limits these prediction methods to subcritical free-stream Mach numbers or to Mach numbers such that only very weak shocks occur on the body.

Grossman and Melnik (ref. 51) describe a method in which the drag for arbitrary axisymmetric boattail contours is calculated including mutual interaction effects between the external flow, the jet plume and attached-boattail boundary layer. A calculative method like that of South and Jameson used for the external flow is applicable to both purely subsonic and transonic flows. However, the computer programs for the jet plume needed for application of the method are not generally available.

A more readily available method is described in reference 34 by Chow and Bober, which accounts for the mutual interaction between attached boundary layers and subsonic or transonic external flow, but approximates the jet plume as a solid boundary. The full transonic potential equation is used, including boundary-layer effects, yielding good agreement with the experimental pressure distribution for the conical boattail of reference 141. Some examples are shown in figure 77.

The method of references 217 and 252 is similar to that of reference 34 in that the full transonic potential equation is used in conjunction with a boundary-layer method for attached flows. In addition, a simplified plume entrainment model is described which yields qualitatively satisfactory results.

In reference 43, the Spreiter-Alksne method of local linearization (ref. 165) is used with a turbulent-boundary-layer method and the method of characteristics to study plume-shape and afterbody-shape effects. The method is limited to slender bodies and separation is not accounted for.

Several methods have been developed for transonic flows with separation. In reference 212, Yaeger describes a method which is a modified
version of that described by Presz and by Presz and Pitkin (refs. 253, 80). It is applicable to bodies with stings and to slightly supercritical flows for which no significant shock-wave-boundary-layer interaction occurs. Some insight into the effects of strong shock-wave-boundary-layer interactions is provided by an analytical study described by Wilmoth in references 211 and 274 where the applicability of Presz's method of reference 253 is shown to be limited to free-stream Mach numbers below about 0.7. Results of that study are shown in figure 78 where the variation of the discriminating streamline separation angle is found to depend strongly on the local Mach number at separation. This also would appear to limit Yaeger's method (ref. 212) to the same Mach-number range since Yaeger employs similar analytical techniques to those used by Presz and restricts his analysis to flows with negligible shock-wave-boundary-layer interaction.

In references 27 and 227, another type of approach is used by Calarese to study the effects of mass and energy injection on boattail drag. An iterative procedure is used whereby viscous effects are added to the inviscid solution of the nonlinear small-perturbation equation. Jet effects are accounted for through empirical correlation and boattail separation is treated with a Korst (ref. 160) base-flow type of analysis. The method is shown to provide good agreement with data for attached flows but only fair agreement for separated flows.

Another different approach is described by Cosner in reference 209. In that approach, separated boundary layers are calculated using an empirical technique to extend the calculation past separation and reattachment points. Jet-plume mixing is also handled in an approximate way. Results presented indicate the method, still in a preliminary state of development, is not very accurate for flows with extensive separated regions.

In the method of references 218, 300, and 301, Kuhn has employed many of the principles developed in several of the other methods. The full-
transonic-potential-equation solution method of South and Jameson is used allowing calculation of flows over non-slender bodies and steep boattail angles. An integral-boundary-layer approach is used whereby both attached and reversed flows are calculated for bodies with solid stings. The separation and reattachment points are calculated as a part of the interaction between the boundary layer and the inviscid flow. An entrainment model is included to account for exhaust-jet effect. Comparisons with data for separated and unseparated flows indicate the method provides good predictions of the afterbody pressure distribution and drag (figs. 80 through 87) even into the region of transonic drag rise, although predicted drags for low-subsonic unseparated flows (fig. 85) are high.

The jet entrainment model used in reference 300 appears to provide fair agreement with data although the accuracy of the drag prediction seems to be a function of the free-stream Mach number as well as the nozzle pressure ratio (figs. 86 and 87).

6.3.2.3 Supersonic afterbody drag calculative methods

Supersonic afterbody drag methods are generally of three types. First are methods applicable to predicting boattail drag with jet plumes or solid plume simulators. Next are methods for predicting base drag. Finally, some methods are available for calculating optimum (minimum-drag) boattail shapes. The first two types consider viscid-inviscid interactions while the third type of method considers only inviscid flows. No comprehensive theoretical method has been found which includes both boattail and base drag.

For calculating afterbody boattail pressure drag in supersonic flow, two methods are described in references 49 and 210. In reference 49, Glasgow, et al., couple the method of characteristics with an attached-boundary-layer method supplemented by approximate methods of accounting for either plume or shock-induced separation based on the correlations of
Bonner and Nixon (ref. 26) and Brazzel and Henderson (ref. 66). Predictions are accurate for cases with no separation, but the approximate theories do not adequately account for the complex interference effects accompanying flow separation on the boattail. A computer program for the CDC 6600 computer is available from AFFDL.

Holst, in the method of reference 210, employs a more fundamental approach by solving the compressible Navier-Stokes equations in the vicinity of the boattail for axisymmetric afterbody-boattail configurations with solid-sting plume simulators. Comparisons with data for pressure distributions on boattails with extensive separated regions are good.

Another method employing a solution of the Navier-Stokes equations was demonstrated by Mikhail, Hankey, and Shang (ref. 267). Their method was applied to an axisymmetric boattail with jet exhaust. Although the method is not a fully developed design technique, it demonstrates that such a complex flow can be computed successfully, and provides a design engineer with good estimates for the surface pressure distribution and skin-friction forces. The method was not demonstrated for flows with separation on the boattail.

A Navier-Stokes solution technique that includes separated flows is described by Forester in reference 296. Progress is reported on development of a computer program with emphasis on computational efficiency and accuracy.

Many methods exist for calculating base-region flow properties. A review of various methods, including those of Korst (ref. 160) and Alber (ref. 184) is presented by Peters and Phares in reference 232 where an analytical model for planar and axisymmetric supersonic turbulent near-wake flows is presented. That model couples an integral form of the boundary-layer equations and the rotational method of characteristics. The base pressure is obtained by an iterative procedure. The method is
designed to handle base bleed of a gas different from the outer-stream gas but the calculative technique has only been developed for single gas flows. The model is shown to adequately predict the effect of free-stream Mach number and initial boundary layer on the planar base pressure. Axisymmetric base pressure and flow-field structure are reasonably well predicted for free-stream Mach numbers greater than 2.0, but the turbulent transport model used yields only fair results for Mach numbers less than 1.7. The effect of base bleed on the axisymmetric base pressure is well predicted.

Another base flow analysis is presented in reference 117. In that method, Dixon, et al., extend Korst's theory (ref. 160) to treat base flow on an axisymmetric afterbody with a single operating exhaust nozzle. The flow around the afterbody and in the jet is supersonic with turbulent axisymmetric mixing occurring along the separated flow boundaries. Initial boundary layers are neglected and an empirical spread-rate parameter is used. A complete description of the base flow is obtained from the analysis permitting the prediction of base pressure and other flow parameters of interest. Results obtained from a parametric study illustrate the influence on the base drag of free-stream Mach number, exhaust-jet total pressure, and base, boattail and nozzle geometry.

Another more advanced application of Korst's method is presented in reference 220, where Bauer and Fox apply Korst's method to estimate the bulk base flow properties of nozzle-afterbody configurations with a supersonic jet and supersonic free-stream separated by a finite area base. Initial boundary layers are included as well as dissimilar thermodynamic properties of the two streams and a third base-bleed gas. The inviscid flows are computed by the method of characteristics. The turbulent-mixing analysis uses the turbulent-kinetic-energy method. Comparisons with data indicate the method tends to overpredict the base pressure due to use of too small a mixing rate.
Methods for determining optimum body shapes are described in references 231, 106, 129, 146, 223, and 284. Reference 231 is a survey of the variational problems relative to wings, fuselages, wing-fuselage combinations and nozzles in supersonic, hypersonic, and free-molecular flows. Within this broad scope of applications, it is an important reference on the calculus of variations as applied to the kind of variational problems associated with optimizing afterbody shapes.

6.4 Experimental Data

In Table XII the parameters associated with each Primary Data Source of the afterbody/airframe interaction data base are listed. This table provides a means for selecting particular references from a group of references which have been chosen for an effect from Table II. Numbers appearing in columns 2 and 3 of Table XII refer to Notes to Table XII following the table. These Notes explain the various types of configurations and data presented in the Primary Data Sources. The Notes are very general with respect to test model and data presented, since a detailed list could entail an almost endless list of configurations and types of data. For example, a reference which is indicated as having tested a twin-engine aircraft configuration may have tested this configuration with tails metric, tails nonmetric, or tails removed. The more specific types of configurations tested are not indicated in Table XII, or in the Notes to Table XII, but could be found by further investigation of the particular reference. In the same manner, the types of data presented in the Notes are general, and the list is not intended to be all-inclusive. The types of data presented for each reference are those types which would be most helpful to someone investigating afterbody/airframe drag. Other types of data which are not directly associated with this subject are not included in the Notes. With respect to the afterbody/boattail/nozzle drag coefficient and pressure coefficient distribution, a special note should be made. This data category is used to indicate that data is presented for either the entire afterbody/boattail/nozzle section or for a single element of this section, such as the nozzle. Also, the drag
coefficients were obtained using drag balance measurements or by integrating pressures, and may or may not include base drag or friction drag.

It should also be noted that the afterbody section of the models tested was that part of the configuration either aft of the maximum area station or aft of the metric-break station. A dictionary of the shorthand configuration classification scheme used in columns 8 through 12 of Table XII is presented in Table XIII.

The usefulness of the experimental data base lies in the degree to which it can be used to extend the range of applicability of prediction methods, both theoretical and empirical. Much of the data listed has been included in one or more of the correlations discussed in section 6.2. The remaining data may be useful for estimating effects not included in the prediction methods. Two areas in which such extensions are particularly needed are discussed in the following sections. These are Reynolds-number effects and problems of scaling from wind-tunnel data to full-scale flight. The adequacy of the data base for prediction of these effects is discussed.

6.4.1 Reynolds-number effects on afterbody drag

No clear and definite conclusion can be drawn from the existing literature about the effect of scaling from wind-tunnel to full-scale flight Reynolds numbers. Tests conducted by NASA and by AEDC over a wide range of Reynolds numbers indicate that for isolated afterbodies (no interfering surfaces such as wings, tails, etc.) large effects are produced on pressure distributions over boattailed afterbodies but the compensating effect of increased expansion over one part of the body and increased compression over the rest of the body combine to result in little, if any, change in the net afterbody pressure drag. The high sensitivity of the calculated pressure drag to errors in the measured
pressure does not appear to be a problem since great care has been taken in the experiments to minimize such errors.

The difficulty in evaluating Reynolds-number effects, or more specifically wind-tunnel to full-scale-flight-scaling effects seems to be in interpreting apparently conflicting results. To date, several experiments have been performed in wind tunnels covering a wide range of Reynolds number from values characteristic of wind-tunnel models to values characteristic of full-scale flight. Those tests indicate no Reynolds-number effect or very slight effects for isolated axisymmetric afterbodies and for the same bodies with wings added. On the other side some tests in which measurements were made both on models in a wind tunnel and on a full-scale model in flight indicate very significant Reynolds-number effects (ref. 127) while other tests indicate wind-tunnel and flight data have similar levels of afterbody/nozzle drag for Mach numbers below the nacelle drag-divergence Mach number (ref. 268). For higher Mach numbers, the drag was found in reference 268 to be lower for flight conditions for wind-tunnel conditions.

The problem lies in isolating the effects of Reynolds number from those due to many other factors, such as: accurate geometrical duplication between wind-tunnel and flight-test models, surface conditions, attitude, control deflections, instrumentation accuracy, model support and tunnel-wall interference, turbulence, propulsion simulation, heat transfer, and possibly noise.

To date, no reliable wind-tunnel-to-flight correlations have been made for real configurations of the type of interest herein. The few tests that have been performed have produced inconclusive results due to unanswered questions. For example, Martens (refs. 68 and 215) described wind-tunnel and flight-test results for the F-15. The data are inconclusive with regard to Reynolds number effects due to the inclusion of ventral fins on the wind-tunnel model but not in flight. Also, the question of hot-jet simulation by cold-gas jets was not considered in the tests.
6.5 Concluding Remarks

Evaluation of the adequacy of the data base for the prediction of afterbody drag leads to the following conclusions:

1. Although a number of systematic studies resulting in the development of empirical methods have been reported, there remain a number of unanswered questions. Perhaps the most important is that of the applicability of the results to flight Reynolds numbers. The experimental data on which the empirical methods are based were gathered at the relatively low Reynolds numbers provided in wind tunnels and therefore may not adequately reflect the viscous effects at flight conditions. Since conflicting results have been published regarding the effects of Reynolds number, this problem needs attention.

2. Although the number of configuration variables that have been tested is sizable, very little data are available for two-dimensional exhaust systems. Moreover, most of the available experimental results cover only a small angle-of-attack range, not large enough to encompass the flight envelope of interest.

3. The application of modern theoretical methods to the prediction of afterbody drag has only recently been undertaken. Methods to account for boundary-layer separation and jet entrainment effects for the simplest of afterbody configurations are only now being developed. Methods to account for the complex interactions for real aircraft configurations with multiple exhaust systems have not been developed. In any event, the large computers and long run times required at this stage of development make them unsuitable for use in preliminary design studies.
The performance of an air induction system can be significantly influenced by the effects of flow produced by (or resulting from) other elements of the configuration, such as fuselage forebody, canards, etc. If the local flow ahead of the inlet differs significantly from a uniform flow at the free-stream Mach number, then the performance of the installed air induction system would be expected to differ from that which would exist with uniform inlet flow conditions. As part of the development of a rational method of predicting these effects, a knowledge of the flow field produced by forebody fuselage, canards, etc., is needed. The following is directed toward this first step, that is, development of a data base for forebody and forebody-wing effects on local flow conditions in the vicinity of wing-body configurations.

The data base for airframe effects on inlet "free-stream" conditions consists of experimentally determined values and theoretical and empirical methods for predicting local flow quantities in the vicinity of fuselage-alone or wing-body configurations. In the following sections, applicable experimental data are presented and methods for predicting local flow fields about fuselage-alone or wing-body configurations are described. The data base is then evaluated with respect to the requirements placed on the data base.

7.1 Presentation of Data Base

7.1.1 Experimental data

Table XIV describes the Primary Data Sources from Table I which present experimental data for airframe effects on inlet "free-stream" conditions. This table identifies the configuration about which
flow-field data was obtained, the range of free-stream parameters, and the types of effects studied.

Column one of this table identifies the reference number of the data report. Columns two, three, and four identify the test configuration, flow-field probe position and type of data obtained, respectively. Definitions of the symbols used in these columns are given in Table XV. A distinction has been made in these tables between a body of revolution and a fuselage with a canopy, so that an evaluation of the experimental data base with respect to "real" versus "simplified" configuration effects can be made. The probe-position column describes the locations at which data was taken. For example, $F_{s,a}$ indicates that data was obtained near the side of the fuselage and ahead of the wing, while $W_{sh}$ indicates that data were taken under the wing at positions aft of the wing leading edge. It is noted here that data obtained at positions which are more than a body diameter away from the fuselage surface are indicated with $W_a$ or $W_{sh}$. The definitions in Table XV of the type of data obtained are self-explanatory.

Columns five through eight of Table XIV identify the ranges of Mach number, unit Reynolds number, angle of attack and angle of sideslip covered for each experimental investigation. The remaining columns indicate various effects for which parametric studies may have been carried out experimentally. A check (✓) in any one of these columns indicates that data was taken to determine the effect of that parameter on one or more flow-field quantities. With respect to these effects, a few comments are made here. First, a check in the General Configuration Data column indicates that a large amount of data, or in some cases, all of the data, was taken for a particular configuration. Second, a check in the Protuberance column indicates that data showing the effects of strakes, canards, or missile components on flow-field quantities were presented. The effects can be due to the addition of these components to a given configuration or to a parametric change of one or more of these components.
7.1.2 Prediction methods

Empirical and theoretical methods from Table I which predict flow fields in the vicinity of body-alone or wing-body configurations are grouped in Table XVI according to the type of flow equations solved (unsteady Euler equations; linearized, compressible, steady potential flow equations; etc.) or the type of method used (transonic equivalence rule, far-field matching, etc.). Each method is then further categorized by the type of flow field for which the method is applicable (axisymmetric or three-dimensional), and the speed range (subsonic, transonic, supersonic) for which the method applies. With respect to the "type of flow field" category, two-dimensional methods, per se, were not included in the data base. However, several methods listed are applicable to both 2-D and axisymmetric configurations (refs. 312, 314, 324), and a few 3-D methods involve solving 2-D equations after assumptions are made about the flow (refs. 34, 292).

