A SIMPLIFIED ANALYSIS OF PROPULSION INSTALLATION LOSSES FOR COMPUTERIZED AIRCRAFT DESIGN

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A simplified method is presented for computing the installation losses of aircraft gas-turbine propulsion systems. The method has been programmed for use in computer-aided conceptual aircraft design studies that cover a broad range of Mach numbers and altitudes. The items computed are: inlet size, pressure recovery, additive drag, subsonic spillage drag, bleed and bypass drags, auxiliary air systems drag, boundary-layer diverter drag, nozzle boattail drag, and the interference drag on the region adjacent to multiple nozzle installations. The methods for computing each of these installation effects are described and computer codes for the calculation of these effects are furnished. The results of these methods are compared with selected data for the F-5A and other aircraft. The computer program can be used with uninstalled engine performance information which is currently supplied by a cycle analysis program. The program, including comments, is about 600 FORTRAN statements long, and uses both theoretical and empirical techniques.
NOTATION

The notation used in the following sections is defined with the corresponding FORTRAN name used in the program indicated parenthetically. Figure 1 shows the nomenclature used for the various inlet and nozzle locations. The values below are defined per engine and the drag coefficients are based on inlet capture area unless noted. The starred (*) items are required program inputs which are either user input or are supplied by another subroutine in the aircraft synthesis program.

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Code</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>A</td>
<td></td>
<td>area, ( m^2, ) ft²</td>
</tr>
<tr>
<td>(A_{\text{AUX}}/A_{\text{ENG}})</td>
<td>(AUAENG)*</td>
<td>auxiliary systems area ratio</td>
</tr>
<tr>
<td>(A_{\text{BL}}/A_c)</td>
<td>(ABLEAC)</td>
<td>bleed mass flow ratio</td>
</tr>
<tr>
<td>(A_{\text{BF}}/A_c)</td>
<td>(ABYPAC)</td>
<td>bypass mass flow ratio</td>
</tr>
<tr>
<td>(A_c)</td>
<td>(AC)</td>
<td>inlet capture area (per engine), ( m^2, ) ft²</td>
</tr>
<tr>
<td>(A_{\text{CC}})</td>
<td>(ACC)</td>
<td>area of exit nozzle (joint point between engine and fuselage)</td>
</tr>
<tr>
<td>(A_E)</td>
<td></td>
<td>area of exit, ( m^2, ) ft²</td>
</tr>
<tr>
<td>(A_{\text{EF}})</td>
<td>(AEF)*</td>
<td>engine face flow area (per engine), ( m^2, ) ft²</td>
</tr>
<tr>
<td>(A_{\text{ENG}})</td>
<td>(AENG)*</td>
<td>engine face total area (per engine), ( m^2, ) ft²</td>
</tr>
<tr>
<td>(A_{\text{EXIT}})</td>
<td>(AEXIT)</td>
<td>nozzle exit area (per engine), ( m^2, ) ft²</td>
</tr>
<tr>
<td>(A_{\text{NOZ,Th}})</td>
<td>(ANOZT)</td>
<td>nozzle throat area (per engine), ( m^2, ) ft²</td>
</tr>
<tr>
<td>(A_o)</td>
<td>(AO)</td>
<td>area of free-stream stream tube (per engine), ( m^2, ) ft²</td>
</tr>
<tr>
<td>(A_o/A_c)</td>
<td>(AOAC)</td>
<td>mass flow ratio of inlet (per engine), ( m^2, ) ft²</td>
</tr>
<tr>
<td>(A_s)</td>
<td>(AS)</td>
<td>projected frontal area of compression surface, ( m^2, ) ft²</td>
</tr>
<tr>
<td>(A_{\text{TH}})</td>
<td>(AT)</td>
<td>inlet throat area (per engine), ( m^2, ) ft²</td>
</tr>
<tr>
<td>(A_{\text{TH_D}})</td>
<td>(ATD)</td>
<td>inlet throat area (per engine) at ( M_{\text{DES}}, m^2, ) ft²</td>
</tr>
<tr>
<td>(A_{\text{VENT}}/A_c)</td>
<td>(AVEACD)</td>
<td>ratio of engine ventilation flow area to inlet capture area (per engine)</td>
</tr>
</tbody>
</table>

iii
\( \frac{A_{\text{WEDGE}}}{A_c} \) (AWAENG)\(^*\) boundary-layer diverter area ratio

\( A_y \) (AY) projected frontal area of compression surface forward of point of normal shock impingement, \( \text{m}^2, \text{ft}^2 \)

\( C_D \) drag coefficient

\( C_{D\text{AD}} \) (CDAD) supersonic spill additive drag coefficient

\( C_{D\text{AUX}} \) (CDAUX) auxiliary systems drag coefficient

\( C_{D\text{BL}} \) (CDBE) bleed drag coefficient

\( C_{D\text{BP}} \) (CDBP) bypass drag coefficient

\( C_{D\text{BT}} \) (CDBT) nozzle boattail drag coefficient

\( C_{D\text{DIV}} \) (CDDIV) boundary-layer diverter drag coefficient

\( C_{D\text{INF}} \) (CDI) nozzle interference drag coefficient

\( C_{D\beta} \) boattail drag coefficient based on \( A_{CC} \)

\( C_{P\text{DIV}} \) (CPCS) pressure coefficient on diverter surface

\( C_P^S \) pressure coefficient on compression surface

\( C_S \) or \( C_{D\text{SP}} \) (CS or CDADS) subsonic spill additive drag coefficient

\( C_T \) thrust coefficient

\( D_{\text{CC}} \) (DCC) nozzle diameter at customer connect, \( \text{m, ft} \)

\( D_{\text{ENG}} \) (DENG) engine face diameter, \( \text{m, ft} \)

\( g \) acceleration of gravity, \( \text{m/sec}^2, \text{ft/sec}^2 \)

\( D_E \) (DEXIT) nozzle exit diameter, \( \text{m, ft} \)

\( h \) altitude, \( \text{m, ft} \)

\( \text{IPR} \) (IPR)* inlet pressure recovery code

\( L \) distance between normal shock position and inlet lip
<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>$L/y_C$</td>
<td>(XLVD) distance between normal shock position and inlet lip ratioed to inlet capture diameter</td>
</tr>
<tr>
<td>$L_{NOZ}$</td>
<td>(XLNOZ) nozzle length, m, ft</td>
</tr>
<tr>
<td>$M$</td>
<td>Mach number</td>
</tr>
<tr>
<td>$m$</td>
<td>mass flow, kg/sec, lb/sec</td>
</tr>
<tr>
<td>$m_{AUX}$</td>
<td>auxiliary systems mass flow, kg/sec, lb/sec</td>
</tr>
<tr>
<td>$m_{BP}$</td>
<td>bypass mass flow, kg/sec, lb/sec</td>
</tr>
<tr>
<td>$M_{cone}$</td>
<td>(XMCON) compression surface Mach number</td>
</tr>
<tr>
<td>$M_{DES}$</td>
<td>(XMDES)* inlet design Mach number</td>
</tr>
<tr>
<td>$M_E$</td>
<td>exit Mach number</td>
</tr>
<tr>
<td>$m_E$</td>
<td>exit mass flow, kg/sec, lb/sec</td>
</tr>
<tr>
<td>$M_{EF}$</td>
<td>(XMEF)* engine face Mach number</td>
</tr>
<tr>
<td>$M_{EXIT}$</td>
<td>(XMEX) nozzle exit Mach number</td>
</tr>
<tr>
<td>$M_{TH}$</td>
<td>(XMT)* inlet throat Mach number</td>
</tr>
<tr>
<td>$M_{\infty}$</td>
<td>(XMO)* free-stream Mach number</td>
</tr>
<tr>
<td>$N_{ENG}$</td>
<td>(EN)* number of engines</td>
</tr>
<tr>
<td>$NPR$</td>
<td>(NPR)* nozzle pressure ratio</td>
</tr>
<tr>
<td>$P$</td>
<td>static pressure, N/m$^2$, lb/ft$^2$</td>
</tr>
<tr>
<td>$P_E$</td>
<td>exit static pressure, N/m$^2$, lb/ft$^2$</td>
</tr>
<tr>
<td>$P_{DES}$</td>
<td>(PRDES) supersonic diffuser pressure recovery at $M_{DES}$</td>
</tr>
<tr>
<td>$P_{SUB}$</td>
<td>(PRSUB) subsonic diffuser pressure recovery</td>
</tr>
<tr>
<td>$P_{SUP}$</td>
<td>(PR) supersonic diffuser pressure recovery</td>
</tr>
<tr>
<td>$P_{TOT}$</td>
<td>(PRTOT) total pressure recovery to engine face</td>
</tr>
<tr>
<td>$P_{SPIN}$</td>
<td>cone surface pressure ratio</td>
</tr>
<tr>
<td>$P_t$</td>
<td>total pressure, N/m$^2$, lb/ft$^2$</td>
</tr>
<tr>
<td>$P_{t_{BLE}}$</td>
<td>(PTBLE) bleed exit total pressure, N/m$^2$, lb/ft$^2$</td>
</tr>
</tbody>
</table>
\[ P_{t_{\text{Bypass}}} \] bypass exit total pressure, N/m², lb/ft²