Table XVII presents the major features of each method shown in Table XVI. This table is arranged by groups in the same manner as was done in Table XVI. This allows one to compare methods within the same group with respect to the basic assumptions, numerical techniques, body modeling techniques, etc. Details concerning particular aspects of methods whose main features are similar or identical (refs. 151 and 306, or 152 and 305, for example) are not presented in this report, since such a task is beyond the scope of the present work. The last name of the first author of a reference is also included after each reference number in Table XVII.

It should be noted at this point that while a large number of methods for predicting forebody and forebody-wing effects are listed in Tables XVI and XVII, these tables surely do not include all methods for predicting such effects. An attempt has been made to include representative examples for each of several groups of methods, and to include the "best known" and most up-to-date methods within each group. Also, as was noted
in section 5.1.3, an analysis of various boundary-layer methods is not presented in this report, although some methods include boundary-layer techniques (e.g., refs. 292, 312, 314, 374, 383, 393) as a part of their solution.

7.2 Evaluation of Data Base

7.2.1 Requirements

The local flow field entering an installed inlet is dependent on an array of parameters. These parameters include nose and fuselage geometry (shape and length), wing geometry (leading edge sweep, flap deflection and thickness), canopy size and shape, protuberances (strakes, canards, and missile components), inlet position (side mounted, fuselage-shielded, or wing-shielded), and free-stream Mach number, angle of attack and angle of sideslip. In the preliminary design process, studies of the effects of several geometric parameters on the flow field generated by each configuration are made over a range of free-stream parameters corresponding to various aircraft missions. The main thrust of these studies is usually to define the configuration for which forebody or forebody-wing effects on local inlet flow-field quantities are most favorable, thereby resulting in high inlet total pressure recovery and low inlet distortion, for a variety of missions or a specified mission. Therefore, as previously discussed in section 5.2.1 with respect to inlet external drag, the data base for forebody and forebody-wing effects on inlet "free-stream" conditions must also satisfy the requirements of breadth, ease of application, and accuracy.

In the following sections, the experimental-data and prediction-methods data base for forebody and forebody-wing effects on inlet "free-stream" conditions is evaluated with respect to these requirements. As previously mentioned in the discussion of the inlet and afterbody drag data bases, the accuracy of all the assembled experimental data has not been evaluated.
7.2.2 Experimental data

The manner in which forebody and forebody-wing effects are often studied experimentally is by starting with a "basic" configuration and perturbing about this configuration over a specified range of free-stream Mach number, angle of attack and angle of sideslip. The "basic" configuration is arrived at through configuration definition studies for which certain mission requirements must be met.

A good example of this kind of study is presented in reference 10 (Project Tailor-Mate). After initial configuration-selection studies were made based on specific mission requirements, five "basic" configurations were chosen for further study. These configurations included two fuselage side-mounted inlet designs (two-dimensional, inlets), and two under-wing inlet designs (two-dimensional and axisymmetric inlets). Flow-field data, which consisted of local Mach number, angle of attack, angle of sideslip, and total pressure ratio, were obtained at several candidate inlet axial positions for each basic configuration, after which parametric studies were carried out at the most promising inlet station for each configuration. Final results indicated that for the desired mission requirements, shielded-inlet locations offered better flow-field characteristics than fuselage side-mounted inlet locations. References 22, 34, 280 and 322 contain similar systematic studies of the type described in reference 10.

The remainder of the experimental data base is made up of less comprehensive studies. Thus, there exist experimental data only for a few of the many interesting combinations of the governing parameters. Indeed, considering the large number of geometric and free-stream parameters that affect the nature of the flow in the vicinity of a forebody and forebody-wing configuration, it is obvious that an experimental data base could never be complete for design in itself.
However, the experimental data base can be useful as a preliminary design tool as follows. As a part of a preliminary design process for determining candidate inlet positions for a particular configuration, it is necessary to determine regions of reasonably uniform local flow about the configuration. Experimental flow-field data showing the effect of various configuration parameters, such as fuselage shape, canopies, etc., on local flow-field gradients in various regions of interest are found in the experimental data base. Since parametric studies are carried out in a number of experimental programs for several types of wing-body configurations, general trends can be discerned, such as, the presence of high local-flow gradients in the vicinity of sharp corners. The experimental data base can be used in such a fashion, that is, to determine general trends in flow-field quantities and gradients due to various configuration parameters. The experimental data base is, however, of limited usefulness when it is applied to configurations for which the geometry is markedly different from that of any of the configurations represented in the existing data base.

7.2.3 Prediction methods

A wide variety of methods for predicting flow fields in the vicinity of a forebody or forebody-wing configuration are available. An attempt at evaluating the data base with respect to the methods listed in Table XVI is presented in this section through the use of available data comparisons. For the most part, data comparisons of forebody or forebody-wing flow fields are not given. Rather, comparisons are usually of surface pressure distributions. This gives rise to some difficulty in evaluating some prediction methods, since although a method may predict surface pressures accurately, there is no assurance that adequate prediction of flow fields in the candidate locations of an inlet system follows. For the most part, however, some conclusions can be drawn about methods for which surface pressure data comparisons alone are given.
Empirical methods for predicting forebody and forebody-wing flow fields are given in references 4, 11 and 398. References 4 and 11 both use a combination of theoretical methods and experimental data to obtain angle-of-attack effects on local flow-field quantities at under-wing and under-fuselage positions. As would be expected, these methods will be most successful when applied to configurations which are similar to those from which the data were obtained. Flow-field data comparisons using these two methods, however, could not be found. The method given in reference 398, which utilizes two empirical correlating parameters in conjunction with techniques of potential flow, Prandtl-Meyer expansions and conical shocks, predicts local Mach number, total flow angle, and roll angle (see fig. 88). Due to the assumptions and correlations made in developing this method, it is suggested by the author that its application be limited to axial locations at least one diameter downstream of the nose. Flow-field predictions were made for a 3.0/1 von Kármán ogive at a position 6 diameters downstream of the nose and 1.8 radii from the body centerline and are compared with data from references 399 and 400 in figure 89. Data comparisons for local Mach number, shown in figure 89(a), range from very good \( (M_{\infty} = 4.0, \alpha = 10^\circ) \) to poor \( (M_{\infty} = 2.0, \alpha = 20^\circ) \). Comparisons with local total-angle data at \( M_{\infty} = 3.0 \) show good agreement over the angle-of-attack range, as do the predicted local roll angle, which is assumed to be independent of \( M_{\infty} \) and \( \alpha \).

The prediction methods listed in the second group in Table XVI are those which solve the incompressible potential flow equations. All of these methods model the body as a distribution of surface singularity panels (i.e., sources, sinks, or vortex quadrilaterals). The panels are represented by plane quadrilaterals (refs. 300, 301, 315, 346), curved quadrilaterals (ref. 302), plane triangles (ref. 315), or conic frustrums (ref. 340), and the singularity strengths either are constant (refs. 300, 301, 315, 346) or vary in some fashion ( refs. 302, 340). The best-known of these methods is that of Hess and Smith (refs. 300, 301, 302) which has been developed into a computer program commonly referred to as the Douglas-Neumann Program. Comparisons of predicted and
experimental surface pressure distributions on wings of wing-body combinations show excellent agreement using this method. Predicted results of surface pressure distribution on an ogive cylinder were recently obtained by Nielsen Engineering & Research, Inc. using the NSRDC version of the Douglas-Neumann code, and were in good agreement with data over most of the ogive-cylinder. With respect to computer time, reference 301 estimates that a configuration consisting of 950 panels takes about 30 CPU minutes to run on an IBM 360/165, while for the NSRDC program described in reference 397, it is estimated that a 500-panel configuration will take about 5 CPU minutes on the CDC 6600. Flow-field data comparisons could not be located for this method or for any of the other methods in this group. The usefulness of these methods is obviously limited, however, since they are applicable only to incompressible flows.

Group 3 of Table XVI contains those methods which solve the linearized, steady, compressible potential flow equations. Of these, only the method of reference 303 requires that the fuselage shape be axisymmetric. With the exception of reference 318, all of these methods employ singularities to model a wing-body configuration. Flow-field data comparisons were available using the methods of references 303, 304, and 347, and therefore these methods are emphasized in this section.

References 303 and 304 utilize 3-D point sources and sinks along the body centerline and 2-D doublets in the crossflow plane to model circular cross sections. Noncircular cross sections are modeled in reference 304 through the use of polar harmonics in the crossflow plane. A distribution of constant-u-velocity-type panels to model wing loading and a distribution of constant-source-type panels to model wing thickness were used in reference 303, while reference 304 uses three-dimensional source panels to model wing thickness and a vortex-lattice model with imaging to account for wing-body interference.
Wing-body configurations for which flow-field data comparisons were obtained in reference 303 are shown in figure 90. Data comparisons of local upwash and sidewash velocities under the wing of WB1 are presented in figures 91 through 94. Comparisons for WB2 are shown in figures 95 and 96. The trends and magnitudes of the data are, for the most part, predicted well, although the predicted upwash and sidewash in the region of the shock compare poorly with data in some cases.

Predictions of flow fields in the vicinity of a body-alone and a wing-body configuration (see fig. 90(a)) were obtained for $M_{\infty} = 0.40$ using the method of reference 304. A noncircular addition was attached to the fuselage of the body-alone and wing-body configuration. Crossflow velocity vector plots were obtained for each of these models (see fig. 97). Predicted results agree well with data except in the region of the sharp corner. Data comparisons of local upwash velocity for the body-alone and wing-body configuration with and without noncircular additions are presented in figures 98 and 99. Predicted results agree well with data for the body-alone configuration at both $0^\circ$ and $6^\circ$ angle of attack. For the wing-body configuration, the trends of the changes due to the noncircular addition, although not all of the magnitudes of the changes, are predicted using this method.

With respect to computer time, a flow-field traverse of the kind presented in figures 91 through 94 for a simple wing-body configuration can be obtained in approximately 30 CPU seconds on the CDC 6600, using the method of reference 303. Using the method of reference 304, it takes about the same amount of time to calculate cross-flow velocities for twenty cross sections along a wing-body configuration of the kind shown in figure 97(b).

Using the method of reference 347, predicted local upwash and sidewash angles in the vicinity of a fighter-type fuselage and fuselage-wing configuration were compared with data from the Tailor-Mate program (ref. 10). Comparisons are presented in figures 100 and 101 for $M_{\infty} = .9$.
and show good agreement between theory and data at the given survey locations. A drawback in using this method is in setting up the geometry for a complicated configuration. This task involves a great deal of time, since the corner points of each surface panel must be input to the program.

Data comparisons using the other methods of group 3 include surface pressure distributions on body-alone and wing-body configurations. Predicted results are usually in good agreement with data for configurations in which viscous effects are small. A disadvantage of several of these methods is the long computer run times needed to obtain pressure distributions on a complete configuration. For example, the finite-element method of reference 318 requires approximately 775 seconds and 1715 seconds on the IBM 360/50 for typical subsonic (200 elements) and supersonic (556 elements) wing-body configurations, respectively. A typical wing-body configuration consisting of 180 wing panels and 72 fuselage panels requires 15 to 20 minutes on the CDC 6600 using the method of reference 309. Also, the method of reference 350, which is presently being incorporated with the method of reference 348 into a program called PAN AIR (Panel Aerodynamics), has recently been exercised on a light-weight supersonic configuration (ref. 403). Calculations for this wing-body model required 254 CPU seconds on a CDC 7600 using a sparse panel layout (380 panels) and 914 seconds using a dense panel layout (810 panels).

Group 4 in Table XVI presents the only method found which solves the unsteady, compressible, linearized potential flow equations. This finite-element method, which is presented in reference 319, is an extension of a previously derived method for steady flow (ref. 318). The method is applicable to complex wing-body configurations, but data comparisons using this method could not be found.

References 215 and 290, presented in group 5 of Table XVI, contain methods which predict three-dimensional and axisymmetric flow fields,
respectively, about axisymmetric bodies. Both methods employ finite-difference relaxation methods which include mixed-differencing in obtaining solutions to the transonic full potential flow equation (compressible, inviscid, steady). A mixed-difference scheme uses central differencing for locally subsonic flow and upwind differencing for locally supersonic flow. While both of these methods were developed for use in the transonic flow-field range, reference 215 can only be used for subsonic free-stream flows.

Once again, data comparisons are only in terms of surface pressure distributions. Both methods are applicable to blunt and pointed bodies, and comparisons with data indicate that predictions compare well with data except in regions where separation occurs. With respect to computer time, reference 215 indicates that a standard run (5 fine mesh $\phi =$ constant planes, 70 $z =$ constant planes, and 40 $r =$ constant surfaces, where $\phi$ is the circumferential angle) requires 12 to 15 minutes on the CDC 600. Reference 290 states that a typical subsonic free-stream case runs at about 3300 grid points per second on the CDC 6600 (RUN compiler), while a supersonic free-stream case runs at about 2500 points per second, with the number of iteration cycles required for convergence varying from 50 to 100. As an example for the reference 290 method, a staged missile configuration with a fine grid (9457 mesh points) and a coarse grid (2425 mesh points) required about 6 minutes and 2 minutes, respectively, to obtain a converged solution.

Group 6 presents methods for solving inviscid steady supersonic flow. Most of these methods were developed as a means of solving the blunt-body problem, i.e., a flow condition in which the bow shock is detached. However, these methods are also capable of solving attached bow-shock problems. With the exception of reference 324, all of these methods predict three-dimensional flow fields about general configurations at angle of attack.
Of special interest in this group are the shock-fitting (refs. 152, 289, 305) and shock-capturing (refs. 151, 153, 293, 306) methods. The shock-fitting methods model shocks (bow and embedded) as sharp discontinuities across which the Rankine-Hugoniot equations are applied, while the shock-capturing methods are inherently capable of predicting the location and strength of all shocks. Shock-capturing methods must employ a fine mesh in the vicinity of shocks, due to numerical instabilities near a shock wave. Shock-fitting methods can employ a coarser mesh in the region of a shock, although they are more difficult to program. As can be seen, various tradeoffs exist between these two types of methods.

Flow-field data comparisons were presented for many of these methods, although several of the comparisons were made for Mach numbers above the range of interest for supersonic aircraft. Figure 102 presents data comparisons of local Mach number, upwash angle, and sidewash angle in the vicinity of a fuselage configuration using the method of reference 34. This is a reference-plane-type method in which flow properties are calculated for several planes around the periphery of the body, with each plane emanating from the centerline. Two-dimensional, conical or axisymmetric methods are used to obtain flow solutions for each plane, and transverse effects between adjacent planes are accounted for using a wave-interference calculation. Prandtl-Meyer expansions are used to model the effects of a canopy or a fuselage corner. Comparisons of predicted results and data range from good for the local Mach number to poor for the local angle of sidewash.

Data comparisons of shock shape about a blunt-nose space shuttle vehicle using the methods of references 151 and 297 are shown in figure 103. Reference 151 is a shock-capturing method, while reference 297 is a semi-characteristic method. Although the Mach number at which comparisons are made is rather high, some comments can be made with respect to the accuracy of these methods. As can be seen in figure 103, the shock-capturing method shock shapes agree very well with data for
both the bow and canopy shocks. The predicted location of the canopy shock is displaced from the experimentally determined location due to slight differences between the actual geometry and the analytical approximation used in the calculations. The semi-characteristic method's predicted bow-shock shape agrees well with data only up to $X/L \approx 0.3$, and fails to predict the canopy shock for reasons that are discussed in detail in reference 404. Computation time comparisons using an IBM 360/67 computer were made in reference 404 for a pointed nose configuration at $M_\infty = 5$ and $\alpha = 5^\circ$. The comparisons show that while the shock capturing method (SCT) code is about four times faster than the method of characteristics (MOC) code on a point-for-point basis, the SCT code required approximately 2-1/2 times the total time used by the MOC code to calculate the flow field to a distance one half the body length. This is due to the fact that the SCT code used nearly three times as many radial points and twice as many radial planes as the MOC code.

Data comparisons of local Mach number and local-to-free-stream static pressure ratio in the vicinity of a fuselage configuration using the shock-fitting method of reference 152 are shown in figure 104. Predicted values agree well with data from reference 280 at the axial position of interest. As an example of computation time requirements, surface-pressure distributions were obtained for a complete fighter-type aircraft flying at $\alpha = 6^\circ$ and $M_\infty = 2.5$. This calculation was obtained in 63 CPU minutes on an IBM 370/165 using a 24 x 29 mesh in each cross-sectional plane with a total of 1200 marching steps.

Another method for which flow-field data comparisons exist is that of reference 337. Predicted values of local flow-field quantities are compared with Tailor-Mate data (ref. 10) in figure 105. Comparisons of local Mach number, upwash angle and sidewash angle show good agreement between theory and data, while the experimental results show a wider
spread of total pressures than the predicted results. The flow field about the entire Tailor-Mate fuselage was computed in about 5 minutes on a CDC 6600. Flow-field calculations for a simpler fuselage model at an angle of attack of $3.8^\circ$ and $M_\infty = 2.0$ were reported to have required about 3 minutes.

A few other flow-field data comparisons were available using other methods of group 6, but these comparisons were made for extremely high Mach numbers. These comparisons show good agreement between theory and data for bow-shock static pressure ratio (sphere-cone, ref. 306) and for bow-shock shape (blunt cone, ref. 289). Methods for which only surface-pressure-distribution data comparisons were available, but for which computation time examples were available, are presented in references 153 and 305. A flow-field calculation up to the inlet face of the B-1 aircraft flying at $M_\infty = 2.2$ and $\alpha = 3^\circ$ required 55 minutes on a CDC 7600 using the method of reference 153. Using the shock-fitting method of reference 305, calculation of axial surface pressure distributions at several circumferential positions around a blunt-nose space shuttle required 20 minutes on a CDC 6600 computer. This calculation was carried out using 25 mesh points between the body and shock, 20 points around the body, and 1000 points along the body.

Group 7 in Table XVI contains methods for solving the unsteady Euler equations. Of the four methods shown in this group, only reference 312 does not handle angle-of-attack cases.

Data comparisons utilizing the methods of reference 291 and 373 could not be found, and therefore no conclusions can be drawn about the accuracy of these methods. However, reference 373 does state that a typical blunt-body case, for which calculations can be carried out approximately two nose radii downstream, requires approximately 50 minutes on a CDC 3600.
References 311 and 312 are both shock-capturing methods which utilize an implicit approximate factorization finite-difference scheme to solve the flow equations. Although the method of reference 311 was derived to handle complicated aerodynamic shapes, data comparisons to date have been only for simple axisymmetric shapes. Predicted surface pressure distributions from reference 311 on a parabolic-arc body and a hemisphere-cylinder at transonic Mach numbers and several angles of attack are good agreement with data. Surface pressure distributions on mildly and severely indented bodies at zero angle of attack and high supersonic Mach numbers indicate good agreement between experiment and predicted values from reference 312. With respect to computation time, reference 311 indicates that flow-field calculations about a simple configuration typically require about one hour on a CDC 7600. A comparison of computer times presented in reference 312 shows it to be less than half as fast on a per-point basis as the method of reference 373 for a sphere at $M_\infty = 4.9$. However, the step size of reference 312 was nearly five times that of reference 373, thereby resulting in a faster convergence to the steady state.

Methods for solving the incompressible and compressible unsteady Navier-Stokes equation are presented in groups 8 and 9, respectively. Reference 316, which proposes two methods of solution for the incompressible case, does not contain data comparisons and therefore will not be discussed further here.

All of the references presented in group 9 can be used to solve for laminar or turbulent flows. The methods presented in references 311 and 312, which have been previously discussed in group 7, make use of the "thin layer" approximation, whereby all viscous terms containing derivatives whose direction is along the body are dropped. This approximation is valid only for high Reynolds number flows. Data comparisons are for surface pressure distributions over a hemisphere cylinder at transonic Mach numbers (ref. 311) and for mildly and severely indented bodies at high supersonic Mach numbers (ref. 312). Predicted values are
in good agreement with data in all cases. With respect to computation
time, reference 311 reports that for free-stream Mach numbers near unity,
a viscous solution for a simple body shape is typically obtained in 2-3
hours on a CDC 7600.

Data comparisons using the method of reference 314 could not be
found. However, Kutler (ref. 312) compares his predicted shock shape and
surface pressure distribution on a sphere-cylinder with that obtained
using the method of reference 314, and the results are in excellent
agreement for the case presented. One might make the assumption that
since Kutler's results agree well with data for a more complicated
configuration (indented body), then most likely his and Viviand's method
(ref. 314) agree well with data for a simple sphere-cylinder model.

Data comparisons using the method of reference 383 are available in
terms of shock shapes and pitot-pressure profiles. Figure 106 shows good
agreement between predicted and experimental results for the outermost
shock shape and fair agreement for the inner shock shape for a sharp 10°
half-angle cone at \( M_{\infty} = 7.95 \) and \( \alpha = 24° \). Figure 107 shows predicted and
experimental pitot-pressure profiles for a 5° sharp cone at \( M_{\infty} = 1.8 \) and
\( \alpha = 0° \). The results were obtained for a turbulent boundary layer at two
circumferential positions. Reference 383 attributes the poor agreement
at \( \phi = 152° \) to flow reattachment and the resulting complicated flow.

Methods which can be used to solve steady, compressible, viscous,
supersonic flow are listed in group 10 of Table XVI. All of these
methods solve some form of the Navier-Stokes equations.

References 292 and 374 both make use of Hayes' Equivalence Principle,
which relates the steady-state flow field over a slender body to an
equivalent time-dependent flow field in one less space dimension. In
reference 292, steady, three-dimensional flow is reduced to two-dimen-
sional, time-dependent flow, while reference 374 makes use of the
equivalence principle to reduce steady, axisymmetric flow to time-
dependent one-dimensional flow. In both references, the axial flow is assumed inviscid. However, axial effects are neglected in reference 292, whereas they are included in reference 374.

Data comparisons from reference 292 of local flow-field quantities for a fuselage configuration at \( M_\infty = 2.5 \) and \( \alpha = 15^\circ \) are shown in figure 108. Local sidewash angle contours indicate that the method gives fairly good results in the lower quadrant, while the predicted location of the canopy shock and the predicted sidewash in the upper quadrant do not agree with the data. The incorrect position of the canopy shock is believed to be due to the equivalence principle approximation of neglecting the velocity perturbation along the free-stream flow direction. The poor agreement between experimental and predicted local Mach number and local upwash angle contours are also attributed to this assumption. Other errors associated with this method are those due to discretization, which has to do with substituting a discrete set of points for the space-time continuum, and neglecting axial viscous effects.

Examples of computation time requirements were also given in this reference. Flow-field calculations for an ogive cylinder at a Mach number of 1.98 and an angle of attack of 10° required 3406 time steps and 10 hours on the UNIVAC 1108. Calculations for the previously mentioned fuselage configuration at \( M_\infty = 2.5 \) and \( \alpha = 15^\circ \) required 2079 time steps and 6 hours on the same machine.

The method of reference 374, which solves for the axisymmetric flow field by iteration, presents data comparisons of shock shapes for a cone-cylinder flare configuration at \( M_\infty = 4.54 \). Results are presented in figure 109 for the zeroth iterate (inviscid solution) and the first viscous iterate. Each successive iterate after the zeroth requires numerical data from the previous iterate and includes viscous effects. Based on a finite-difference mesh of 50 points, the zeroth iterate results were obtained using 2500 timesteps, while the first iterate required 7000 time steps. These cases required 5 minutes and 40 minutes.
As can be seen in figure 109, the first iterate solution cone and flare shock shapes agree well with data, while the method does not predict a separation shock for this iterate. More iterations would be required to determine separation in this region.

Reference 393 uses the parabolic Navier-Stokes marching finite-difference method, in which the streamwise viscous derivatives are neglected. This assumption rules out streamwise separation but allows crossflow separation. Laminar or turbulent boundary layer codes can be incorporated into this method. Predicted surface pressure distributions on an ogive-cylinder at $M_\infty = 3$ and $\alpha = 42^\circ$ are in good agreement with data. Flow-field data comparisons of shock shapes for an ogive-cylinder with the method of reference 297 show good agreement between the two methods. The ogive-cylinder calculations were obtained using 20 circumferential planes and 46 points between the body and shock and required about 1 second per marching step on a CDC 7600.

The method of reference 395 also employs the parabolic Navier-Stokes equations, but uses the "thin-layer" approximation (see discussion in group 9 section) to model viscous effects. Data comparisons of surface pressure distributions on a hemisphere-cylinder at $M_\infty = 1.4$ and $\alpha = 5^\circ$ indicate good agreement between predicted and experimental results.

Groups 11, 12, and 13 contain methods for predicting flows in the transonic Mach number regime. These methods make use of transonic small-disturbance theory (group 11), the transonic equivalence rule (group 12) and a far-field matching method (group 13).

The methods of references 296, 298, and 299 were developed using the transonic small-disturbance equation and are applicable to slender body, thin wing configurations under the assumptions of small flow deflections and $M_\infty$ near one. These methods incorporate mixed-difference schemes (see group 5 discussion) and a numerical method based on the relaxation method of Murman and Cole. Reference 298 outlines the basic method and
includes results for which both fully conservative and nonconservative differencing was used. Reference 296 uses the inviscid method of reference 298 coupled with a boundary-layer method which accounts for viscous effects on the wing. Reference 299 presents a modified transonic small-disturbance equation which can be used to improve the calculation of shocks due to the wing. Of these references, only reference 298 contains pressure distributions on wing-body combinations. The predicted results, which were obtained for a rectangular-wing-body and a swept-wing-body configuration, varied from fair to good when compared to experimental data. With respect to computation time, only reference 296 contained any useful information. It reported that calculations for a supersonic-wing aircraft at $M = .90$ and $\alpha = 3.56^\circ$ required about 10 minutes on a CDC 7600.

Reference 341 presents a method which utilizes the well-known transonic equivalence rule, which is summarized schematically in figure 110. As this figure shows, the total solution is composed of the inner solution, the outer limit of the inner solution, and the outer solution. The inner solution accounts for the near-field lift and thickness effects of the aircraft configuration. The outer solution calculates the flow about an equivalent axisymmetric body to obtain far-field effects. Details of the solutions of the inner and outer fields are found in Table XVII.

Data comparisons of flow-field pressures in the vicinity of a "bumpy" axisymmetric body are shown in figure 111 and upwash and sidewash comparisons for a scaled F-16 wing-body configuration are presented in figures 112 and 113. Predicted values of pressure for the bumpy body are in very good agreement with data at all survey locations. Figures 112 and 113 indicate good agreement between predicted and experimental sidewash and upwash except in the region of the wing trailing edge. These discrepancies are associated with discontinuities in the axial area distribution which are not accounted for in the prediction method. Flow-
field calculations for wing-body configurations usually require under 30 seconds on a CDC 7600.

The last group in Table XVI presents a far-field matching method for solving transonic flows. A schematic of the method is presented in figure 114 which illustrates the decomposition of the flow field into near-field, mid-field and far-field regions. The method of reference 317 uses the transonic small-disturbance equation in the near-field and middle-field regions. However, the only restrictions made on the inner-field flow equations is that they must reduce to the subsonic Prandtl-Glauert equations in the far-field. Other details of the method can be found in Table XVII. No data comparisons were included in this reference, but some information concerning computation time requirements was included. Calculations of the flow field in the vicinity of a nonlifting rectangular wing of aspect ratio 6 at \( M_\infty = .82 \) required 232 CPU seconds. The type of machine used was not given.

Data comparisons of flow fields were also given in reference 10 using a combination of a method of characteristics (MOC) and linear theory. The MOC method, which is restricted to axisymmetric body cases, is supplemented by the linear theory method to calculate the effects of forebody camber and the canopy. Also, a perturbation method is used in conjunction with the \( \alpha = 0^\circ \) MOC solution to obtain angle-of-attack effects. The linear theory, which is used exclusively at subsonic speeds, is the method of reference 308, and has been previously discussed.

Flow field data comparisons for a cambered nose fuselage configuration at Mach numbers .90 and 2.5 are presented in figures 115 and 116, respectively. The fuselage cross-sectional shape in the region of interest is also included in these figures. For the subsonic case the total pressure ratio was assumed to be equal to 1.0 for prediction purposes. The subsonic prediction results show the general angle-of-attack trends of the data with respect to local upwash and sidewash
angles, although the proper levels and gradients are not always in good agreement. Figure 116(a) indicates fairly good agreement between predicted and experimental total-pressure ratio, while the predicted results show a region of much higher Mach numbers and much lower sidewash angles in the upper part of the grid than those seen in the data. Figures 116(b) and (c) also indicate that the general angle-of-attack and angle-of-sideslip trends seem to be predicted reasonably well.

Figures 117 and 118 present flow-field data comparisons for a wing-fuselage configuration at Mach numbers of .90 and 2.5, respectively. The fuselage, which has a strake attached to it in the region where data was obtained, was represented as a wing with varying leading-edge sweep and with dihedral and camber. Since the linear method did not allow calculation of yawed wings, a special means of obtaining sideslip effects was also devised. Figure 117 shows predicted upwash and sidewash angles to be in rather poor agreement with data, while the predicted local Mach number comparisons are somewhat better. Figure 118 also shows rather poor agreement between experimental and predicted flow-field distributions. The consistent trends in the predicted total-pressure ratios are not seen in the data, and the predicted bow shock is not suggested by the test results. The predicted results also show a region of higher upwash angle on the inboard portion of the grid than those seen in the data.

7.2.4 Concluding remarks

The experimental data and prediction method data base for forebody and forebody-wing effects on inlet "free-stream" conditions has been presented. An evaluation of this data base with respect to completeness, ease of application, and accuracy has led to the following conclusions:

1. Although a few systematic studies for fuselage or wing-fuselage configurations exist (e.g., references 10, 22, 34, 280), the experimental data portion of the data base consists mostly of studies of body-of-
revolution configurations or simple wing-body configurations. Because of this, this portion of the data base is adequate for preliminary prediction only for configurations which are similar to those previously tested. Since the experimental data base for "real" fuselage shapes is very small, the usefulness of this portion of the data base is quite limited.

2. Empirical flow-field methods, as one might expect, are usually only useful for configurations similar to those used in obtaining the method. Also, the empirical methods presented are for bodies of revolution and simple wing-body configurations.

3. As the type of flow and configuration becomes more complicated, the cost of obtaining accurate predictions of flow fields increases. In other words, as the flow equations become more complicated to solve, such as the incompressible, steady, potential flow equations versus the unsteady Euler equations, and as the geometry becomes more complicated, such as a body of revolution versus a fighter-type wing-fuselage configuration, the cost of the computerized prediction methods and the complexity of the associated geometry package will increase. In terms of preliminary design, therefore, a tradeoff exists between cheaper methods which calculate flow fields for simplified configurations and more expensive methods which can be used for complicated geometries. The accuracy gained by using methods capable of handling complicated geometries may indeed be offset by increased costs due to long computer run times.
8. REFERENCES


APPENDIX A

REFERENCE LIST FOR INLET/AIRFRAME INTERACTION EFFECTS


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I-54. Black, J. A.: A Wind Tunnel Test of a 0.07-Scale Inlet Drag Model of the B-1 Aircraft at Mach Numbers from 0.60 to 1.60. AEDC-TR-73-188, Nov. 1973.


1-93. Steffen, F. W.: Cruise Performance of an Isolated 1.15 Pressure Ratio Turbomfan Propulsion System Simulator at Mach Numbers from 0.6 to 0.85. NASA TM X-3064, June 1974.


1-96. Re, R. J.: An Investigation of Several NACA 1-Series Axisymmetric Inlets at Mach Numbers from 0.4 to 1.29. NASA TM X-2917, Mar. 1974.


I-238. Lauer, R. F., Jr.: Results of a 0.2-Scale B-1 Production Inlet Development and Verification Tests at Subsonic and Supersonic Mach Numbers. AEDC TR-75-57, May 1975.


I-246. Riddell, J. F.: Results of a 0.2-Scale B-1 Inlet Verification Model Test at Transonic and Supersonic Mach Numbers. AEDC TR-72-156, Oct. 1972.


I-254. Keeney, F. J.: Longitudinal Static Stability and Inlet Drag Characteristics of the McDonnell F-15 Tactical Fighter Aircraft at Mach Numbers of 1.60 and 2.20. AEDC TR-71-100, May 1971. (Confidential)


APPENDIX B

REFERENCE LIST FOR AFTERBODY/AIRFRAME INTERACTION EFFECTS


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A-55. Homan, M. L.: Nacelle Afterbody Drag Investigation of a B-1 0.06-Scale Model at Mach Numbers from 0.55 to 2.2 AEDC-TR-72-176, Nov. 1972.