\[ P_{t_{\text{EF}}} \] total pressure at engine face, N/m², lb/ft²

\[ P_{t_{\text{TH}}} \] cone static pressure at the throat, N/m², lb/ft²

\[ P_{t_{\text{NOZ}}} \] nozzle exit total pressure, N/m², lb/ft²

\[ P_{t_{\text{TH}}} \] total pressure at inlet face, N/m², lb/ft²

\[ P_{t_{\infty}} \] free-stream total pressure, N/m², lb/ft²

\[ P_{\infty} \] free-stream static pressure, N/m², lb/ft²

\[ Q \text{ or } q_{\infty} \] free-stream dynamic pressure, N/m², lb/ft²

\[ \text{SFC} \] specific fuel consumption, kg/N-hr, lb/ib-hr

\[ S/D_g \] nozzle spacing ratio

\[ S_{\text{ref}} \] wing reference area, m², ft²

\[ T \] thrust, N, lb

\[ T_g \] gross thrust per engine, N, lb

\[ T_t \] total temperature, K, R

\[ T_{t_{\text{NOZ}}} \] nozzle exit total temperature, K, R

\[ T_{t_{\infty}} \] free-stream total temperature, K, R

\[ V_E \] exit velocity, m/sec, ft/sec

\[ V_{\infty} \] free-stream velocity, m/sec, ft/sec

\[ W_a \] engine airflow, kg/sec, lb/sec

\[ X_{\text{cone}}/y_c \] distance from cone tip to inlet face ratioed to inlet capture diameter

\[ y_c \] inlet capture diameter, m, ft

\[ y_s \] diameter of inlet centerbody at inlet throat, m, ft

\[ \beta \] nozzle boattail angle, deg
ΔPR  (DELPR)*  incremental pressure recovery correction
γ  (GAMMA)  isentropic constant
λ  (LAMBDA)  angle at inlet lip between average direction of flow and longitudinal axis of inlet
ρ₀  (RHO)  free-stream static density, kg/m³, lb/ft³
θ  (THETA)  cone half angle, deg
θ_D  (THDIV)  boundary-layer diverter wedge angle, deg
θ_E  exit angle, deg (COSDE is cosine of exit angle in program)
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SUMMARY

A simplified method is presented for computing the installation losses of aircraft gas-turbine propulsion systems. The method has been programmed for use in computer-aided conceptual aircraft design studies that cover a broad range of Mach numbers and altitudes. The items computed are: inlet size, pressure recovery, additive drag, subsonic spillage drag, bleed and bypass drags, auxiliary air systems drag, boundary-layer diverter drag, nozzle boattail drag, and the interference drag on the region adjacent to multiple nozzle installations. The methods for computing each of these installation effects are described and computer codes for the calculation of these effects are furnished. The results of these methods are compared with selected data for the F-5A and other aircraft. The computer program can be used with uninstalled engine performance information which is currently supplied by a cycle analysis program. The program, including comments, is about 600 FORTRAN statements long, and uses both theoretical and empirical techniques.

INTRODUCTION

The design of advanced aircraft systems requires the consideration of many different tradeoffs and parameters to arrive at an optimum design for a particular requirement or group of requirements. One is the effect of interaction between the aerodynamics and the propulsion of these systems. Propulsion installation effects on high-speed aircraft can amount to 10 percent or more of the aircraft drag and can also degrade the propulsion thrust via inlet total-pressure recovery penalties and nozzle-flow penalties. These effects are significant in high-speed aircraft design, and thus require attention, even in early design studies.

Tradeoff studies are usually done manually or, more recently, by many large computer programs with manual communication between them. As computer capabilities have increased, it has become possible to communicate between these disciplines within the computer in an automated or integrated fashion. This integration allows computation of the trajectory of the aircraft over its entire mission, thereby providing the ability to determine the effects of various parameters and to optimize the aircraft for specific requirements subject to various constraints. The method and computer code presented in this report is intended to supply the propulsion installation losses as required in this process. The code is designed to work as part of a propulsion module.
in the framework of the Aircraft Synthesis Program, ACSYNT (fig. 2), which has been developed at the Ames Research Center (ref. 1).

The purpose of this report is to document the methods and the computer code for propulsion installation losses as presently employed in ACSYNT. Limited example comparisons of calculations with data are made and areas of further research identified. It should be emphasized that, at present, the methods are preliminary in nature and further work is needed to improve the techniques and to perform additional correlations with data.

PROGRAM PHILOSOPHY

The purpose of the Propulsion Installation Calculation (PRINC) module is to compute the air induction system and nozzle/afterbody effects in the ACSYNT program. The procedures employed in the present subroutine are general, since the methods must be applicable to a variety of inlet, engine, and nozzle types over a broad range of Mach numbers and altitudes. An additional important requirement is that the calculations be very rapid, since installation losses are computed many times (over 1000) in a run of the ACSYNT program.

Figure 3 shows a block diagram of the method. A modular approach is used so that future additions and improvements can be easily incorporated. Items computed include (1) inlet pressure recovery, (2) inlet size, (3) additive and spillage drags, (4) bleed and bypass drags, (5) auxiliary system drag, (6) boundary-layer diverter drag, (7) nozzle boattail drag, and (8) nozzle interference drag. In figure 3, those parameters listed inside the boxes are output from the various modules and those parameters listed beneath each box are required inputs to each module.

There are varied accounting approaches for the aerodynamic propulsion system and propulsion system/airframe interaction losses. The method employed in the PRINC module is to charge all losses (listed above) to the engine thrust and specific fuel consumption (SFC) as indicated in figure 4. However, the total propulsion installation drag as well as the individual propulsion-related drags are computed separately so that any desired accounting method may be adopted by the user. An available option in the program is a multiplying factor for any or all of the propulsion installation losses to adjust the level of these penalties at the user's discretion.

DESCRIPTION OF METHODS

This section documents the methods used in the propulsion installation loss module (PRINC) and diagrammed in figure 3. It is assumed, for the inlet drag calculations, that the inlet is an axisymmetric, external compression design and, for the additive drag calculation, that the surface pressures are for a cone of an average half angle of 20°. The drag coefficients computed in the following development are based on inlet capture area, except where
noted. The equations, derivations, and programming details are presented in appendix A. A FORTRAN listing of all the modules is included in appendix B.