Angewandte Gasdynamik of the DFVLR, Porz-Wahn. Reference I-D,
AGARD-AG-208, Improved Nozzle Testing Techniques in Transonic

A-241. Rozendal, D., Groothoff, C. C., and Derksen, W. B. G.: Results
of NLR Contribution to AGARD Ad Hoc Study. Reference I-E,
AGARD-AG-208, Improved Nozzle Testing Techniques in Transonic

Varying Reynolds Number and Boundary Layer Displacement
Thickness on the External Flow Over Nozzle Boattails. Reference I-F,
AGARD-AG-208, Improved Nozzle Testing Techniques in Transonic

National Aeronautics and Space Administration Langley Research
Center. Reference I-H, AGARD-AG-208, Improved Nozzle Testing

Geometry. Reference II-A, AGARD-AG-208, Improved Nozzle Testing

and Discharge Coefficients. Reference II-C, AGARD-AG-208,

A-246. Dissen, H., Edmunds, H., Riedel, H., and Zacharias, A.:
Influence of Jet Parameters: Boattail Pressure Distribution and
Pressure Drag. (Evaluation of preliminary results contributed
by the participants in the AGARD ad-hoc study.) Reference II-D,
AGARD-AG-208, Improved Nozzle Testing Techniques in Transonic

A-247. Laughrey, J. A. and Richey, G. K.: Data Variance Due to
Different Testing Techniques. Reference II-E, AGARD-AG-208,

Interaction with Consideration of Reynolds Number Effects.
Reference II-F, AGARD-AG-208, Improved Nozzle Testing Techniques

A-249. Hiley, P. E. and Wallace, H. W.: Investigation of Non-
Axisymmetric Nozzles Installed in Tactical Aircraft. Volume I -
Study Results. AFFDL TR-75-61, June 1975.


APPENDIX C
CROSS-REFERENCE BY AUTHOR OF APPENDICES A AND B

In the following list, reference numbers prefixed by "I" are for documents dealing with Inlet/Airframe Interaction Effects (Appendix A), those prefixed by "A" deal with Afterbody/Airframe Interaction Effects (Appendix B).

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*Of the reports listed as containing data for this effect, refs. 16, 17, 24, 54, 121-123, 255, 390 pertain to the B-1. Detailed breakdown of the effects may be possible but requires further analysis to eliminate the effects of the metric partial forebody and stub wing and the use of the "propulsion reference inlets" as the datum for the measurements presented.
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### TABLE IV.- DATA FOR INLET EXTERNAL DRAG

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<th>Angle of Sidewall Deg</th>
<th>Cowl Shape</th>
<th>Side Plate Shape</th>
<th>Canopy Angle</th>
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<td>(C_{D_{EXT}})</td>
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<td>49 - 115</td>
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<td>(N^2)</td>
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注：数据表明，对于不同的攻角和距离，侧板/端板面积的对应值在0.65范围内波动。
TABLE IV. - (Concluded)

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<th>Angle of Side Lip, Deg</th>
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<th>Side Plate Shape</th>
<th>Rump Angle</th>
<th>Capture Area</th>
<th>Varying A, L Bleed</th>
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NOTES TO TABLE IV

1. Sum of external drag, drag due to the boundary-layer control system, drag of the partial fuselage and stub wing, measured relative to a "propulsion reference inlet"

2. Partial fuselage and stub wing included

3. Sum of external drag and drag of partial fuselage and stub wing, measured relative to a "propulsion reference inlet"

4. I denotes isolated inlet test

5. N.R. denotes quantity not reported

6. Inlet tested in isolated configuration and with non-metric wing simulator

7. Inlet tested in isolated configuration and with non-metric forebody

8. Sum of external drag, drag due to the bypass and boundary-layer control systems, and drag of partial fuselage and wing stub

9. Sum of external and forebody drag

10. Tested with metric forebody

11. Full aircraft model with full aircraft on one force balance, one inlet on another

12. Quarter-round inlet of F-111 type

13. Reynolds number range given based on cowl height; cowl height not reported

14. Sum of external drag, drag due to the bypass and boundary-layer control systems, and the drag of the partial fuselage and wing stub, measured relative to a "propulsion reference inlet"

15. Drag of entire airplane model

16. Model of entire F-15

17. Sum of external drag, boundary-layer control drag, forebody and external-store drag

18. Tested with forebody, with and without various external stores

19. Additive drag plus pressure and friction drag of entire model

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NOTES TO TABLE IV (Concluded)

20. Flight test of rocket models having common afterbodies

21. Sum of external drag and bleed bypass system drag

22. External drag does not include drag on lower cowl

23. Full B-1 aircraft model with balance installed in one inlet. Metric portion included nozzles and part of wing. Inlet configurations included are: flow through inlet, faired-over inlet w/BLC and bypass, flow through inlet w/ramps installed.

24. Inlet cowl and ramp integrated pressure-area chord force coefficient

25. Inlets mounted at the rear of an uncambered slender gothic wing

26. Sum of external drag and drag on wing

27. Drag increment; drag of wing-and-inlet configuration (minus base drag and inlet internal drag) minus drag of wing-alone configuration

28. Sum of external drag and bleed drag

29. Sum of spillage and interference drag

30. Reynolds number range given based on inlet radius; inlet radius not reported

31. External drag on a nacelle configuration

32. Nacelle configuration

33. Type of configuration tested with inlet is not known.

34. Sum of additive and bleed drag

35. Inlet design includes auxiliary airflow system

36. Sum of external drag, drag due to boundary-layer control system, and drag of the partial fuselage and stub wing

37. Numbers in parenthesis indicate percentage of external/internal contraction
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### TABLE VII.- MAJOR FEATURES OF THE PARTIAL PREDICTION METHODS

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<tr>
<td>4,5,7 (Ball)</td>
<td></td>
</tr>
</tbody>
</table>
- $C_{DA}$: one-dimensional momentum method. At critical mass flow, external compression surface contribution taken from data correlations. For subcritical mass flows in a supersonic free stream, the method of ref. 33 is used for 2-D inlets, and another method (referenced in 4) is used for axisymmetric inlets.
- $K_{add}$: $K_{add}$ factors for a number of inlets are shown. |
| 10 (Cawthon) | 
- $C_{DA}$: pressure integration along inlet capture streamline. For subcritical mass flows, the methods of refs. 32, 72, and 364 are used. 
- $C_{LS}$: correlations are given for 2-D and axisymmetric inlets ($\text{MHF}_{\text{ref}} = 1$). No provision for effects of details of cowl geometry. |
| 11 (Crosthwait) | 
- $C_{DA}$: one-dimensional momentum analysis. Curves given for compression surface contribution. Uses approach of ref. 72 for normal shock location. 
- $C_{LS}$: curves given for calculation of $C_{LS}$ depending on detailed cowl geometry ($\text{MHF}_{\text{ref}} = 1$). |
| 21 (Kamman) | "Pitot-Inlet Analogy" |
- $C_{DA_{\text{crit}}}$: pressure integration along stagnation streamline. 
- $C_{DC_{\text{crit}}}$: method not specified. 
- $\frac{\partial C_{D_{\text{EXT}}}}{\partial (\text{MHFR})}$: uses methods of refs. 59,71,72,102. |
<table>
<thead>
<tr>
<th>Reference</th>
<th>Major Features</th>
</tr>
</thead>
<tbody>
<tr>
<td>29 (Mount)</td>
<td>( C_{DA} ): one-dimensional momentum method. ( K_{add} ): for cowls of varying bluntness and external contours.</td>
</tr>
<tr>
<td>32 (Osmon)</td>
<td>( C_{DA} ): extends method of ref. 72 for locating terminal normal shock empirically for sidespills; one-dimensional momentum method used for ( C_{DA} ). ( C_{LS} ): correlations of data from ref. 33 in terms of detailed cowl geometry (MFR_{ref} \neq 1).</td>
</tr>
<tr>
<td>33 (Petersen)</td>
<td>( C_{DA} ): one-dimensional momentum method. In supersonic flow at subcritical mass flow ratios, a model for supersonic sidespills is included, subsonic and supersonic spillage over the cowl is allowed. No normal shock movement in model. Correlation presented for ramp drag at critical mass flows. ( K_{add} ): curves given for two ramp inlets with varying ramp angles, cowl and sideplate geometry.</td>
</tr>
<tr>
<td>39 (Smith)</td>
<td>( C_{LS} ): curves given for calculation of ( C_{LS} ) depending on detailed cowl geometry (MFR_{ref} = 1).</td>
</tr>
<tr>
<td>48 (Mascitti)</td>
<td>( C_{DA_{crit}} ): exact solution for right circular cones at zero angle of attack.</td>
</tr>
<tr>
<td>49 (Sibulkin)</td>
<td>( C_{DA} ): one-dimensional momentum method. For subcritical mass flows, the method of ref. 72 is used. For inlets with centerbodies, at critical or supercritical conditions conical flow theory is used.</td>
</tr>
<tr>
<td>Reference</td>
<td>Major Features</td>
</tr>
<tr>
<td>-----------</td>
<td>----------------</td>
</tr>
</tbody>
</table>
| 59 (Fraenkel) | $C_{DC\text{crit}}$: linear theory, references given in ref. 59.  
$\frac{\partial C_{D\text{EXT}}}{\partial (\text{MFR})}$: analysis ignores details of cowl shape, result is linear variation of $C_{D\text{EXT}}$ with MFR. |
| 71 (Goldsmith) | "Pitot-Inlet Analogy"  
$C_{DA\text{crit}}$: pressure integration along stagnation streamline in conical flow solution.  
$C_{DC\text{crit}}$: linear theory.  
$\frac{\partial C_{D\text{EXT}}}{\partial (\text{MFR})}$: extends method of refs. 59 and 72. |
<p>| 72 (Moeckel) | &quot;Continuity&quot; method for location of terminal normal shock. |
| 73 (Love) | Extension of method of ref. 72 to hypersonic Mach numbers |
| 81 (Schulte) | Prediction of location of terminal normal shock in 2-D inlets allowing for sidespil. |
| 86 (Moeckel) | $C_{LS}$: one-dimensional analysis for uncambered cowls ($MFR_{ref} = 1$), no dependence on detailed cowl geometry. |
| 90 (Crosthwait) | $C_{DA}$: empirical relations. No dependence on details of geometry. |
| 92 (Mascitti) | $C_{DA\text{crit}}$: approximate solution for right circular cones at zero angle of attack. |</p>
<table>
<thead>
<tr>
<th>Reference</th>
<th>Major Features</th>
</tr>
</thead>
<tbody>
<tr>
<td>102 (Dutton)</td>
<td>&quot;Pitot-Inlet Analogy&quot;</td>
</tr>
<tr>
<td></td>
<td>$C_{D_{A_{crit}}}$: pressure integration along stagnation streamline for 2-D inlets in supersonic flow with wedge shocks attached; with detached wedge shocks, uses method of ref. 59.</td>
</tr>
<tr>
<td></td>
<td>$C_{D_{crit}}$: for wedge cowls, and for elliptical contours by Prandtl-Meyer expansion theory.</td>
</tr>
<tr>
<td></td>
<td>$\frac{\delta C_{DEXT}}{\delta (MFR)}$: extends method of ref. 71 to two-dimensional geometry.</td>
</tr>
<tr>
<td>106 (Sharp)</td>
<td>In supersonic flow, uses &quot;Pitot-Inlet Analogy&quot;</td>
</tr>
<tr>
<td></td>
<td>$C_{D_{A_{crit}}}$: pressure integration along stagnation streamline; supersonic side-spill and sideplate contraction allowed for.</td>
</tr>
<tr>
<td></td>
<td>$C_{D_{crit}}$: wave drag calculated using Prandtl-Meyer theory; contributions due to lip bluntness, sideplates accounted for.</td>
</tr>
<tr>
<td></td>
<td>$\frac{\delta C_{DEXT}}{\delta (MFR)}$: extends analysis of refs. 71 and 102 to multi-ramp 2-D inlets.</td>
</tr>
<tr>
<td></td>
<td>In transonic flow, 2-D inlets are treated: $C_{D_{A_{crit}}}$ is calculated from one-dimensional momentum analysis with experimental data correlation, used for ramp contribution, $\frac{\delta C_{DA}}{\delta (MFR)}$ is determined as a function of free-stream Mach number from a correlation of data.</td>
</tr>
<tr>
<td>364 (Savage)</td>
<td>Method of locating terminal normal shock from ref. 32 for 2-D inlet; modified version of ref. 72 for axisymmetric inlet.</td>
</tr>
<tr>
<td>Reference</td>
<td>Major Features</td>
</tr>
<tr>
<td>-----------</td>
<td>----------------</td>
</tr>
<tr>
<td>364 (cont.)</td>
<td>$C_{DA}$: pressure integration along stagnation streamline; for subcritical flow, subsonic contribution calculated by multiplying relevant spillage area by average of pressure behind normal shock and pressure at cowl lip.</td>
</tr>
<tr>
<td>Solution Method</td>
<td>Applicable Geometry</td>
</tr>
<tr>
<td>-----------------------------------------------------</td>
<td>---------------------</td>
</tr>
<tr>
<td>Numerical solution of the unsteady Euler Equations</td>
<td>Pitot 2-D Axi</td>
</tr>
<tr>
<td>Finite-difference solution of full potential equation</td>
<td></td>
</tr>
<tr>
<td>Finite-difference solution of incompressible potential equation with compressibility correction</td>
<td></td>
</tr>
<tr>
<td>Streamtube curvature</td>
<td></td>
</tr>
<tr>
<td>Reference</td>
<td>Major Features</td>
</tr>
<tr>
<td>-----------</td>
<td>----------------</td>
</tr>
<tr>
<td>19 (Hawkins)</td>
<td>Explicit finite-difference solution of unsteady Euler equations (Godunov method, ref. 8.3). Boundary condition at inlet duct subsonic-outflow boundary (critical or subcritical flow) fixes axial velocity at one-dimensional ideal value, extrapolates pressure and density. Initial conditions used include free stream, &quot;best guess&quot;, and terminal flow fields from previous calculations. Automatic mesh generation is used. Calculations require large computer storage, long run times.</td>
</tr>
<tr>
<td>154 (Bansod)</td>
<td>Explicit finite-difference solution of unsteady Euler equations for two-dimensional and axisymmetric blunt-lipped pitot inlets in a uniform supersonic free stream. Upstream boundary is the bow shock. Boundary condition at inlet duct subsonic-outflow boundary is specially treated to enable mass flow ratio and/or the axial velocity component to be specified. Initial conditions used include estimates from previous calculations and/or empirical methods. Stability and accuracy problems encountered for thin lip shapes were stated to be due to treatment of boundary conditions.</td>
</tr>
<tr>
<td>354 (Rizzi)</td>
<td>Finite-volume solution of unsteady Euler equations written in integral conservation-law form for a pitot inlet in a supersonic free stream. The body and bow shock fitted mesh adjusts in time to the motion of the captured bow shock. The inlet duct subsonic-outflow boundary condition is a specified static pressure.</td>
</tr>
<tr>
<td>37 (Rochow)</td>
<td>Subsonic, compressible, steady potential flow field is solved using iterative finite-difference technique. Supersonic &quot;bubbles&quot; allowed in solution but strong shock waves invalidate potential analysis.</td>
</tr>
<tr>
<td>Reference</td>
<td>Major Features</td>
</tr>
<tr>
<td>-----------</td>
<td>----------------</td>
</tr>
<tr>
<td>52 (McVey)</td>
<td>Another description of the method given in ref. 37.</td>
</tr>
<tr>
<td>149 (Caughey)</td>
<td>Finite-difference solution of steady potential equation. Type-dependent differencing used in rectangular computational domain obtained by &quot;nearly conformal&quot; mapping procedure. Accelerated convergence of iteration solution obtained using several schemes.</td>
</tr>
<tr>
<td>150 (Arlinger)</td>
<td>Similar to ref. 149 but uses different mapping scheme to achieve computational domain and accelerated convergence schemes not included.</td>
</tr>
<tr>
<td>215 (Reyhner)</td>
<td>Line relaxation finite-difference solution of full potential equation in cylindrical coordinate system. Body must be axisymmetric, flow field three-dimensional (with plane of symmetry). Free stream must be subsonic, local flow may be supersonic.</td>
</tr>
<tr>
<td>52 (McVey)</td>
<td>The incompressible &quot;Douglas Neumann Potential Flow Program&quot; (refs. 8.4-8.6) which uses surface source distributions is used and corrected for compressibility effects using the Prandtl-Glauert, Karman-Tsein, Laitone, or Krahn methods. The Laitone correction is felt to yield best agreement to data.</td>
</tr>
<tr>
<td>405 (Hess)</td>
<td>Panel method applicable to three-dimensional pitot inlets (with or without centerbodies) in a subsonic free stream. Geometry must have plane of symmetry. Exact inviscid incompressible solution obtained by linear superposition of solutions for axial flow, simple pitch and yaw, and static operation. Compressibility effects accounted for through Lieblein-Stockman correction (ref. 8.7).</td>
</tr>
</tbody>
</table>
TABLE IX.- (Concluded)