Inlet Pressure Recovery — The inlet pressure recovery is divided into two parts, the pressure recovery in the region ahead of the inlet face and the pressure recovery in the subsonic diffuser after the inlet face. The pressure recovery in the region ahead of the inlet face is estimated by the use of the standard AIA or Military Specification 5008B methods or by the assumption of normal shock pressure recovery (appendix A). The pressure recovery versus Mach number computed by these three methods is shown in figure 5.

The subsonic diffuser pressure recovery is estimated by the empirical method of Ball (ref. 2), which gives this pressure recovery as a function of the throat Mach number, the inlet lip bluntness, and the free-stream Mach number. For the present study, the inlet lip has been assumed to be sharp and, thus, the inlet subsonic diffuser pressure recovery is independent of lip bluntness or free-stream Mach number. Also, the geometric inlet throat Mach number is equal to the effective inlet throat Mach number as described in reference 2.

A fourth method available in the program is to input the inlet total pressure recovery as a function of free-stream Mach number in tabularized form.

Inlet Sizing — The inlet face flow area is determined by a mass balance (conservation of mass) between the inlet face and the engine face. The mass flow at the engine face is determined by the requirements of the engine. The inlet face flow area is increased over that of the engine to allow for bypass, bleed, and powerplant ventilation mass-flow requirements. The free-stream stream-tube cross-sectional area is determined by a mass balance between the free stream and the inlet face. The inlet design Mach number is used to define the inlet capture area, which is equal to the free-stream stream-tube cross-sectional area at the engine's maximum power setting. The inlet capture area is held constant at off-design conditions; however, the centerbody is allowed to move so that the inlet throat Mach number is held at some specified value. No check is made on the mechanical difficulty of achieving this variation. The key assumption in this analysis is that the inlet throat Mach number is constant. The programming details of this subroutine are included in appendix A.

Additive Drag — The engine thrust is referenced to free-stream conditions. The loss in momentum of the airflow ahead of the inlet system must be accounted for in the bookkeeping system. This loss in momentum ahead of the inlet face is called "additive drag" and is a function of the inlet geometry, the free-stream Mach number, and the mass flow of the engine.

The inlet additive drag is computed by a momentum balance between the inlet face and the free stream. The cosine of average flow angle (with respect to the inlet centerline) at the inlet face is assumed to be 1.0. The inlet is assumed to be external compression (that is the normal shock is outside of the cowl lip). The inlet throat Mach number is held constant at some specified
value. The inlet geometry is assumed to be axisymmetric. The additive drag can be computed (ref. 3) from

\[
C_{D_{AD}} = \frac{2}{\gamma M_{\infty}^2} \left[ \frac{A_{TH}}{A_c} \frac{P_{T_{TH}}}{P_{\infty}} \frac{P_{T_{TH}}}{P_{T_{TH}}} (\gamma M_{TH}^2 + 1) \cos \lambda \right.
\]

\[
+ \frac{A_c - A_{TH}}{A_c} \left( \frac{P_{cone}}{P_{\infty}} - 1.0 - \frac{A_c}{A_c} \gamma M_{\infty}^2 \right) + C_S
\]

The cone pressure calculation uses a polynomial approximation presented by Lighthill (ref. 4). The subsonic spillage effect \( C_S \) is computed using an empirical technique described by Sibulkin (ref. 3). A complete description of the method is included in appendix A.

Bypass Drag — In high-Mach-number aircraft design the inlet is usually sized at the maximum design Mach number. During off-design operation at lower Mach numbers, the inlet usually has the capacity to supply an excess airflow to the engine. This excess airflow must be either taken onboard the aircraft and passed (bypassed) around the engine or diverted (spilled) around the inlet system.

The bypass drag is computed from a momentum balance between the free-stream and the bypass exit. The bypass exit nozzle can be either sonic or fully expanded. After considerable simplification (see appendix A), the momentum balance yields

\[
\frac{C_D}{(A_{bp}/A_c)} = 2 \left[ 1 - \cos \theta_E \frac{M_E}{M_{\infty}} \left( \frac{1 + 0.2 M_E^2}{1 + 0.2 M_{\infty}^2} \right)^{0.5} \right]
\]

\[
+ \left( \frac{\cos \theta_E}{0.7 M_{\infty}^2} \frac{M_{\infty}}{M_E} \left( \frac{1 + 0.2 M_E^2}{1 + 0.2 M_{\infty}^2} \right)^3 \right) \left[ \frac{1}{(P/E)/P_{\infty}^2} - \left( \frac{1 + 0.2 M_{\infty}^2}{1 + 0.2 M_E^2} \right)^{3.5} \right]
\]

where \( \gamma \) is assumed to be 1.4. If it is assumed that the bypass exit nozzle is sonic, then

\[
M_E = 1.0
\]

If it is assumed that the bypass exit nozzle is fully expanded, then

\[
P_E = P_{\infty}
\]

\[
M_E = \left[ 5 \left( \frac{P_E}{P_{\infty}} \right)^{0.286} - 1 \right]^{0.5}
\]
The bypass exit pressure recovery is assumed to be a fraction of the inlet total pressure recovery (to the engine face). Typical values for this fraction are

\[ \frac{P_{tE}}{P_{t\infty}} = K \frac{P_{t\infty}}{P_{tE}} \]

where \( 0.3 \leq K \leq 0.7 \).

**Bleed Drag** — The inlet compression ramp or cone for typical supersonic inlet designs often have a considerable length exposed to an adverse pressure gradient. This can create a boundary layer which is thick enough to cause losses in engine performance. The problem is particularly acute in regions where a shock wave interacts with this boundary layer. In order to maintain efficient engine performance, part of the boundary layer is removed on these compression surfaces in some inlets, and it is necessary to account for the momentum loss of this bleed flow. A momentum balance between the free stream and the bleed exit yields an expression similar to the bypass drag formulation. The bleed exit can be assumed to be either sonic or fully expanded. The momentum balance yields

\[
\frac{C_D}{(A_{BL}/A_c)} = 2 \left[ 1 - \cos \theta_E \frac{M_{E}}{M_{\infty}} \left( \frac{1 + 0.2M_{\infty}^2}{1 + 0.2M_{E}^2} \right)^{0.5} \right] + \left\{ \frac{\cos \theta_E}{0.7M_{\infty}^2} \left( \frac{1 + 0.2M_{\infty}^2}{1 + 0.2M_{E}^2} \right)^3 \left[ \frac{1}{P_{tE}/P_{t\infty}} - \left( \frac{1 + 0.2M_{\infty}^2}{1 + 0.2M_{E}^2} \right)^{3.5} \right] \right\}
\]

where \( \gamma \) is assumed to be equal to 1.4. If it is assumed that the bleed exit nozzle is sonic, then

\[ M_E = 1.0 \]

If it is assumed that the bleed exit nozzle is fully expanded, then

\[ P_E = P_{\infty} \]

\[ M_E = [5(P_{tE}/P_{t\infty})^{0.286} - 1]^{0.5} \]

The bleed exit pressure recovery is assumed to be a fraction of the inlet total pressure recovery (to the engine face). Typical values for this fraction are

\[ \frac{P_{tE}}{P_{t\infty}} = K \frac{P_{t\infty}}{P_{tE}} \]

where \( 0.3 \leq K \leq 0.7 \).

A complete derivation of these equations is contained in appendix A.
Auxiliary Systems Drag — The auxiliary systems drag accounts for the airflow taken into the aircraft for systems cooling and auxiliary power generation. Many aircraft have small auxiliary inlets mounted at some convenient place to serve this purpose, and the drag created can be significant. It is assumed that the total momentum of the flow into these systems is lost. Therefore the auxiliary system drag is

$$C_{DAUX} = \frac{m_{AUX}v_{\infty}}{QA_c} = \frac{1}{2} \frac{\rho_{\infty} A_{AUX} v_{\infty}^2}{V_c} = 2 \frac{A_{AUX}}{A_c}$$

where $A_{AUX}/A_c$ is the ratio of the auxiliary system inlet capture area to aircraft inlet capture area. Typical values for this quantity range from 0.005 to 0.01.

Boundary-Layer Diverter Drag — In many inlet installation systems, the inlets are located close to the aircraft's larger components (i.e., wings, fuselage) which generate regions of low momentum ahead of the inlet. The ingestion of these boundary layers into the inlet creates a nonuniform flow distribution which can cause considerable performance degradation in the engine. This problem has been avoided by the addition of a ramp (a plow) between the inlet and the boundary-layer generating surface. The turning of the flow in these systems adds drag to the aircraft, which must be accounted for. A fit of data (refs. 5 and 6) yields

$$C_{DIV} = \frac{1.2}{M_\infty} \frac{\theta D A_{WEDGE}}{20 A_c} ; \quad M_\infty > 1.55$$

$$= 0.499 \frac{\theta D A_{WEDGE}}{20 A_c} ; \quad 0.95 \leq M_\infty \leq 1.55$$

$$= 0.499 \frac{M_\infty - 0.8}{(0.95 - 0.80)} \frac{\theta D A_{WEDGE}}{20 A_c} ; \quad 0.80 \leq M_\infty \leq 0.95$$

$$= 0.0 ; \quad M \leq 0.8$$

Details on the data and a comparison with the fit are given in appendix A.

Boattail Drag — The boattail drag on the airframe back to the point where the nacelle and engine are joined (see fig. 1b) is calculated as part of the aircraft drag. The boattail drag on the portion of the engine which includes the engine nozzle after this joint is charged to the engine performance in the present accounting system. The boattail drag estimation method used is an empirical technique developed by Ball (ref. 2) from wind-tunnel data on isolated boattail nozzles. The nozzle interference drag described in the next section corrects this for installations of more than one engine. The boattail drag is based on the area at the point where the engine is joined to the airframe. The formulation is for an engine nozzle pressure ratio (engine exit
total pressure to free-stream static pressure) of 2.5; however, correction terms are included for different nozzle pressure ratios. The engine nozzle exit area is computed from the engine thermodynamic data. The boattail angle is computed from the engine diameter and the assumption that the length of the boattail is equal to the engine diameter. It is also assumed that the diameter of the boattail at the connection point between the engine and aft fuselage or nacelle is 10 percent greater than the engine diameter. A complete description of this procedure is included in appendix A.

Nozzle Interference Drag — The nozzle interference drag accounts for the drag on the base area between multiple nozzles. The independent variables are free-stream Mach number and nozzle spacing ratio $S/D_n$ (ratio of the distance between nozzle centerlines to nozzle exit diameter). The calculation technique, developed by Ball (ref. 2) from wind-tunnel data, estimates the ratio of the drag due to nozzle interference divided by ideal gross thrust at a nozzle pressure ratio of 2.5. This value is corrected to a drag coefficient based on inlet capture area. A complete description of this computation is included in appendix A.

EXAMPLE CALCULATIONS

This section presents example computations from the PRINC module of typical installation drags, net propulsive thrust, and specific fuel consumption values. After PRINC module calculations of inlet mass flow and propulsion installation drags for a simulated F-5A are presented, these results are then used to determine the overall installed thrust and SFC of an ACSYNT simulated F-5A. Comparisons are made of these results with F-5A flight test data.

Mass Flow Summary

The effect of Mach number on engine mass flow ratio $A_o/A_c$ for the PRINC module simulated F-5A is presented in figure 6. Note that the F-5A has no bleed or bypass. The spillage mass flow is the difference between $A_o/A_c = 1.0$ and the $A_o/A_c$ set by the engine (plotted). This difference would be much larger for an aircraft with a higher inlet design Mach number $M_{DES}$. The method is capable of handling bleed and bypass in the manner described in the section on bypass and bleed drag.

Total Installation Drag

Figure 7 is an example PRINC module calculation of the installation drag coefficients based on wing reference area as a function of $M_o$ for a simulated F-5A inlet system. The total installation drag coefficient is shown, as well as the various components for maximum afterburning (A/B) and military power settings. For this same inlet system, the effects of engine throttling at $M = 0.9$ and 1.2 are shown in figures 8a and b.
Net Propulsive Thrust Correlation

A comparison of the thrust calculated by the ACSYNT propulsion subroutine and the PRINC module with data determined from F-5A flight tests is shown in figures 9a and b for maximum A/B and military power settings. The results are presented for two engines over a range of Mach numbers at 10 973 m (36 000 ft). The upper portion of each figure compares the uninstalled thrust from the ACSYNT propulsion module with corresponding values from the J-85-GE-13 engine specifications (ref. 7). Both thrust values are based on the AIA standard ram recovery schedule. The table shows the percentage difference between the calculated results and data for selected Mach numbers; that is,

\[ \frac{\text{Calculated} - \text{Actual}}{\text{Actual}} \times 100 \]

The lower portion of the figure shows a comparison between the installed thrust calculated by the ACSYNT propulsion subroutine with corrections calculated by the PRINC module and flight-test modified data from reference 8. The PRINC module calculations include corrections for a pressure recovery schedule based on a corrected airflow of 20.4 kg/sec (45 lb/sec) (ref. 9) and for the following installation losses — additive drag, auxiliary systems drag, boundary-layer diverter drag, and nozzle boattail and interference drags. Bleed and bypass drags are zero. Exactly what corrections are included in the flight-test modified data of reference 8 is not clear, but it is suspected that losses for the boundary-layer diverter and the nozzle are not included. This would account for some of the overcorrection by the PRINC module. With a few exceptions, the percentage differences for both power settings are within 10 percent.

SFC Correlation

Figures 10a and b show comparisons between specific fuel consumption values from the ACSYNT propulsion subroutine and the PRINC module and data determined from F-5A flight tests. These comparisons correspond to the thrust correlations shown in figures 9a and b. As with thrust, the percentage differences are generally within 10 percent. It should be noted that the F-5A flight-test evaluation may use a different method of bookkeeping, which could account for some of the differences.

CONCLUDING REMARKS

A simplified method has been presented for computing the installation losses of aircraft gas-turbine propulsion systems. The program employs rapid and sufficiently accurate estimating procedures suitable for use in computer-aided conceptual design studies of aircraft systems over a broad range of Mach numbers and altitudes. The items which can be computed are: inlet size and pressure recovery, additive drag, subsonic spillage drag, bleed and bypass drag, auxiliary air systems drag, boundary-layer diverter drag, nozzle boattail

8
drag, and the interference drag on the region adjacent to multiple nozzle installations. The methods for computing each of these installation effects have been described and compared with either data or the results of more elaborate computing procedures. Finally, a comparison of the overall results of the method with F-5A performance specifications indicates an accuracy within about 10 percent in installed thrust and specific fuel consumption. This is considered sufficiently accurate for computerized design at the early stages of vehicle definition.
APPENDIX A

DEVELOPMENT OF PROGRAMMED EQUATIONS

This appendix contains a brief development and description of the equations that are used in the PRINC program. The equations are presented by subroutine.