Streamtube curvature

<table>
<thead>
<tr>
<th>Reference</th>
<th>Major Features</th>
</tr>
</thead>
<tbody>
<tr>
<td>274 (Keith)</td>
<td>The inviscid streamtube curvature technique is coupled via displacement thickness effects with the compressible turbulent boundary layer integral method of Stratford and Beavers (ref. 8.8). Turbulent separation is included via the Stratford criterion (ref. 8.9). The inviscid procedure utilizes automatic grid refinement and a matrix relaxation technique for use in a uniform free stream.</td>
</tr>
</tbody>
</table>
TABLE X.- PARAMETERS USED IN NADC CORRECTION OF CALAC DRAG PREDICTION

<table>
<thead>
<tr>
<th>Parameter Number</th>
<th>Expression</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>( \left{ \frac{\text{INSTA}}{2} \right}^{5/3} \left{ \frac{1}{(\gamma + 1)M^2} \right}^{1/3} )</td>
<td>Similarity relation; pressure</td>
</tr>
<tr>
<td></td>
<td>( \left{ \frac{\text{LBTA}}{\text{DNAX}} \sqrt{</td>
<td>M^2 - 1</td>
</tr>
<tr>
<td></td>
<td>( \left{ \frac{M^2 - 1}{(\gamma + 1) \left[ \frac{\text{INSTA}}{2} \right] M^2} \right}^{2/3} )</td>
<td></td>
</tr>
<tr>
<td>2</td>
<td>( \frac{M^2}{R_e} )</td>
<td>Pressure spreading estimator</td>
</tr>
<tr>
<td>3</td>
<td>( \frac{\text{ALXN}}{\text{ATHR}} )</td>
<td>Nozzle expansion ratio</td>
</tr>
<tr>
<td>4</td>
<td>( \frac{\text{ATHR}}{\text{AMB}} )</td>
<td>Ratio of internal flow area (nozzle throat) to fuselage area (area at metric break station)</td>
</tr>
<tr>
<td>5</td>
<td>( \frac{\text{INSTA}}{2} \left{ \frac{1}{(\gamma + 1)M^2} \right}^{1/3} )</td>
<td>Similarity relation; pressure drag estimator</td>
</tr>
<tr>
<td></td>
<td>( \left{ \frac{\text{LBTA}}{\text{DNAX}} \sqrt{</td>
<td>M^2 - 1</td>
</tr>
<tr>
<td></td>
<td>( \left{ \frac{M^2 - 1}{(\gamma + 1) \left[ \frac{\text{INSTA}}{2} \right] M^2} \right}^{2/3} )</td>
<td></td>
</tr>
<tr>
<td>Parameter Number</td>
<td>Expression</td>
<td>Definition</td>
</tr>
<tr>
<td>------------------</td>
<td>------------</td>
<td>------------</td>
</tr>
<tr>
<td>6</td>
<td>$\text{IMSTA}$</td>
<td>Integral-mean-slope of aftbody</td>
</tr>
<tr>
<td>7</td>
<td>$\frac{M}{Re^{3/2}}$</td>
<td>Boundary layer parameter; estimator of skin friction</td>
</tr>
<tr>
<td>8</td>
<td>$\left(\frac{\text{IMSTA}}{2}\right)^{5/3} \frac{(M^2 - 1)}{M^2 \left(\frac{\text{IMSTA}}{2}\right) (Y + 1)^{2/3}} \left[ (Y + 1) M^2 \right]^{1/3}$</td>
<td>Similarity relation; pressure drag estimator</td>
</tr>
<tr>
<td>9</td>
<td>$\left(\frac{\text{IMSTA}}{2}\right)^{5/3} \frac{(M^2 - 1)}{M^2 \left(\frac{\text{IMSTA}}{2}\right) (Y + 1)^{2/3}} \left[ (Y + 1) M^2 \right]^{1/3}$</td>
<td>Similarity relation; pressure drag estimator</td>
</tr>
<tr>
<td>10</td>
<td>$\frac{1}{Re^{5/6} \left[ (Y + 1) M^2 \right]^{1/3}} \frac{(M^2 - 1) Re^{1/3}}{\left[ (Y + 1) M^2 \right]^{2/3}}$</td>
<td>Similarity relation; estimator of effect of boundary layer on external pressure field (displacement effect)</td>
</tr>
</tbody>
</table>

*Reproduced from best available copy.*
<table>
<thead>
<tr>
<th>Parameter Number</th>
<th>Expression</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>11</td>
<td>( \frac{1}{R_e^{3/2}} )</td>
<td>Boundary layer parameter; estimator of skin friction</td>
</tr>
<tr>
<td>12</td>
<td>( \left{ \frac{\text{INSTA}}{2} \right}^{5/3} ) [( \frac{y + 1}{M^2} )]^{1/3} * { \sqrt{\frac{M^2}{1}} } )</td>
<td>Similarity relation; pressure drag estimator</td>
</tr>
<tr>
<td>13</td>
<td>( \left{ \frac{\text{INSTA}}{2} \right}^{5/3} ) [( \frac{y + 1}{M^2} )]^{1/3} * { \frac{\text{LBTAS}}{\text{DIAx}} } \sqrt{\frac{M^2}{1}} ) ^2</td>
<td>Similarity relation; pressure drag estimator</td>
</tr>
<tr>
<td>14</td>
<td>( \left{ \frac{\text{INSTA}}{2} \right}^{5/3} ) [( \frac{y + 1}{M^2} )]^{1/3} * { \frac{\text{LBTAS}}{\text{DIAx}} } \sqrt{\frac{M^2}{1}} ) ^3 * { \frac{M^2}{1} } [( \frac{\text{INSTA}}{3} ) ] ^{2/3}</td>
<td>Similarity relation; pressure drag estimator</td>
</tr>
</tbody>
</table>
**TABLE X.- (Concluded)**

<table>
<thead>
<tr>
<th>Parameter Number</th>
<th>Expression</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>15</td>
<td>NPR</td>
<td>Nozzle pressure ratio</td>
</tr>
<tr>
<td>16</td>
<td>CD&lt;sub&gt;CALAC&lt;/sub&gt;</td>
<td>Aftend drag; predicted value from CALAC &quot;Twin Jet Aftend Drag and Nozzle Internal Performance Computer Deck&quot;</td>
</tr>
<tr>
<td>17</td>
<td>ACC/AMB</td>
<td>Ratio of aftend projected areas</td>
</tr>
<tr>
<td>18</td>
<td>( \left{ \frac{\left( \frac{\text{THSTA}}{2} \right)^{5/3}}{\left( (Y + 1) \ N^2 \right)^{1/3}} \right} ) *</td>
<td>Similarity relation; pressure drag estimator</td>
</tr>
<tr>
<td>19</td>
<td>( \frac{\text{AEXN}}{\text{ACC}} )</td>
<td>Ratio of nozzle projected areas</td>
</tr>
</tbody>
</table>

\* \( \text{THSTA} \), \( Y \), and \( N \) are defined later in the text.
TABLE XI.- THEORETICAL PREDICTION METHODS

<table>
<thead>
<tr>
<th>Effects Treated</th>
<th>Subsonic</th>
<th>Transonic</th>
<th>Supersonic</th>
</tr>
</thead>
<tbody>
<tr>
<td>Boundary Layer</td>
<td>✓</td>
<td>✓</td>
<td></td>
</tr>
<tr>
<td>Separation</td>
<td>✓</td>
<td>✓</td>
<td>✓</td>
</tr>
<tr>
<td>Plume Shape</td>
<td>✓</td>
<td>✓</td>
<td>✓</td>
</tr>
<tr>
<td>Entrainment</td>
<td>✓</td>
<td>✓</td>
<td>✓</td>
</tr>
<tr>
<td>Compressibility</td>
<td>✓</td>
<td>✓</td>
<td>✓</td>
</tr>
<tr>
<td>Base Flow</td>
<td>✓</td>
<td>✓</td>
<td></td>
</tr>
</tbody>
</table>

References:

<p>| References | 77 | 46 | 48 | 51 | 80 | 106 | 117 | 118 | 160 | 162 | 164 | 210 | 211 | 212 | 217 | 227 | 228 | 230 | 232 | 253 | 264 | 267 | 282 | 286 | 292 | 299 | 301 | 306 |
|------------|----|----|----|----|----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|
|            | ✓  | ✓  | ✓  | ✓  | ✓  | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   | ✓   |</p>
<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Test Model</th>
<th>Data Presented</th>
<th>Mach Number</th>
<th>Reynolds Number Per Meter, x10^6</th>
<th>Angle of Attack</th>
<th>Nozzle Pressure Ratio</th>
<th>Nozzle Type</th>
<th>Interfairing</th>
<th>Nozzle Spacing</th>
<th>Empennage</th>
<th>Jet Simulation</th>
</tr>
</thead>
<tbody>
<tr>
<td>8</td>
<td>19</td>
<td>21</td>
<td>6-1.3</td>
<td></td>
<td></td>
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<td></td>
<td></td>
<td></td>
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<tr>
<td>9</td>
<td>16</td>
<td>1.2,3,4</td>
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<td>1-11</td>
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<tr>
<td>11</td>
<td>17</td>
<td>1.2,3,4</td>
<td>0.5-2.2</td>
<td>9.51-14.69</td>
<td>0*</td>
<td>1-22</td>
<td>C, CD</td>
<td>HM, O</td>
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<td>12</td>
<td>17</td>
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<td>0.5-1.3,2</td>
<td>9.61-12.76</td>
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<td>P</td>
<td>HM, O</td>
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<td>1.2,3,11</td>
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<td>9.12-12.33</td>
<td>-2* to 0.5*</td>
<td>1-9</td>
<td>P</td>
<td>HM, O</td>
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<td>18</td>
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<td>9.71-12.60</td>
<td>0*</td>
<td>1-6</td>
<td>CD</td>
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<tr>
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<td></td>
<td>0*</td>
<td>1-6</td>
<td>CD</td>
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<td>55-95</td>
<td>10.17-12.14</td>
<td>-2* to 6*</td>
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</tr>
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<td>21</td>
<td>16</td>
<td>1.2,11,13</td>
<td>6-1.2</td>
<td>10.07-14.04</td>
<td>0*</td>
<td>1-14</td>
<td>C, CD</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>22</td>
<td>19</td>
<td>4.1,2.1,15,3,2</td>
<td>3-1.5</td>
<td>3.78-19.68</td>
<td>-2* to 10*</td>
<td>1-6</td>
<td>C, CD</td>
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<tr>
<td>23</td>
<td>19</td>
<td>1.13</td>
<td>0-3-1.5</td>
<td>3.28-19.68</td>
<td>0*</td>
<td>1-14</td>
<td>C, CD</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>25</td>
<td>16</td>
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**TABLE XII. - DATA FOR AFTERBODY DRAG**
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**TABLE XII.** (Concluded)
### NOTES TO TABLE XII

1. Afterbody/Boattail/Nozzle pressure coefficient distribution
2. Afterbody/Boattail/Nozzle drag coefficient
3. Thrust-minus-drag coefficient, indicates thrust-minus-total drag or thrust-minus-nozzle drag
4. Thrust coefficient
5. Forebody pressure coefficient distribution
6. Forebody drag coefficient
7. Total configuration pressure coefficient distribution
8. Total configuration drag coefficient
9. Base pressure coefficient distribution
10. Base drag coefficient
11. Pressure or drag coefficient increment due to parametric changes - parameters include nozzle pressure ratio, empennage, area distribution on the afterbody, etc.
12. Aerodynamic characteristics - lift or pitching-moment coefficients and increments of lift or pitching-moment coefficients due to parametric changes
13. Boundary-layer data - includes Mach number profiles and stagnation pressure profiles
14. Interference pressure coefficient distribution - based on the pressure coefficient distribution of a reference model
15. Nozzle exhaust flow-field data - includes local Mach number and flow angle
16. Isolated boattail or single engine configuration - no empennage
17. Isolated twin-engine configuration - no empennage
18. Single engine aircraft configuration
19. Twin engine fighter-type aircraft configuration - includes isolated twin-engine afterbody configuration with empennage
20. Other configurations - includes twin-engine aircraft configuration with single engine under each wing, isolated boattail configuration with attached delta wing, etc.
NOTES TO TABLE XII (Concluded)

21. Any quantities not recorded for confidential reports indicates those quantities are confidential information.

22. Sum of afterbody, nozzle, and tail drag coefficient.

23. Tail drag coefficient.

24. Interference coefficients (drag, lift, pitching-moment) based on drag, lift, or pitching-moment coefficients of a reference model.

25. Determines shapes of boattail bodies of revolution for $C_{D_n}^{\text{min}}$.


27. Axial-force coefficient.

28. Isolated boattail with empennage.

29. Reynolds number range given based on maximum diameter of body; maximum diameter of body not reported.

30. Based on cruise nozzle diameter.

31. Horizontal tails attached to single vertical tail.

FS: Refers to full-scale configuration.

N.R.: Quantity not reported.
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<td>S</td>
<td>Solid Plume Simulator</td>
</tr>
<tr>
<td>Empennage</td>
<td>H</td>
<td>Horizontal stabilizer</td>
</tr>
<tr>
<td></td>
<td>V</td>
<td>Vertical stabilizer</td>
</tr>
<tr>
<td>Empennage Subscripts</td>
<td>δ</td>
<td>Deflected</td>
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<tr>
<td></td>
<td>1</td>
<td>Single vertical tail</td>
</tr>
<tr>
<td></td>
<td>2</td>
<td>Twin vertical tail</td>
</tr>
<tr>
<td>Nozzle</td>
<td>C</td>
<td>Convergent</td>
</tr>
<tr>
<td></td>
<td>CD</td>
<td>Convergent-divergent</td>
</tr>
<tr>
<td></td>
<td>E</td>
<td>Ejector</td>
</tr>
<tr>
<td></td>
<td>P</td>
<td>Plug</td>
</tr>
<tr>
<td></td>
<td>2D</td>
<td>Two dimensional</td>
</tr>
<tr>
<td>Reference Number</td>
<td>Configuration</td>
<td>Probe Function</td>
</tr>
<tr>
<td>------------------</td>
<td>---------------</td>
<td>----------------</td>
</tr>
<tr>
<td>10</td>
<td>F, W</td>
<td>$F_{r,a}$, $F_{b}$</td>
</tr>
<tr>
<td>22</td>
<td>W</td>
<td>$F_{r,a}$, $W_{s}$</td>
</tr>
<tr>
<td>34</td>
<td>F, W</td>
<td>$F_{r,a}$, $W_{s}$</td>
</tr>
<tr>
<td>174</td>
<td>F</td>
<td>$F_{r,a}$, $W_{s}$</td>
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<tr>
<td>175</td>
<td>W</td>
<td>$F_{r,a}$, $W_{s}$</td>
</tr>
<tr>
<td>179</td>
<td>F</td>
<td>$F_{r,a}$, $W_{s}$</td>
</tr>
<tr>
<td>220</td>
<td>W</td>
<td>$F_{r,a}$, $W_{s}$</td>
</tr>
<tr>
<td>222</td>
<td>W</td>
<td>$F_{r,a}$, $W_{s}$</td>
</tr>
<tr>
<td>335</td>
<td>W</td>
<td>$F_{r,a}$, $W_{s}$</td>
</tr>
<tr>
<td>338</td>
<td>W</td>
<td>$F_{r,a}$, $W_{s}$</td>
</tr>
<tr>
<td>339</td>
<td>W</td>
<td>$F_{r,a}$, $W_{s}$</td>
</tr>
<tr>
<td>341</td>
<td>W</td>
<td>$F_{r,a}$, $W_{s}$</td>
</tr>
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<tr>
<td>344</td>
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<tr>
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<td>369</td>
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<tr>
<td>370</td>
<td>W</td>
<td>$F_{r,a}$, $W_{s}$</td>
</tr>
<tr>
<td>371</td>
<td>W</td>
<td>$F_{r,a}$, $W_{s}$</td>
</tr>
</tbody>
</table>