INLET PRESSURE RECOVERY
(MODULES PRSUBS AND PRINL)

This section is divided into two modules, one to calculate the subsonic diffuser pressure recovery \( \text{PR}_{\text{SUB}} \) and another to calculate both the supersonic diffuser recovery \( \text{PR}_{\text{SUP}} \) and the total pressure recovery to the engine face \( \text{PR}_{\text{TOT}} \).

Subsonic Diffuser Recovery

The empirical method of reference 2 is used. For \( \gamma = 1.4 \),

\[
\text{PR}_{\text{SUB}} = \frac{P_{\text{EF}}}{P_{\text{TH}}} = 1.0 - \text{EPS} \left\{ 1.0 - \frac{1.0}{[1.0 + 0.2(M_{\text{TH}})^2]^{3.5}} \right\}
\]

where

\[
\text{EPS} = 0.37148(M_{\text{TH}})^2 - 0.231428(M_{\text{TH}}) + 0.06
\]

Supersonic Diffuser Recovery

Four different options are available for calculating the supersonic diffuser recovery:

1. AIA standard ram recovery — From reference 10, we have

\[
\text{PR}_{\text{SUP}} = \frac{P_{\text{TH}}}{P_{\infty}} = 1.0 \quad \text{for} \quad M_{\infty} \leq 1.0
\]

\[
\text{PR}_{\text{SUP}} = \frac{P_{\text{TH}}}{P_{\infty}} = 1.0 - 0.1(M_{\infty} - 1.0)^{1.5} \quad \text{for} \quad M_{\infty} > 1.0
\]
(2) Military Specification 5008B - Also from reference 10, we have

\[
\frac{PR_{SUP}}{P_{T\infty}} = \begin{cases} 
1.0 & ; \ M_{\infty} \leq 1.0 \\
1.0 - 0.075(M_{\infty} - 1.0)^{1.35} & ; \ M_{\infty} > 1.0 
\end{cases}
\]

(3) Normal shock - From reference 11, we have

\[
\frac{PR_{SUP}}{P_{T\infty}} = \begin{cases} 
1.0 & ; \ M_{\infty} \leq 1.0 \\
\left(\frac{M_{\infty}^2 + 5.0}{6M_{\infty}^2 - 1.0}\right)^{5/2} & ; \ M_{\infty} > 1.0 
\end{cases}
\]

(4) Input table of \(PR_{SUP}\) vs \(M_{\infty}\) - See program listing in appendix B.

Figure 5 shows a comparison of the first three supersonic diffuser pressure recovery schedules described above.

The particular total pressure recovery schedule to be used is selected by use of the control parameter IPR, as follows:

<table>
<thead>
<tr>
<th>IPR Code</th>
<th>Recovery schedule</th>
</tr>
</thead>
<tbody>
<tr>
<td>1,</td>
<td>AIA standard ram recovery - (\Delta PR)</td>
</tr>
<tr>
<td>2,</td>
<td>MIL Specification 5008B - (\Delta PR)</td>
</tr>
<tr>
<td>3,</td>
<td>normal shock - (\Delta PR)</td>
</tr>
<tr>
<td>4,</td>
<td>table look up</td>
</tr>
</tbody>
</table>

where \(\Delta PR\) is an input incremental pressure recovery correction. If IPR is positive the installation effects are included. If IPR is input with a minus sign, the installation effects are neglected and the thrust is corrected only for the pressure recovery losses (i.e., IPR = -1 gives AIA ram recovery - \(\Delta PR\) and no installation losses).

If IPR is input as a positive number, but preceded by a one (i.e., 11, 12, 13, or 14), the installation effects are included and the subsonic diffuser pressure recovery is computed from the empirical results of reference 2 (see subsonic diffuser recovery in the previous section). Thus, IPR = 11 gives the AIA ram recovery multiplied by \(PR_{SUB}\) with the installation effects included.
The subsonic diffuser pressure recovery is multiplied by the supersonic diffuser pressure recovery to give the total pressure recovery to the engine face. That is,

\[
\frac{P_{t_{EF}}}{P_{t_{TH}}} \times \frac{P_{t_{TH}}}{P_{t_{\infty}}} = PR_{SU_{B}} \times PR_{SU_{P}}
\]

Also in this module, the supersonic diffuser pressure recovery at the inlet design Mach number (PR_{DES} at M_{DES}) is multiplied by the subsonic pressure recovery to give the total pressure recovery to the engine face.

**INLET SIZING (MODULE SIZIN)**

This module is used to compute the inlet capture area \( A_c \). The inlet capture area is defined to be the total projected frontal area of the inlet, including the projected frontal area of the centerbody (see fig. 1). The inlet capture area is computed at the design Mach number, altitude, and power setting, and is held fixed for off-design operation.

A useful relationship which is needed in the following development is the corrected airflow per unit area, which is defined to be

\[
WFF = \frac{W}{\sqrt{\frac{t}{P_{t_{A}}}}} = g\sqrt{\frac{y}{R}} M \left( 1 + \frac{y-1}{2} M^2 \right)^{-\frac{y+1}{2(y-1)}}
\]

\[
= 0.92M \left( \frac{1}{1 + 0.2M^2} \right)^3 ; \quad y = 1.4, \quad g = 32.2, \quad \text{and} \quad R = 1716
\]

WFF(M) denotes the corrected airflow per unit area (sometimes called the weight flow function) calculated for the Mach number specified in the parenthesis. For example, \( WFF(M_{EF}) \) means the weight flow function calculated for the engine face Mach number.

**Inlet Throat Area**

For external compression inlet designs with sharp lips the inlet face flow area is equal to the inlet throat area. The inlet throat Mach number is input to the program and the engine face Mach number and engine face flow area \( A_{EF} \) are obtained from the engine description. Therefore, using conservation of mass between the engine face and inlet throat, the inlet throat area can be calculated.

\[
A_{TH} = A_{EF} \left( \frac{WFF(M_{EF})}{WFF(M_{TH})} \right) \frac{P_{t_{EF}}}{P_{t_{TH}}} \left[ 1 + \frac{A_{BP}}{A_c} + \frac{A_{VENT}}{A_c} \right]
\]
The above relation is used with the appropriate design point input values to calculate the design point inlet throat area.

Inlet Capture Area

The inlet capture area can be computed by using the conservation of mass relation between the inlet throat and the free-stream conditions. The inlet capture area is equal to the free-stream flow area (i.e., \(A_o/A_c = 1.0\)) at the inlet design point. Therefore,

\[
A_c = A_o = A_{TH} D \left[ \frac{\text{WFF}(M_{TH})}{\text{WFF}(M_{DES})} \right] \left( \frac{P_{TH}}{P_{\infty}} \right)_{\text{DES}} \left[ 1 + \left( \frac{A_{BL}}{A_c} \right)_{\text{DES}} \right]
\]

ADDITIVE DRAG (MODULE CDADDI)

The additive and subsonic spillage drag computational approach follows Sibulkin (ref. 3). The inputs and outputs of the module are shown in figure 3. If the design Mach number (MDES) is less than or equal to one, the bleed and bypass area ratios, as well as the additive and subsonic spill drag, are set equal to zero. If the design Mach number (MDES) is greater than one, the following are assumed:

1. Axisymmetric cone geometry
2. External compression inlet
3. 20° cone half angle (\(\text{THETA} = 20^\circ\)) can be varied internally
4. \(\cos \lambda = 1.0\)
5. Throat Mach number is constant at input value.

The ratio of \(A_o/A_c\) for the engine airflow is calculated to be

\[
\left( \frac{A_o}{A_c} \right)_{\text{ENG}} = \frac{\rho_o A_o V_o}{\rho_o A_c V_c} = \frac{W_a}{\rho_o A_c V_c}
\]

where

\[
\rho_o V_c = \frac{\text{WFF}(M_{\infty}) P_{\infty}}{(T_{\infty})^{1/2}}
\]
The bleed and bypass area ratios are then computed from a predetermined schedule which can be changed if desired. The schedules are currently

\[
\frac{A_{BL}}{A_c} = 0.10 \text{ SFBEP} \left( \frac{M_{DES}}{3.0} \right)^3 \left( \frac{M_\infty - 1.0}{M_{DES} - 1.0} \right)
\]

\[
\frac{A_{BP}}{A_c} = \text{SFBPP} \left[ 1.0 - \left( \frac{A_\infty}{A_{ENG}} \right) \right]^{0.5}
\]

where

SFBEP = an input scale factor for the bleed flow schedule

SFBPP = an input scale factor for the bypass flow schedule

(Note: If the bleed and/or bypass airflow schedules are changed here, they must also be changed in subroutine SIZIN.)

The ratio of \( \frac{A_o}{A_c} \) for the inlet is computed from the engine airflow characteristics and the bleed, bypass, and vent airflow characteristics:

\[
\frac{A_o}{A_c} = \left( \frac{A_\infty}{A_{ENG}} \right) (1.0 + \text{WEXWEF})
\]

where

\[
\text{WEXWEF} = \frac{\rho_c A_\infty^2}{W_a} \left( \frac{A_{BL}}{A_c} + \frac{A_{BP}}{A_c} + \frac{A_{VENT}}{A_c} \right)
\]

and

\[
\frac{A_{VENT}}{A_c} \text{ is input (0.03 is typical)}
\]

The additive drag is computed using Sibulkin's formulation (ref. 3):

\[
C_{DAD} = \frac{2}{\gamma M_\infty^2} \left[ \frac{A_{TH}}{A_c} \frac{P_{t\infty}}{P_\infty} \frac{P_{TH}}{P_{t\infty}} \frac{P_{TH}}{P_{t\infty}} (\gamma M_{TH}^2 + 1) \cos \lambda \right.
\]

\[
+ \frac{(A_c - A_{TH})}{A_c} \frac{P_{cone}}{P_\infty} - 1.0 - \frac{A_\infty}{A_c} \gamma M_\infty^2 \left] + C_S \right.
\]

14
where

\[ \frac{P_{t\infty}}{P_{\infty}} = \left( \frac{1 + M_{\infty}^2}{5} \right)^{3.5} \]

\[ \frac{P_{tTH}}{P_{t\infty}} = PR_{SUP} \]

\[ \frac{P_{TH}}{P_{tTH}} = \left( \frac{1}{1 + M_{TH}^2} \right)^{3.5} \left( \frac{5}{5} \right) \]

\[ A_{TH} = \frac{W_{a} (1.0 + WEXWEF)(T_{t\infty})^{0.5}}{WFF(M_{TH})PR_{SUP}P_{t\infty}} \]

For \( M_{\infty} \leq 1.0 \), the cone surface Mach number and cone surface pressure ratio are estimated, as follows:

\[ M_{cone} = M_{\infty} \]

\[ \frac{P_{TH}}{P_{\infty}} = \left( \frac{1}{(P_{\infty}/P_{t\infty})} \left( \frac{P_{tTH}}{P_{t\infty}} \right) \left( \frac{P_{TH}}{P_{tTH}} \right) \right) \]

where \( P_{TH} \) is the cone static pressure at the throat. The cone average pressure is

\[ P_{cone} = \frac{\left( \frac{P_{TH}}{P_{\infty}} \right) P_{\infty} + P_{\infty}}{2} \]

and the cone surface pressure ratio is

\[ PSPIN = \frac{P_{cone}}{P_{\infty}} \]

For \( M_{\infty} > 1.0 \), the cone surface pressure coefficient can be estimated using an approximation presented by Lighthill (ref. 4):
\[ CP_S = \text{cone average pressure coefficient} \]

\[ = -\theta^2 + 2\theta^2 \ln \left[ \frac{2}{(M_{\infty}^2 - 1)^{1/2}} \right] \]

\[ + 3(M_{\infty}^2 - 1)\theta^4 \left\{ \ln \left[ \frac{2}{(M_{\infty}^2 - 1)^{1/2}} \right] \right\}^2 \]

\[ - (5M_{\infty}^2 - 1)\theta^4 \left\{ \ln \left[ \frac{2}{(M_{\infty}^2 - 1)^{1/2}} \right] \right\} \]

\[ + \left[ \frac{13}{4} M_{\infty}^2 + \frac{1}{2} + \frac{(\gamma + 1)M_{\infty}^4}{(M_{\infty}^2 - 1)} \right] \theta^4 \]

where \( \theta \) is the cone half angle in radians.

The cone surface pressure ratio can be obtained from the definition of the pressure coefficient

\[ \text{PSPIN} = \frac{P_{\text{cone}}/P_\infty}{C_{P_S}} = \frac{(Q/P_\infty) + 1.0}{C_{P_S}} \]

where

\[ Q/P_\infty = 0.7M_{\infty}^2 \]

The cone surface Mach number can be approximated by using a formulation of Lighthill (ref. 4):

\[ M_{\text{cone}} = \left\{ \frac{M_{\infty}^2 \left[ \frac{1}{4} M_{\infty}^2 C_{P_S} (\gamma + 1) + 1 \right] - M_{\infty}^2 C_{P_S} \left( \frac{\gamma + 1}{4} M_{\infty}^2 C_{P_S} + 1 \right) \right\}^{1/2} \]

or, for \( \gamma = 1.4 \),

\[ M_{\text{cone}} = M_{\infty} \left[ \frac{(0.6M_{\infty}^2 C_{P_S} + 1.0) - C_{P_S}(0.35M_{\infty}^2 C_{P_S} + 1)}{(0.7M_{\infty}^2 C_{P_S} + 1)(0.1M_{\infty}^2 C_{P_S} + 1)} \right]^{1/2} \]

To complete the additive drag calculation, it is necessary to evaluate the subsonic spillage drag \( C_S \). \( C_S \) is the drag of the inlet spillage that occurs behind a normal shock. This drag is equal to zero if the free-stream Mach number is subsonic.
Using Sibulkin’s formulation (ref. 3), we have

\[ c_s = \frac{2}{\gamma M^2} \left( \frac{A_s - A_y}{A_c} \right) \left( \frac{\bar{P}/P_{cone} - 1}{P_{\infty}} \right) \]

\[ A_s = A_c - A_{TH} \cos \lambda \]  
(see fig. 1)

\[ A_y = A_c \left\{ \left[ \left( \frac{A_s}{A_c} \right)^{1/2} - \frac{L \tan \theta}{y_c} \right]^2 \right\} \]

\[ y_c = \left( \frac{A_c}{\pi} \right)^{1/2} \]

\[ \frac{L}{y_c} = K \left( 1.0 - \frac{A_o}{A} \right) \]

\[ K = f(M_{\infty}) = 0.2505M_{\infty}^2 - 1.492625M_{\infty} + 2.8921 \]
(see ref. 3, p. 7)

where \( \beta \) is the ratio of mass flow with supersonic flow at the inlet to the maximum theoretical capture area mass flow.