**TABLE XIV. - DATA FOR EFFECTS OF AIRFRAME ON INLET FLOW QUANTITIES**
<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Configuration</th>
<th>Probe Position</th>
<th>Data Presented</th>
<th>Mach Number</th>
<th>Reynolds Number (Re)</th>
<th>Angle of Attack (deg)</th>
<th>Angle of Sidewall (deg)</th>
<th>General Configuration Data</th>
<th>Nose Shape</th>
<th>Nose Length</th>
<th>Fuselage Shape</th>
<th>Airplane</th>
<th>Wing L.E.</th>
<th>Wing Tip</th>
<th>Nacelle</th>
<th>Protrusions</th>
</tr>
</thead>
<tbody>
<tr>
<td>394</td>
<td>B</td>
<td>$F_{D}P_{L}F_{T}$</td>
<td>$P_{L}$</td>
<td>2.01</td>
<td>12.1</td>
<td>-9 to 15</td>
<td>0</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
</tr>
<tr>
<td>399</td>
<td>B</td>
<td>$F_{D}P_{L}F_{T}$</td>
<td>$P_{L}$</td>
<td>2.0 - 4.5</td>
<td>19.4 - 51.2</td>
<td>0 to 24</td>
<td>0</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
</tr>
<tr>
<td>400</td>
<td>B</td>
<td>$F_{D}P_{L}F_{T}$</td>
<td>$P_{L}$</td>
<td>2.1 - 4.3</td>
<td>19.7 - 40.7</td>
<td>0 to 23</td>
<td>0</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
</tr>
<tr>
<td>401</td>
<td>B</td>
<td>$F_{D}P_{L}F_{T}$</td>
<td>$P_{L}$</td>
<td>2.5 - 4.3</td>
<td>39.4 - 59.4</td>
<td>-10 to 10</td>
<td>0</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
</tr>
<tr>
<td>402</td>
<td>B</td>
<td>$F_{D}P_{L}F_{T}$</td>
<td>$P_{L}$</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
</tr>
<tr>
<td>406</td>
<td>B</td>
<td>$F_{D}P_{L}F_{T}$</td>
<td>$P_{L}$</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>*</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
<td>/</td>
</tr>
</tbody>
</table>

*Range is confidential
<table>
<thead>
<tr>
<th>Column Heading</th>
<th>Symbol</th>
<th>Subscripts or Superscripts</th>
</tr>
</thead>
</table>
| Configuration    | B: Body of revolution  
                 | F: Fuselage with canopy  
                 | I: Installed inlet  
                 | W: Wing |
|                  | l: Fuselage with various cross-sectional shapes, no canopy |
| Probe Position   | C: Inlet cowl-lip plane  
                 | F: Within a body diameter of the fuselage or body of revolution  
                 | W: Under-wing position |
|                  | a: ahead of wing  
                 | aft: aft of wing  
                 | b: bottom of fuselage  
                 | s: side of fuselage  
                 | sh: shielded by wing  
                 | t: top of fuselage |
| Data Presented  | $M_L$: Local Mach number  
                 | $P_L$: Local to freestream total pressure ratio ($P_{T_L}/P_{T_\infty}$)  
                 | $P_L$: Local to freestream static pressure ratio ($P_L/P_{\infty}$)  
                 | $Q_L$: Local to freestream dynamic pressure ratio ($q_L/q_{\infty}$)  
                 | $V_L$: Local to freestream total velocity ratio ($V_{L}/V_{\infty}$)  
                 | $v_L$: Local perturbation velocities:  
                 | $W_L$: local sidwash velocity  
                 | $w_L$: local upwash velocity  
                 | $u_L$: local velocity component in free-stream direction  
                 | $\alpha_L$: Local upwash angle ($\tan^{-1} \frac{W_L}{V_{L}/V_{\infty}}$)  
                 | $\beta_L$: Local sidwash angle ($\tan^{-1} \frac{V_{L}/V_{\infty}}{u_L}$)  
                 | $\epsilon_L$: Local total flow angle (total included angle between the local velocity vector and the vehicle axis, see fig. 88)  
                 | $\phi_L$: Local roll angle (defines the plane in which $\epsilon_L$ is contained, measured relative to the tangent of the arc of radius of the point of interest, fig. 88)  
<p>|                  | $\ldots$ indicates average local quantity |</p>
<table>
<thead>
<tr>
<th>Group</th>
<th>Flow Equations or Type of Method</th>
<th>Type of Solution</th>
<th>Applicable Speed Range</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
<td>Subsonic</td>
</tr>
<tr>
<td>1</td>
<td>Empirical methods</td>
<td>3-D</td>
<td>4.11$^1$</td>
</tr>
<tr>
<td>2</td>
<td>Incompressible, steady potential flow</td>
<td>3-D</td>
<td>300,301,302,315, 340$^1$,346</td>
</tr>
<tr>
<td>3</td>
<td>Linearized, steady, compressible potential flow</td>
<td>3-D</td>
<td>304,309,318,347, 348</td>
</tr>
<tr>
<td>4</td>
<td>Unsteady, compressible linearized potential flow</td>
<td>3-D</td>
<td>319</td>
</tr>
<tr>
<td>5</td>
<td>Full compressible potential flow</td>
<td>3-D</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>Ax1</td>
<td></td>
</tr>
<tr>
<td>6</td>
<td>Steady Euler equations</td>
<td>3-D</td>
<td></td>
</tr>
<tr>
<td></td>
<td></td>
<td>Ax1</td>
<td></td>
</tr>
<tr>
<td>7</td>
<td>Unsteady Euler equations</td>
<td>3-D</td>
<td>291$^3$,311</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Ax1</td>
<td></td>
</tr>
<tr>
<td>8</td>
<td>Unsteady, incompressible Navier-Stokes equations</td>
<td>3-D</td>
<td>316</td>
</tr>
</tbody>
</table>

$^1$Axisymmetric fuselage only  
$^2$Slender bodies and thin wings  
$^3$Slender bodies
<table>
<thead>
<tr>
<th>Group</th>
<th>Flow Equations or Type of Method</th>
<th>Type of Solution</th>
<th>Applicable Speed Range</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>Subsonic</td>
<td>Transonic</td>
</tr>
<tr>
<td>9</td>
<td>Unsteady, compressible Navier-Stokes equations</td>
<td>3-D</td>
<td>311</td>
</tr>
<tr>
<td></td>
<td></td>
<td>3-D</td>
<td>312,314</td>
</tr>
<tr>
<td>10</td>
<td>Steady, compressible viscous flow</td>
<td>3-D</td>
<td>292,3,393,395</td>
</tr>
<tr>
<td>11</td>
<td>3-D transonic small disturbance theory</td>
<td>3-D</td>
<td>296,298,299$_2$</td>
</tr>
<tr>
<td>12</td>
<td>Transonic equivalence rule</td>
<td>3-D</td>
<td>317</td>
</tr>
<tr>
<td>13</td>
<td>Far-field matching method</td>
<td>3-D</td>
<td>317</td>
</tr>
</tbody>
</table>

1 axisymmetric fuselage only
2 slender bodies and thin wings
3 slender bodies
<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Features of Prediction Method</th>
</tr>
</thead>
</table>
| 4 (Ball)         | - under-wing configuration - empirical relations based on Prandtl-Meyer relations for a shock under a 2-D wing used to determine effects of angle of attack (supersonic free stream only)  
                   - under-fuselage configuration - effects of angle of attack based on experimental data (subsonic and supersonic free stream)  
                   - side-mounted configurations - angle-of-attack effects assumed negligible based on data |
| 11 (Crosthwait)  | Fuselage Effects:  
                   - superpose potential flow solution of Laitone (axial flow) and incompressible slender-body cross-flow theory solution for subsonic flows  
                   - combined supersonic/hypersonic similarity rule for supersonic flows ($M_\infty \geq 2$) |
| 398 (Mahoney)    | Wing Effects:  
                   - experimental results on a limited number of wings mounted on a circular fuselage are used to determine angle-of-attack effects for subsonic and supersonic flows  
                   - forebody represented by suitable tangent cone  
                   - two correlating parameters, used in conjunction with a conical shock and 2-D Prandtl-Meyer expansions, are employed to determine flow angles at body surface points  
                   - potential flow theory used to determine flow angles at off-body points |
<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Features of Theoretical Prediction Method</th>
</tr>
</thead>
<tbody>
<tr>
<td>300, 301 (Hess)</td>
<td>· singularity method</td>
</tr>
<tr>
<td></td>
<td>· non-lifting surfaces modeled using constant-strength source panels on surface (ref. 300)</td>
</tr>
<tr>
<td></td>
<td>· lifting surfaces modeled using constant-strength source panels and doublet panels on the surface and trailing vorticity (ref. 301)</td>
</tr>
<tr>
<td></td>
<td>· panels are plane quadrilaterals</td>
</tr>
<tr>
<td>302 (Hess)</td>
<td>· replaces plane surface panels having constant source strengths (ref. 300) with curved surface panels having variable singularity strengths</td>
</tr>
<tr>
<td>315 (Asfar)</td>
<td>· singularity method</td>
</tr>
<tr>
<td></td>
<td>· body is modeled using constant-strength source panels on the surface and surface distributions of vorticity (vortex lattice)</td>
</tr>
<tr>
<td></td>
<td>· panels are plane quadrilaterals and triangles</td>
</tr>
<tr>
<td>340 (Geissler)</td>
<td>· singularity method</td>
</tr>
<tr>
<td></td>
<td>· body modeled using surface distribution of sources, sinks, and vortices</td>
</tr>
<tr>
<td></td>
<td>· surface of a body of revolution divided into conic frustrums of small axial extension</td>
</tr>
<tr>
<td></td>
<td>· bodies of revolution only</td>
</tr>
<tr>
<td></td>
<td>· weighting factors, which are functions of body geometry, are used to combine the singularities</td>
</tr>
<tr>
<td>346 (Maskew)</td>
<td>· singularity method</td>
</tr>
<tr>
<td></td>
<td>· body modeled using surface distribution of quadrilateral vortex rings of constant strength</td>
</tr>
<tr>
<td>Reference Number</td>
<td>Features of Theoretical Prediction Method</td>
</tr>
<tr>
<td>------------------</td>
<td>------------------------------------------</td>
</tr>
<tr>
<td>303 (Dillenius)</td>
<td>• singularity method</td>
</tr>
<tr>
<td></td>
<td>• axisymmetric cross section modeled using 3-D point sources and sinks along body centerline and 2-D doublets in cross-flow plane</td>
</tr>
<tr>
<td></td>
<td>• wing modeled using constant u-velocity-type panels and constant source-type panels</td>
</tr>
<tr>
<td></td>
<td>• position of wing leading-edge shock determined using nonlinear correction to linear solution based on wing thickness</td>
</tr>
<tr>
<td>304 (Dillenius)</td>
<td>• singularity method</td>
</tr>
<tr>
<td></td>
<td>• axisymmetric cross section modeled same as that for reference 303</td>
</tr>
<tr>
<td></td>
<td>• noncircular cross section modeled using 3-D point sources and sinks along centerline of equivalent body and polar harmonics and 2-D sources in cross-flow plane</td>
</tr>
<tr>
<td></td>
<td>• wing-fuselage interference modeled using vortex-lattice model with imaging</td>
</tr>
<tr>
<td></td>
<td>• wing modeled using 3-D source panels on wing surface</td>
</tr>
<tr>
<td>309 (Woodward)</td>
<td>• singularity method</td>
</tr>
<tr>
<td></td>
<td>• constant source distribution on body panels</td>
</tr>
<tr>
<td></td>
<td>• linearly varying vortex distribution on wing panels</td>
</tr>
<tr>
<td>308 (Woodward)</td>
<td>• singularity method</td>
</tr>
<tr>
<td></td>
<td>• wing modeled using a surface distribution of vorticity and sources</td>
</tr>
<tr>
<td>Reference Number</td>
<td>Features of Theoretical Prediction Method</td>
</tr>
<tr>
<td>------------------</td>
<td>------------------------------------------</td>
</tr>
</tbody>
</table>
| 308 (cont.)      | • fuselage modeled using line sources and doublets distributed along body centerline  
                     • wing-body-interference modeled using a surface distribution of vorticity on the body in the region of influence of wing-body intersection |
| 318 (Morino)     | • finite-element method  
                     • surface of body divided into small hyperboloidal elements  
                     • potential and normal derivative of potential within each element are constant at the element centroid |
| 349,350 (Ehlers) | • singularity method  
                     • planar source and doublet panels  
                     • source distribution varies linearly, doublet distribution varies quadratically  
                     • in reference 349, there is a single quadratic distribution of doublet strength over each quadrilateral panel  
                     • in reference 350, each quadrilateral panel is divided into eight triangular subpanels in such a way that all panel edges are contiguous with adjacent panels; separate quadratic distributions over each triangular subpanel are prescribed |
| 347 (AEDC)       | • singularity method  
                     • configuration modeled using surface distribution of vorticity and sources |
| 348 (Johnson)    | • singularity method  
                     • curved source and doublet panels  
                     • singularity strengths vary as polynomials |
### TABLE XVII.- (Continued)

**Group 4.- Unsteady, Compressible, Linearized Potential Flow Equations**

<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Features of Theoretical Prediction Method</th>
</tr>
</thead>
<tbody>
<tr>
<td>319 (Morino)</td>
<td>• finite-element method</td>
</tr>
</tbody>
</table>

**Group 5.- Full Compressible Potential Flow Equations**

<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Features of Theoretical Prediction Method</th>
</tr>
</thead>
</table>
| 215 (Reyhner)    | • finite-difference relaxation method
|                  | • mixed-difference scheme
|                  | • subsonic free stream only (within transonic range)                                                     |
| 290 (South)      | • finite-difference relaxation method
|                  | • modified mixed-difference scheme: difference scheme simulates differencing along and normal to local velocity vector
<p>|                  | • subsonic and supersonic free stream (within transonic range)                                           |</p>
<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Features of Theoretical Prediction Method</th>
</tr>
</thead>
</table>
| 34 (Prokop)      | • reference plane-type method (flow properties calculated for several planes around periphery of body, each emanating from centerline)  
• two-dimensional shock-expansion method; conical method of Ferri, two-dimensional m.o.c. or axisymmetric m.o.c. used to obtain flow solutions for each plane  
• transverse flow effects between adjacent planes accounted for using a wave-interference calculation which modifies static pressure and streamline direction at each point of interest  
• effects of canopy and fuselage corner modeled using Prandtl-Meyer expansions |
| 151 (Kutler), 306 (Solomon) | • shock-capturing technique  
• MacCormack's second-order predictor-corrector method  
• conservative form of equations of motion  
• embedded shocks |
| 152 (Marconi), 305 (Moretti) | • shock-fitting technique  
• MacCormack's second-order predictor-corrector method  
• non-conservative form of equations of motion  
• embedded shocks |
| 153 (D'Attorre) | • shock-capturing technique  
• Lax-Wendroff finite-difference method  
• conservative form of equations of motion  
• embedded shocks |
**TABLE XVII.-(Continued)**

**Group 6.-(Continued)**

<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Features of Theoretical Prediction Method</th>
</tr>
</thead>
</table>
| 289 (Thomas)     | • shock-fitting technique
|                  | • MacCormack's second-order predictor-corrector method
|                  | • non-conservative form of equations of motion
|                  | • no embedded shocks |
| 293 (Walkden)    | • shock-capturing technique
|                  | • pseudo method of characteristics method
|                  | • non-conservative form of equations of motion
|                  | • embedded shocks |
| 297 (Rakich)     | • method of characteristics (reference plane, or semi-characteristic, method)
|                  | • MacCormack's second-order predictor-corrector used to obtain cross derivatives
|                  | • embedded shocks |
| 324 (Inouye)     | • inverse, integral method (shock shape assumed and equations of motion integrated numerically by finite-difference method to determine corresponding body shape)
|                  | • predictor-corrector scheme for subsonic and transonic regions
|                  | • method of characteristics used in supersonic region
|                  | • iterative process used for embedded shocks |
| 337 (Chu)        | • method of characteristics |
TABLE XVII.- (Continued)

Group 6.- (Concluded)

<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Features of Theoretical Prediction Method</th>
</tr>
</thead>
<tbody>
<tr>
<td>337 (cont.)</td>
<td>• bicharacteristic, inverse redundant method</td>
</tr>
<tr>
<td></td>
<td>(employ governing differential equations along generators of Mach cone through a point in flow)</td>
</tr>
<tr>
<td></td>
<td>• embedded shocks</td>
</tr>
<tr>
<td>375 (Babenko)</td>
<td>• semi-implicit finite-difference scheme</td>
</tr>
<tr>
<td></td>
<td>• non-conservative form of equations of motion</td>
</tr>
</tbody>
</table>
TABLE XVII.- (Continued)