Note that \( \beta \) is a function of \( X_{cone}/y_c, M_{\infty}, \theta \) and, according to Sibulkin (ref. 3), \( \beta \) can be considered equivalent "in most cases" to the supercritical mass flow ratio. The supercritical mass flow ratio is presented by Barry (ref. 12) where \( \beta \) is equal to Barry's \( A_o/A \). For the present purposes, it is assumed that

\[ \beta = 1.0 \; ; \; \text{for} \; X_{cone}/y_c < 1.2 \]
\[ = 1.0 - (X_{cone}/y_c - 1.2)/(2.75 - 1.2) \; ; \; \text{for} \; X_{cone}/y_c \geq 1.2 \]

\[ \bar{F}/P_{cone} = PNSPC = (7M_{cone}^2 - 1)/6 \]

For \( M_{\infty} < 0.4 \) or \( A_o/A_c > 1.0 \),

\[ C_{AD} = 0.0 \]

\[ c_s = CD_{SP} = 0.0 \]
Figure 11 shows a comparison of additive drag coefficient as computed by the methods of reference 3 and by the PRINC program. Sibulkin (ref. 3) assumes the spike position to be a function of $M_\infty$. The PRINC method assumes a spike position that is a function of $M_\infty$ and throttle setting such that the inlet throat Mach number $M_{TH}$ is a constant at the input value.

**BYPASS AND BLEED DRAGS**

**(MODULE CDBYPA)**

This module computes the drag coefficients associated with the bypass (CDBP) and bleed (CDBL) systems. The derivation of these drag effects is the same; however, it is usually assumed that the pressure recovery for the bleed system is lower than for the bypass system.

Two assumptions may be made for the bleed and bypass exit nozzles; namely, that they are either (1) sonic nozzles, with $M_E = 1$, or (2) fully expanded nozzles, with

$$
P_E = P_\infty
$$

$$
M_E = \{5 \left(\frac{P_{TE}}{P_\infty}\right)^{0.286} - 1\}^{1/2}
$$

The assumption currently used in the bleed and bypass subroutine is that the exit nozzles are sonic; however, if it is desired to use the fully expanded assumption, the changes necessary are contained in subroutine CDBYPA as comment cards. Also, the bleed and bypass drags consider momentum losses only, and do not include any drag that may be associated with the exits themselves. The derivation of the governing equation for the bypass (or bleed) drag is discussed next.

The thrust for the bypass (or bleed) is (see fig. 4)

$$
T = (m_{E}V_{E} + P_{E}A_{E} - P_{A}A_{E}) \cos \theta_{E} - m_{BP}V_{\infty}
$$

where $( )_E = \text{exit conditions for the bypass}$ and $m_{BP} = m_{E}$ from continuity considerations.

The thrust coefficient (based on $A_c$) is

$$
C_D = -C_T = \frac{m_{BP}V_{\infty} - (m_{E}V_{E} + P_{E}A_{E} - P_{A}A_{E}) \cos \theta_{E}}{QA_c}
$$

from reference 13.
\[ F = \frac{mV + P(A)}{P} = A(1 + \gamma M^2) \]

\[ \frac{f}{p} = \frac{F}{PA} = (1 + \gamma M^2) \]

where \( F \) is stream thrust, \( A \) is area, and \( P \) is static pressure. Using the definition of dynamic pressure,

\[ Q = \frac{1}{2} \rho \infty V^2 = \frac{1}{2} \gamma M^2 p_\infty \]

and using the \( f/p \) definition, the thrust coefficient can be rewritten

\[ C_T = \frac{\cos \theta_E \left( \frac{f}{p} \right) e^{(1/2) \gamma M^2} \frac{P_t}{P_t \infty} \frac{P_t \infty}{P_t \infty} - \left[ \frac{2A_E \cos \theta_E}{\gamma M^2 A_c} + \frac{A_{BP}}{(1/2)A_c} \right] }{ (1/2) \gamma M^2 p_\infty } \]

However,

\[ \frac{m_{BP}}{E} = \rho_\infty A_{BP} V_\infty = \rho_\infty A_{BP} V_E \]

\[ \frac{f}{p} = \frac{P_{tE}}{P_{tE} \infty} \frac{P_{tE} \infty}{P_{tE} \infty} = \frac{1}{P_{tE} \infty} \]

\[ (\frac{f}{p})_E = \frac{m_{EE} + P_{EE}}{P_{EE} A_{EE}} = (1 + \gamma M_E^2) \]

and, from conservation of energy,

\[ T_{t\infty} = T_{tE} \]

Using the weight flow function, which, for \( \gamma = 1.4 \), is

\[ WFF(M) = 0.92M \left( \frac{1}{1 + 0.2M^2} \right)^3 = \frac{W_{\sqrt{T_{tE}}}}{P_t A} \]

Therefore, the ratio of the exit flow area to the free-stream flow area for the bypass (or bleed) is

\[ \frac{A_E}{A_{BP}} = \frac{0.92M}{0.92M_E} \left( \frac{1}{1 + 0.2M_E^2} \right)^3 \frac{P_t}{P_t \infty} = \frac{M_\infty}{M_E} \left( \frac{1 + 0.2M_E^2}{1 + 0.2M_\infty^2} \right)^3 \frac{P_t \infty}{P_t \infty} \]

and thus the thrust coefficient for the bypass (or bleed) is

19
or, rearranging terms and using the definition of $C_D$, gives

$$
\frac{C_D}{(A_{BP}/A_c)} = 2 \left[ 1 - \cos \theta \frac{M_E}{M_\infty} \left( \frac{1 + 0.2M_\infty^2}{1 + 0.2M_E^2} \right)^{0.5} \right] \\
+ \left\{ \frac{\cos \theta \frac{M_\infty}{M_E} \left( \frac{1 + 0.2M_\infty^2}{1 + 0.2M_E^2} \right)^3 \left[ \frac{1}{(P_t/P_t^\infty)} - \left( \frac{1 + 0.2M_\infty^2}{1 + 0.2M_E^2} \right)^{3.5} \right] }{(-\gamma/2)M_\infty^2} \right\}
$$

Note: The derivation of the bleed drag coefficient is identical to the above derivation with the exception of the appropriate subscripts.

It is currently assumed that

$$
\frac{P_tE_{Bleed}}{P_t^\infty} = 0.3 \quad \frac{P_tE_{Bypass}}{P_t^\infty} = 0.7
$$

It is also assumed that both the bleed and bypass systems have sonic exit nozzles.

Figures 12 and 13 show example calculations of bypass and bleed drag coefficients for sonic exit Mach numbers. Engine face total pressure recovery and bypass and bleed mass flow schedules for a study supersonic transport configuration from reference 14 are presented in figure 12. These values are used as inputs to the PRINC module and the calculated drag coefficients that are based on inlet capture area are shown in figure 13. The bypass results (fig. 13a) of reference 14, and the PRINC module calculations (dashed curve) are based on an exit angle of 10° and on a bypass pressure recovery that is assumed equal to the engine face recovery. The PRINC module results agree well with those of reference 14. A calculated curve from PRINC module that indicates the effects of bypass recovery and exit angle is also shown in figure 13a. PRINC module calculated bleed drag coefficients, shown in
figure 13b, are compared to reference 14 values for a recovery that is three-
tenths the engine face recovery. Again, the agreement is good. Also, the
effect of changing bleed exit angle on the PRINC module results is indicated
in the figure.

AUXILIARY SYSTEMS DRAG

(MODULE CDAUX)

This module computes the drag coefficient (based on $A_c$) associated with
the auxiliary system ($C_{DAUX}$) such as losses for cooling air for various
equipment and compartments. A description of this drag increment is given in
reference 6. For these calculations, the total momentum is assumed lost.

Therefore,

$$
C_{D_{AUX}} = \frac{m_{AUX} V_\infty}{\rho A_c V_\infty} = \frac{\rho A_{AUX} V_\infty^2}{(1/2) \rho \infty V_\infty^2 A_c}
$$

$$
= \frac{A_{AUX}}{A_c}
$$

where $A_{AUX}/A_c$ is a user input and is generally a small value on the order of
0.005 to 0.01.