Group 7.- Unsteady Euler Equations

<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Features of Theoretical Prediction Method</th>
</tr>
</thead>
<tbody>
<tr>
<td>291 (Li)</td>
<td>• Euler equations are linearized, then reduced to one equation through the use of a potential function&lt;br&gt;• boundary conditions satisfied at body, then solution worked outward toward infinity</td>
</tr>
<tr>
<td>311 (Pulliam), 312 (Kutler)</td>
<td>• shock-capturing technique&lt;br&gt;• conservative form of equations of motion&lt;br&gt;• embedded shocks&lt;br&gt;• implicit approximate factorization finite-difference technique</td>
</tr>
<tr>
<td>373 (Moretti)</td>
<td>• four-dimensional method of characteristics used to determine quantities at shock points and body points&lt;br&gt;• time-dependent explicit Lax-Wendroff type numerical scheme used to determine quantities within shock layer</td>
</tr>
</tbody>
</table>
### Group 8. Unsteady, Incompressible Navier-Stokes Equations

<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Features of Theoretical Prediction Method</th>
</tr>
</thead>
<tbody>
<tr>
<td>316 (Mastin)</td>
<td>• two partially implicit finite-difference formulations of Navier-Stokes equations developed</td>
</tr>
</tbody>
</table>

1. one-step first-order time differencing scheme
   - viscous and pressure terms treated implicitly
   - successive overrelaxation method used

2. two-step projection method
   - pressure equation uncoupled from momentum equation
   - successive overrelaxation method used

• body-fitted curvilinear coordinate system used
TABLE XVII.- (Continued)

Group 9.- Unsteady, Compressible Navier-Stokes Equations

<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Features of Theoretical Prediction Method</th>
</tr>
</thead>
</table>
| 311 (Pulliam), 312 (Kutler) | • shock-capturing technique  
• conservative form of equations of motion  
• embedded shocks  
• implicit approximate factorization finite-difference technique (noniterative, alternating direction implicit)  
• "thin-layer" approximation used to handle viscous effects  
• high Reynolds number flows only |
| 314 (Viviand) | • shock-fitting technique  
• MacCormack's second-order predictor-corrector method  
• non-conservative form of equations of motion  
• no embedded shocks |
| 383 (McRae) | • conical symmetry assumption applied to full Navier-Stokes equations  
• shock-capturing technique  
• MacCormack's second-order predictor-corrector method  
• weak conservation form of equations of motion  
• embedded shocks |

For refs. 314 and 383: laminar or turbulent boundary layer can be used.
TABLE XVII.- (Continued)

Group 10.- Steady, Compressible, Viscous Flow

<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Features of Theoretical Prediction Method</th>
</tr>
</thead>
<tbody>
<tr>
<td>292 (Walitt)</td>
<td>- Hayes' Equivalence Principle used to relate the time-dependent 2-D flow to a steady 3-D flow</td>
</tr>
<tr>
<td></td>
<td>- Navier-Stokes equation solved in cross-flow plane using finite-difference method</td>
</tr>
<tr>
<td></td>
<td>- simultaneous solution of laminar boundary layer and inviscid flow field</td>
</tr>
<tr>
<td></td>
<td>- inviscid axial flow, axial effects neglected</td>
</tr>
<tr>
<td></td>
<td>- laminar boundary layer (2-D)</td>
</tr>
<tr>
<td>374 (Walitt)</td>
<td>- steady-state axisymmetric flow field made analogous to time-dependent 1-D flow using Hayes' Equivalence Principle</td>
</tr>
<tr>
<td></td>
<td>- &quot;full&quot; Navier-Stokes equations for axisymmetric flow solved by iteration</td>
</tr>
<tr>
<td></td>
<td>- inviscid axial flow, axial effects included</td>
</tr>
<tr>
<td></td>
<td>- turbulent or laminar boundary layer (1-D, corrected for axial symmetry)</td>
</tr>
<tr>
<td>393 (Rakich)</td>
<td>- parabolized Navier-Stokes equations (streamwise viscous derivatives neglected; rules out streamwise separation but allows crossflow separation)</td>
</tr>
<tr>
<td></td>
<td>- laminar and turbulent boundary layers</td>
</tr>
<tr>
<td></td>
<td>- implicit factorization method of Beam &amp; Warming used</td>
</tr>
<tr>
<td></td>
<td>- conservative form of equations of motion</td>
</tr>
<tr>
<td>395 (Schiff)</td>
<td>- parabolized Navier-Stokes equation</td>
</tr>
<tr>
<td></td>
<td>- &quot;thin-layer&quot; approximation used to handle viscous effects</td>
</tr>
<tr>
<td></td>
<td>- high Reynolds number flows only</td>
</tr>
<tr>
<td></td>
<td>- conservative form of equations of motion</td>
</tr>
</tbody>
</table>
TABLE XVII.- (Continued)

Group 11.- 3-D Transonic Small-Disturbance Theory

<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Features of Theoretical Prediction Method</th>
</tr>
</thead>
<tbody>
<tr>
<td>298 (Bailey)</td>
<td>• equations derived from transonic small disturbance theory under assumptions of small flow deflections and free-stream Mach number near unity</td>
</tr>
<tr>
<td></td>
<td>• finite-difference method based on relaxation method of Murman &amp; Cole</td>
</tr>
<tr>
<td></td>
<td>• shock-capturing technique</td>
</tr>
<tr>
<td></td>
<td>• mixed-difference scheme</td>
</tr>
<tr>
<td></td>
<td>• handles subsonic upstream/supersonic downstream and supersonic upstream/subsonic downstream conditions</td>
</tr>
<tr>
<td></td>
<td>• can use fully conservative or nonconservative differencing</td>
</tr>
<tr>
<td>296 (Mason)</td>
<td>• inviscid method is that of references 298, 299</td>
</tr>
<tr>
<td></td>
<td>• viscous effects on the wing also included</td>
</tr>
<tr>
<td>299 (Ballhaus)</td>
<td>• several modified 3-D transonic small-disturbance equations introduced to improve prediction of shocks</td>
</tr>
<tr>
<td></td>
<td>• other techniques mentioned for reference 298 also apply</td>
</tr>
<tr>
<td>Reference Number</td>
<td>Features of Theoretical Prediction Method</td>
</tr>
<tr>
<td>------------------</td>
<td>-------------------------------------------</td>
</tr>
<tr>
<td>341 (Stahara)</td>
<td>total solution made up of inner solution, far-field behavior of inner solution, and outer solution</td>
</tr>
<tr>
<td></td>
<td>inner solution:</td>
</tr>
<tr>
<td></td>
<td>- superposition of solutions to thickness and lift problems obtained from 2-D Laplace equation in cross flow plane at each x-station</td>
</tr>
<tr>
<td></td>
<td>- thickness solution obtained using method of distributed singularities developed by Stocker</td>
</tr>
<tr>
<td></td>
<td>- lift solution obtained using analytic conformal mapping solution determined by Spreiter for circular body with mid-mounted zero-thickness wings</td>
</tr>
<tr>
<td></td>
<td>outer solution:</td>
</tr>
<tr>
<td></td>
<td>- 3-D nonlinear transonic differential equation is solved</td>
</tr>
<tr>
<td></td>
<td>- wing-body combination is modeled using a line distribution along body axis of a combination of sources (related to total cross-sectional area, axial lift distribution, and spanwise wing loading) and doublets (related to axial lift distribution)</td>
</tr>
<tr>
<td></td>
<td>- in thickness-dominated domain, basic nonlinear outer flow determined principally by line source distribution. For this case, governing PDE is nonlinear axisymmetric transonic small-disturbance equation which is solved using a finite-difference successive line over-relaxation procedure which uses Murman-Cole type-dependent difference operators</td>
</tr>
<tr>
<td></td>
<td>- in lift-dominated domain, nonlinear outer flow determined by line source and line doublet distribution</td>
</tr>
</tbody>
</table>
### TABLE XVII.- (Concluded)

**Group 13.- Far-Field Matching Method**

<table>
<thead>
<tr>
<th>Reference Number</th>
<th>Features of Theoretical Prediction Method</th>
</tr>
</thead>
<tbody>
<tr>
<td>317 (Lee)</td>
<td>• computational field divided into near-field, mid-field, and far-field regions</td>
</tr>
<tr>
<td></td>
<td>• near-field equations must reduce to subsonic linearized potential flow equations (Prandtl-Glauert) in far-field region</td>
</tr>
<tr>
<td></td>
<td>• far-field solution is a linear combination of source distributions at the far-field/mid-field boundary and (for lifting flows), a vortex centered somewhere in the near field</td>
</tr>
<tr>
<td></td>
<td>• near-field is solved using finite-difference methods</td>
</tr>
<tr>
<td></td>
<td>• mid-field is solved using finite-element methods</td>
</tr>
<tr>
<td></td>
<td>• present method solves transonic small-disturbance equation in mid-field and near-field</td>
</tr>
</tbody>
</table>
Figure 1.— Inlet external drag terms.
Figure 2.- External drag for various cowls from reference 33 (2-D inlet).
Figure 2.- Continued.

(b) \( M_\infty = 0.84 \).
Cowls C1 through C6 as numbered

(c) $M_\infty = 1.29.$

Figure 2.- Concluded.
Model assembly. Rectangular sideplate shown. Alternate sideplate leading edge is swept from ramp leading edge to cowl lip leading edge.

Cowl lip shapes
(a) Experimental configurations.

Figure 3.- External drag for various cowls from reference 107 (2-D inlet).
Figure 3 - Continued.
Figure 3.- Concluded.
**Table 1**

<table>
<thead>
<tr>
<th>Configuration</th>
<th>$\theta_C$ (deg)</th>
<th>$r/e$ (cm)</th>
<th>$\theta_{IB}$ (deg)</th>
<th>$A_{LIP}/A_C$</th>
</tr>
</thead>
<tbody>
<tr>
<td>A9 (conical lip)</td>
<td>15</td>
<td>0.020</td>
<td>20</td>
<td>0.619</td>
</tr>
<tr>
<td>A18 (elliptical lip)</td>
<td>NACA</td>
<td>0.020</td>
<td>20</td>
<td>0.619</td>
</tr>
</tbody>
</table>

(a) Experimental configurations.

Figure 4.— External drag for two cowls from reference 27 (axisymmetric inlet).
(b) External drag data.

Figure 4. – Concluded.
Figure 5.- Comparison of the lip suction correlation of reference 10 with data from reference 33. See figure 2(a) for experimental geometry.
Figure 6.- Extrapolations required to calculate $C_{LS}$ from data of reference 33 ($M_\infty = 0.84$). See figure 2(a) for experimental geometry.
Figure 7.- Comparison of lip suction from the method of reference 32 with the data of reference 107; $\alpha = 0$, $M_\infty = 0.8$. See figure 3(a) for experimental geometry.
(c) Cowl 90-2, rectangular side plates.

(d) Cowl 20-2, rectangular side plates.

Figure 7.- Concluded.
Figure 8.- Variation of $K_{\text{add}}$ with mass flow ratio from reference 33; two-dimensional, two-ramp, external compression inlets with design Mach number = 3.0.

(a) $M_\infty = 0.7$. 

Range due to variations in cowl shape
Range due to variations in cowl shape

(b) $M_w = 1.3$.

Figure 8.- Concluded.
Figure 9.- Experimental configurations (ref. 59) for comparison of experimental and theoretical inlet drag shown in figure 10.
Figure 10. - Comparison of experimental and theoretical inlet drag for axisymmetric pitot inlets from reference 21.

External drag data

\[ C_{D_{\text{EXT}}} \]

\[ M_{\infty} = 1.41 \]

\[ C_{D_{\text{EXT}}} \]

\[ M_{\infty} = 1.61 \]

\[ C_{D_{\text{EXT}}} \]

\[ M_{\infty} = 1.81 \]

--- + Theory for Models 25-1, 50-1

Experimental Results, Reference 59

△ Model 25-1

▼ Model 50-1
Figure 11.- Comparison of experimental and theoretical inlet drag for a two-dimensional inlet from reference 21. Data from reference 33.
Figure 12. Comparison of experimental and theoretical inlet drag for a two-dimensional inlet.
(b) Comparison and data from reference 102. \( M_\infty = 1.56 \).

Figure 12. - Concluded.
Figure 13.- Comparison of experimental and theoretical inlet drag for a two-dimensional inlet from reference 21. Data from reference 102. $M_{\infty} = 1.86$. 

$C_{D_{\text{EXT}}}$ vs MFR
Rotating cowl reference line

$\Delta_2 = 4.5^\circ$

$\Delta_3 = 6.0$

$\rho = 7^\circ$

$\Delta_1 = 0^\circ$

Aircraft reference line

Figure 14.- Comparison of experimental and theoretical inlet drag from reference 224. $M_\infty = 1.5$. 

Data

---

Inlet drag calculated using method of reference 106
Figure 15.- Comparison of experimental and theoretical inlet drag from reference 224. $M_{\infty} = 1.5$. 

$\Delta_1' = 7^\circ$, $\Delta_2' = 10.9^\circ$, $\Delta_3' = 11.2^\circ$. 

- Data
- Inlet drag calculated using method of reference 106
Figure 16. - Comparison of experimental and theoretical inlet drag for SD6 inlet from reference 71.
Figure 16.- Concluded.
Figure 17.- Comparison of experimental and theoretical inlet drag for SD6 inlet with translated centerbody from reference 71.

(a) $M_\infty = 2.14$.

(b) $M_\infty = 2.48$. 
Figure 17.— Concluded.

(c) $M_\infty = 2.90$. 

Figure 17.— Concluded.
Figure 18. Comparison of experimental and theoretical inlet drag for 35 III inlet from reference 71.

(a) $M_\infty = 2.14$.

(b) $M_\infty = 2.48$.
Figure 18.- Concluded.
Figure 19.- Comparison of theoretical additive drag values from reference 372.
Figure 20. - Comparison of experimental and theoretical additive drag from reference 49.
Figure 21: Inlet geometry from reference 47.
Figure 22.- Comparison of experimental and theoretical additive drag from reference 47.

(a) $M_\infty = 1.28$.

(b) $M_\infty = 0.60$. 

Theory (ref. 33)
Figure 23.—Comparison of experimental and theoretical additive drag from reference 32. Data from reference 33.
Figure 24.- Two-dimensional inlet geometry for finite-difference calculation of reference 19.
Figure 25.— Comparison of experimental and theoretical surface pressures at $M_\infty = 0.7$, $MFR = 0.5$ from reference 19.

(a) Ramp pressures.
(b) Cowl external pressures.

Figure 25. - Continued.
Figure 25. (c) Cowl lip internal pressures. Concluded.
Figure 26.- Comparison of experimental and theoretical inlet drag for a 2-D inlet. From reference 63.

(a) $M_\infty = 0.6$.  

Inlet balance data  

Theoretical performance (ref. 37)
Figure 26.— Concluded.

(b) \( M_\infty = 0.9 \).

Inlet balance data

Theoretical performance (ref. 37)
Figure 27.- Comparison of experimental and theoretical inlet drag for an axisymmetric inlet (config. A10, ref. 52). From reference 37.

(a) $M_{\infty} = 0.7$.

(b) $M_{\infty} = 0.9$.

Data (ref. 52, config. A10)

Modified theory
Figure 28.- Pitot inlet geometry for streamtube curvature analysis of reference 274. Identified in reference 274 as NASA 1-85-100 no. 8 inlet, NACA-1 series contour.
Figure 29. - Comparisons of experimental and theoretical surface pressures for the pitot inlet from reference 274.
Figure 29. - Concluded.
Figure 30. - B-1 external compression inlet geometry from reference 239. Comparisons in following figures use $R_B = 7.2^\circ$, $H_L = 27.3$. 
Figure 31.- Comparison of experimental and theoretical surface pressures for the inlet of figure 30 from reference 239. $M_{\infty} = 0.85$, MFR = 0.55.
Figure 31.- Concluded.

(b) Cowl pressures.
Figure 32.- Comparison of experimental and theoretical surface pressures for the inlet of figure 30 from reference 239. $M_{\infty} = 0.85$, MFR = 0.75.
Figure 32.— Concluded.

(b) Cowl pressures.
Figure 33.- Comparison of experimental and theoretical additive drag for the inlet of figure 30 from reference 239. $M_\infty = 0.85$. 
- Fineness ratio, $\tilde{\pi} = L_A / D_m$
- External area ratio $A_U / A_m = (D_c / D_m)^2$
- Internal area ratio $A_o / A_t = (D_e / D_t)^2$

Figure 34. - Afterbody geometric parameters.
Afterbody drag = basic drag + (Plume shape effect) + (Plume entrainment effect)

\[ C_D = C_{DB} + \Delta C_D \]

Figure 35. Afterbody drag components.
Figure 36: Afterbody drag effects of solid cylinders vs true plumes.
Figure 37. Basic drag nozzle pressure ratio, NPRₐ as a function of boattail geometry.
(a) Subsonic \( (M_\infty < 0.85) \).

Figure 38.- Typical correlations of basic drag.
(b) Supersonic ($M_\infty = 1.10$).

Figure 38.- Concluded.
- Inverse pressure ratio is that required of a convergent nozzle to match the fully expanded plume area (theoretical) of a convergent-divergent nozzle being analyzed.

Figure 39: Corrected pressure ratio definition.
Figure 40.— Example pressure-ratio transforms.
Figure 42: Plume entrainment effect on drag.
For: $A_e/A_m = 0.53$

$FR = 2.3$
$NPR = 6$
$ER = 1.1$
$\gamma = 1.4$

$\text{(NSPR)}_{corr} = 2.6$

**Figure 43.** Example of ADAP interpolation scheme.
Figure 44. - Nonaxisymmetric nozzle data, $M_\infty = 0.9$, NPR = 2.0, $A_e/A_m = 0.25$. 

\[ C_{D_{	ext{APPROX}}} \]

Terminal boattail angle, $2$ deg

Fineness ratio - FR

Same geometry as above, same data pts., however, ordinate and abscissa are different.
Figure 45. - Data comparison: 2D nozzle data vs ADAP axisymmetric solution.
Figure 46.- Two-dimensional effect.
Figure 47.- Example of ADAP drag prediction for axisymmetric nozzles.
Figure 48.- Example of ADAP drag predictions for 2-D nozzles.

- $A_e/A_m = 0.25$
- $NPR = 2.0$
- $M_\infty = 0.85$
- $A_e/A_t = 1.0$

Data (ref. 249)

ADAP 2D solution
Figure 49.— Comparison between data and ADAP predicted drag coefficients on several circular arc boattails.

(a) $\overline{FR} = 0.8$, $A_e/A_m = 0.25$. 

$\frac{p_{t_j}}{p_\infty}$
Figure 49.- Continued.

(b) \( \overline{F} = 1.0, \frac{A_e}{A_m} = 0.25. \)

\[
\frac{P_{tj}}{P_\infty}
\]

---

330
Figure 49.- Continued.

(c) $\bar{FR} = 1.768$, $A_e/A_m = 0.25$. 

$C_D$ vs. $p_{t_j}/p_\infty$
Figure 49.- Concluded.

(d) $\overline{FR} = 1.5$, $A_e/A_m = 0.49$. 

Figure 49.- Concluded.
Figure 50. - Comparison between data and ADAP prediction for Mach number dependence of boattail-drag coefficients at $\frac{p_t}{p_{\infty}} = 3.0$.

(a) $\overline{FR} = 0.8, \frac{A_e}{A_m} = 0.25$. 
Figure 50.- Continued.

(b) $FR = 1.0, \frac{A_e}{A_m} = 0.25.$
(c) $\overline{FR} = 1.768$, $A_e/A_m = 0.25$.

Figure 50. - Continued.
Data (ref. 92) — ADAP prediction

(d) \( FR = 1.5, \frac{A_e}{A_m} = 0.49. \)

Figure 50.— Concluded.
Figure 51.— Combined drag correlation for single and twin vertical configurations.
Figure 52.- Final correlation curve for single and twin vertical configurations.
Figure 53.- ESIP drag prediction procedure.
(a) Subsonic flow.

Figure 54. - Schematic of Caluc drag prediction procedure.
Compute IMS

\[ IMS = \frac{1}{\frac{1}{\frac{A_e}{A_m}} - \frac{d\left(\frac{A_e}{A_m}\right)}{d\left(\frac{A}{A_m}\right)}} } \]

Calculate drag for equivalent body

Vertical stabilizers

\[ \frac{C_d}{C_{D_{EB}}} \]

Twin

Single

Calculate jet-off drag ratio

\[ (M_{\infty}^2 - 1) \frac{C_d}{C_{D_{EB}}} \]

Calculate jet-on drag increment

\[ \frac{\Delta D}{(P_e - P_f) \lambda_{so}} \]

Calculate boattail pressure drag coefficient (see eq. 27)

\[ C_D = f \left( \frac{C_{D_{TU}}}{C_{D_{EB}}} , \frac{\Delta D}{(P_e - P_f) \lambda_{so}} \right) \]

(b) Supersonic flow.

Figure 54. - Concluded.
Figure 55. Transonic similarity correlation of total aft-end pressure drag; single vertical stabilizer and horizontal interfairing.

Figure 56. Correlation of drag increment from design to operating pressure ratio, convergent-flap nozzle.
Figure 57.—Correlation of drag increment from jet-off to design pressure ratio.
Figure 58: IMS/supersonic similarity correlation of method of characteristics boattail pressure drag.
Figure 59.—Equivalent body correlation.

Figure 60.—Correlation of drag increment from jet-off to jet-on operation, supersonic flow.
Figure 61.- Comparison between measured and predicted-drag coefficient for cruise nozzle (ref. 41).

(a) NPR = 3.3.
(b) NPR = 4.9.

Figure 61.— Concluded.
Figure 62.- Comparison between measured and predicted-drag coefficient for reheat nozzle (ref. 41).

(a) NPR = 4.2.
Figure 62.— Continued.

(b) NPR = 5.2.
Figure 62. - Concluded.

(c) NPR = 7.8.
Figure 63.- Comparison of predicted and measured twin afterbody drag coefficients referenced to wing area.

(a) Cruise nozzle, NPR = 3.3.
(b) Cruise nozzle, NPR = 4.9.

Figure 63. - Continued.
(c) Reheat nozzle, NPR = 4.2.

Figure 63.—Continued.
(d) Reheat nozzle, NPR = 5.2.

Figure 63.—Continued.
(e) Reheat nozzle, NPR = 7.8.

Figure 63.—Concluded.
Figure 64.- Comparison of predicted and measured afterbody drag on an F-18 model.
Figure 64.-Continued.

(c) $M_\infty = 1.6$, maximum reheat nozzle.
(d) $M_{x0} = 2.0$, maximum reheat nozzle.

*Figure 64.* Concluded.
Inviscid flow (region I)

Mixing layer viscous flow (region III)

Plume boundary (under-expanded jet)

Boundary layer

Entrainment

Separation

Boattail

Nozzle

Inviscid flow (region II)

$\mathbf{M}_\infty$

$M_j > 1$

Figure 65.- Afterbody/nozzle flow-field schematic.
Figure 66. - Qualitative effect of entrainment and plume shape on boattail drag.
Figure 67. - Comparison of predicted and measured boattail pressure distribution on a conical boattail (ref. 48).
Figure 68. - Comparison of predicted and measured convergent-divergent nozzle boattail pressure distributions (ref. 48). $M_\infty = 0.6$. 

$X/L_A$ - Axial distance to length ratio

$C_p$ - Pressure coefficient

Solution with boundary layer

Test data
Figure 69. - Comparison of predicted and measured plug nozzle boattail pressure distributions (ref. 48). $M_\infty = 0.6$. 

**Figure 69.** Comparison of predicted and measured plug nozzle boattail pressure distributions (ref. 48). $M_\infty = 0.6$. 

$C_p$ - Pressure coefficient

$X/L_A$ - Axial distance to length ratio
Figure 70. - Comparison of measured pressure distributions on an elliptical afterbody with calculations from the method of Presz (ref. 253).
Figure 71.- Measured and calculated pressure drag coefficients of an elliptical afterbody (ref. 253).

\[ M_\infty = 0.5 \]
\[ P_T = 30 \text{ psia} \]
\[ (\delta^*/R)_o = 0.04 \]
\[ D_{\text{MAX}}/D_{\text{MIN}} = 2.0 \]
Figure 72.- Comparison of measured and predicted afterbody pressures for a circular arc conical afterbody configuration at $M_a = 0.601$ and a Reynolds number of $4.29 \times 10^7$. 

$C_{D, EXP} = 0.013$

$C_{D, ANAL} = 0.019$

Analysis (ref. 264)

Experiment (ref. 21)
Figure 73. - Comparison of measured and predicted afterbody pressures for a circular arc nozzle configuration at $M_\infty = 0.397$. 

$C_{D,\text{EXP.}} = 0.021$

$C_{D,\text{ANAL.}} = 0.012$
Figure 74.- Comparison of measured and predicted afterbody pressures for a circular arc nozzle configuration at $M_{\infty} = 0.6$. 

$C_D, \text{EXP} = 0.024$

$C_D, \text{ANAL} = 0.017$
Figure 75.- Comparison of measured and predicted afterbody pressures for a circular arc afterbody at $M_\infty = 0.6$ and $NPR = 2.91$. 
Figure 76.— Comparison of measured and predicted afterbody pressures for a circular arc afterbody at $M_\infty = 0.598$ and NPR = 2.002.
Figure 77.- Comparison of theoretical and experimental boattail pressure distributions.
Figure 78. - Variation of discriminating streamline separation angle with local Mach number at separation (ref. 274).
Figure 79. - Body geometry for sample calculation from reference 218.
Figure 80.- Comparison of theory with data on a circular arc cone boattail from reference 218.

(a) \( M_\infty = 0.6 \).

(b) \( M_\infty = 0.9 \).
Figure 81.- Comparison of calculated and measured afterbody drag coefficients on the body of figure 79 (ref. 218).
All dimensions are referenced to the maximum diameter.

(a) Forebody.

(b) Boattail configuration 1.

(c) Boattail configuration 2.

Figure 82.- Body with circular arc boattail (ref. 92).
Figure 83. Comparison between calculated and measured pressure coefficient distributions on boattail-sting configuration 2 (ref. 300).
Figure 84.- Comparison between calculated and measured pressure coefficient distributions on boattail-sting configuration 1 (ref. 300).
Figure 85. - Comparison between the theory and experimental data for integrated boattail drag coefficient on boattails with solid plume simulators (ref. 300).
Figure 86. - Comparison between calculated and measured pressure coefficient distributions for separated flow on configuration 1 with exhaust plume simulated by high pressure air (ref. 300).
Figure 87. — Comparison of experimental and calculated boattail drag on configuration 1 with plume simulated with high pressure air (ref. 300).
Figure 88. - Definition of flow field geometry.
Empirical method, reference 398

(a) Local Mach number.

(b) Local total flow angle, $M_\infty = 3.0$.

Range of experimental data
(ref. 399) $M_\infty = 2, 3, 4$ $\gamma = 5^\circ, 15^\circ$

Experimental data, reference 400
$M_\infty = 3.50$ $X/D = 5.0$

Empirical method, reference 398

(c) Local roll angle, $M_\infty = 3.0$.

Figure 89.- Data comparisons of local flow quantities at $X/D = 6.0$, $r/R = 1.8$ for a 3.0/l von Kármán ogive-cylinder.
Figure 90.—Wing-body configurations modeled in prediction of local flow fields using method of reference 303.
Figure 91.- Flow field under the wing of WB1 at the one-third semispan location, $M_\infty = 1.5$, $\alpha = 0^\circ$, $Z = 1.37$ inches.
(b) Local upwash velocity.

Figure 91.- Concluded.
Experimental data

Ref. 303

Prediction method

(a) Local sidewash velocity.

(b) Local upwash velocity.

Figure 92. - Flow field under the wing of WB1 at the one-third semispan location, $M_\infty = 1.5$, $\alpha = 5^\circ$, $Z = 1.37$ inches.
Mach lines based on $M_\infty$

(a) Local sidewash velocity.

Figure 93. Flow field under the wing of WB1 at the one-third semispan location, $M_\infty = 2.5$, $\alpha = 0^\circ$, $Z = 1.37$ inches.
Mach lines based on $M_a$

Experimental data

Prediction method

(b) Local upwash velocity.

Figure 93.- Concluded.
Figure 94.— Flow field under the wing of WBl at the one-third semispan location, $M_{\infty} = 2.5$, $\alpha = 5^\circ$, $Z = 1.37$ inches.

(a) Local sideward velocity.
(b) Local upwash velocity.

Figure 94.- Concluded.
Figure 9.5: Flow field under the wing of W82 at $\gamma = 90^\circ$, $\mu = 0$, $Z = -3.6$ inches.

(a) Local sideward angle.

Experimental data, reference 339
Prediction method, reference 303

0.09 inch
2.1 inches
Fuselage centerline
Mach lines based on $M_v$

$X$, in.
-12 -10 -8 -6 -4 -2 0 2 4 6 8 10

$Y$, deg
(a) Local sidewash angle.

Figure 96 - Flow field under the wing of W12 at $Y = -6.6$ inches, $M_\infty = 1.61$, $a = 4^\circ$, $Z = 1.41$ inches.
(b) Local upwash angle.

Figure 96.—Concluded.
(b) Wing-body model with noncircular fuselage.

Figure 97—Concluded.
Figure 9B - Distribution of upwash 3 inches under the fuselage centerline for the circular fuselage and for noncircular addition attached to the fuselage, $M_a = 0.4$. 

(a) $\alpha = 0^\circ$. 
Figure 9B - Concluded.
Figure 99.- Distribution of upwash 3 inches under the wing-body fuselage centerline for the circular fuselage and for noncircular addition attached to the fuselage, $M_\infty = 0.4$. 

(a) $\alpha = 0^\circ$. 

(see fig. 97(b))
Figure 99.- Concluded.
Figure 100.—Flow field at the side of a fighter-type fuselage, $M_\infty = 0.9$, $\alpha = 25^\circ$.

Figure 101.—Flow field under the wing of a fighter-type wing-fuselage configuration, $M_\infty = 0.9$, $\alpha = 10^\circ$. 
Prediction method, reference 34
Experiment, reference 34

Figure 102.- Flow field at the side of a fuselage configuration, $M_{\infty} = 2.2, \alpha = 15^\circ$. 

(a) Local Mach number.
(b) Local angle of upwash.

Figure 102. - Continued.
(c) Local sidewash angle.

Figure 102.- Concluded.
Figure 103. - Shock shape for a blunt nose vehicle, 
$M_\infty = 7.4$, $\alpha = 15.3^\circ$. 
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\( M_\infty = 2.2, \alpha = 5^\circ \).
Relative inlet area at F.S. 345

(a) Tailor-Mate fuselage configuration.

- Prediction method, reference 337
- Experiment, reference 10

(b) Contour plots of flow properties at inlet face.

Figure 105. - Flow field in the vicinity of the Tailor-Mate fuselage configuration, $M_{\infty} = 2.5$, $\alpha = 15^\circ$
<table>
<thead>
<tr>
<th>Experiment</th>
<th>Prediction method, reference 383</th>
</tr>
</thead>
<tbody>
<tr>
<td>$Re_x$</td>
<td>$3.6 \times 10^5$</td>
</tr>
<tr>
<td>Shock, sonic lines</td>
<td></td>
</tr>
<tr>
<td></td>
<td>$4.2 \times 10^5$</td>
</tr>
</tbody>
</table>

Figure 106.- Comparison of calculated and experimental flow-field features for a sharp $10^\circ$ half-angle cone, $M_\infty = 7.95$, $\alpha = 24^\circ$. 
Figure 107. - Comparison of predicted and experimental pitot-pressure profiles for a sharp 5° cone, $M_\infty = 1.8, \alpha = 12.5°$.

(a) $\alpha = 168°$.

(b) $\alpha = 152°$. 
(a) Local sidewash angle, degrees.

Figure 108.- Flow field contours in the vicinity of a fuselage configuration; 
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(c) Local Mach number.

Figure 108. Concluded.
Figure 110.- Schematic representation of transonic equivalence rule (ref. 341) for slender wings and bodies.
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(c) Local upwash angle, $Y = 2.0$ in.

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(a) $\alpha = 0^\circ$. 

Predictions, reference 10 
Test results, reference 10
Figure 115. - Concluded.
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(a) $\alpha = 0^\circ$, $\beta = 0^\circ$. 

Predictions, reference 10

Test results, reference 10
Figure 116.- Continued.
Figure 116.- Concluded.
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(a) $\alpha = 0^\circ$. 

Predictions, reference 10 

Test results, reference 10
Predictions, reference 10

Test results, reference 10
(b) $\alpha = 5^\circ$.

Figure 117.- Concluded.
Figure 118.- Flow field under the strake of a wing-fuselage configuration, $M_{\infty} = 2.5$. 

(a) $\alpha = 0^\circ$, $\beta = 0^\circ$. 

Test results, reference 10
Figure 118.- Continued.

Predictions, reference 10

Test results, reference 10

(b) $\alpha = 5^\circ$, $\beta = 0^\circ$.
Figure 118.—Concluded.

(c) $\alpha = 5^\circ$, $\beta = 4^\circ$. 

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