BOUNDARY-LAYER DIVERTER DRAG

(MODULE CDDIVI)

This module computes the drag coefficient (based on $A_c$) of the nacelle/
airframe boundary-layer diverter system $C_{DIV}$. A diverter half angle $\theta_D$
of 20° is assumed and the ratio of diverter height to boundary-layer height is
approximately 0.5. The procedure used is to curve fit the empirical diverter
pressure coefficients from two references:

Reference 5, pg. 3-24, gives data at $M = 0.9$, 1.57 and 1.97.

Reference 6, pg. III.B.4.2, gives data at $M = 2.0$ and 3.0.

The curve fit yields the following relations:

$$
C_{D_{DIV}} = \frac{1.2 \theta_D}{M_\infty^2} \frac{A_{WEDGE}}{A_c}; \quad \text{for } M_\infty \geq 1.55
$$

$$
= 0.499 \frac{\theta_D}{20} \frac{A_{WEDGE}}{A_c}; \quad \text{for } 0.95 \leq M_\infty \leq 1.55
$$
\[ C_{D_{DIV}} = \frac{(M_\infty - 0.8)}{(0.95 - 0.80)} \frac{6 \Delta WEDGE}{20 A_c} \times 0.499; \quad \text{for } 0.80 \leq M_\infty \leq 0.95 \]

\[ = 0.0; \quad \text{for } M_\infty \leq 0.8 \]

where \( \frac{\Delta WEDGE}{A_c} \) is a user input.

Figure 14 shows a comparison of the diverter pressure coefficients computed by the PRINC module with data from the two references for various Mach numbers.

**BOATTAIl DRAG (MODULE CDBTA)**

The drag on the airframe back to the fuselage end point (the "customer connect" point, see fig. 1b) is calculated as part of the airplane drag. The drag on the portion of the engine nozzle aft of this point is defined as the boattail drag. The boattail drag is a function of the free-stream Mach number, the boattail angle, and the length of the boattail. The performance penalty for this drag is charged to the engine performance in accordance with the ACSYNT bookkeeping system. The boattail drag estimation method used here is described in reference 2. The boattail drag coefficient is based on the nozzle area per engine at the "customer connect" point in reference 2; however, the basis is changed to the inlet capture area per engine in the program. The ratio of nozzle area per engine at the customer connect to inlet capture area per engine required for the change is

\[ \frac{A_{CC}}{A_c} = \frac{\pi (D_{CC})^2}{4A_c} \]

The curve fit of drag coefficients based on \( A_{CC} \) (from ref. 2, fig. 41) yields

\[ C_{D_B} = 0.0102 \left( \frac{\theta}{16} \right) \frac{1}{(1 - M_\infty^{1.5})}; \quad \text{for } M_\infty \leq 0.95 \]

\[ C_{D_B} = \frac{1.4 \tan \theta}{M_\infty^{1.53}} \left[ 1 - \left( \frac{D}{D_{CC}} \right)^2 \right]; \quad \text{for } M_\infty \geq 1.0 \]

For Mach numbers between 0.95 and 1.0, interpolate linearly between the above relations. These equations are for a nozzle pressure ratio of 2.5.
Values for the above equations are

\[ D_{\text{ENG}} = \sqrt[4]{\frac{4A_{\text{ENG}}}{\pi}} \]

where \( A_{\text{ENG}} \) is an input from the engine calculation.

\[ D_{\text{CC}} = 1.10 \times D_{\text{ENG}} \]

\[ M_{\text{EXIT}} = \left[ \frac{\left( \frac{P_{\infty}}{P_{\text{tNOZ}}} \right)^{\frac{(\gamma-1)}{\gamma}} - 1}{\frac{P_{\infty}}{P_{\text{tNOZ}}}^{\frac{\gamma-1}{2}}} \right]^{1/2} \]

where \( \frac{P_{\infty}}{P_{\text{tNOZ}}} = 1/NPR \) which is input.

\[ A_{\text{NOZ}}^{\text{TH}} = \frac{1}{WFF(1)} \left( \frac{T_{\text{tNOZ}}}{P_{\text{tNOZ}}} \right)^\gamma \]

where \( WFF(1) \) is the weight flow function at \( M = 1.0; T_{\text{tNOZ}}, P_{\text{tNOZ}}, \) and \( W_a \) are input.

\[ \frac{A_{\text{NOZ}}^{\text{TH}}}{A_{\text{EXIT}}} = \left( \frac{\gamma + 1}{2(\gamma - 1)} \right)^\gamma M_{\text{EXIT}} \left( 1 + \frac{\gamma - 1}{2} M_{\text{EXIT}}^2 \right)^{-\frac{\gamma + 1}{2(\gamma - 1)}} \]

(see ref. 11)

\[ A_{\text{EXIT}} = \frac{A_{\text{NOZ}}^{\text{TH}}}{A_{\text{EXIT}}} \]

\[ D_{\text{g}} = \sqrt[4]{\frac{4A_{\text{EXIT}}}{\pi}} \]

Assume \( L_{\text{NOZ}} = D_{\text{ENG}} \), then

\[ \beta' = \tan^{-1} \left( \frac{D_{\text{CC}} - D_{\text{g}}}{2L_{\text{NOZ}}} \right) \text{ in radians} \]

\[ \beta = 57.3 \times \beta' \text{ in degrees} \]

To correct for nozzle pressure ratio \( NPR \), which is an input value from the engine calculation, use reference 2, figure 42:
$$\Delta C_D = 0 \quad \text{if} \quad \text{NPR} \leq 3$$
$$\Delta C_D = 0.005(\text{NPR} - 3) \quad \text{if} \quad \text{NPR} \text{ is between } 3 \text{ and } 4$$
$$\Delta C_D = 0.01(\text{NPR} - 4) + 0.005 \quad \text{if} \quad \text{NPR} \text{ is between } 4 \text{ and } 8$$

and

$$\Delta C_D = 0.045 \quad \text{if} \quad \text{NPR} \geq 8$$

The corrected $C_D$ is then

$$C_D = C_{D_{2.5}} - \Delta C_D$$

To base coefficient on capture area,

$$C_{D_{BT}} = C_D \left( \frac{A_{CC}}{A_c} \right)$$

where $A_{CC} = \pi D_{CC}^2/4$, as previously described. Finally, if

$$C_{D_{BT}} \leq 0$$

set $C_{D_{BT}} = 0$.

Figure 15a is a plot of the PRINC module computed $C_{D_{BT}}$ (based on a customer-connect area of 3 ft$^2$) for a nozzle pressure ratio of 2.5 and for two different boattail angles. Data from reference 2 is also shown (symbols) for the same conditions, indicating the ACSYNT calculations are low for Mach numbers below about 0.8.

A comparison of Boeing lightweight fighter data (ref. 2) and PRINC module calculations for the same nozzle (based on a reference area of 20.2 ft$^2$) is shown in figure 15b. The nozzle pressure ratio for the data is not known, so several values are shown for the calculations. The PRINC module overpredicts at supersonic speeds and underpredicts at subsonic Mach numbers for this nozzle configuration.

**NOZZLE INTERFERENCE DRAG (MODULE ENGCDI)**

This module calculates the interference drag on the base between multiple nozzle afterbodies. The procedure used is an interpolation between the curves ($C_{D_{T}}'$) of reference 2, figure 46, which have been tabularized and put into the program. $C_{D_{T}}'$ is the interference drag coefficient between two engines for a nozzle pressure ratio of 2.5. The independent variables are Mach number ($M_\infty$).
and nozzle spacing ratio \( \frac{S}{D_j} \), where \( S \) is the distance between adjacent nozzle centerlines and \( D_j \) is the jet diameter. The value \( \frac{S}{D_j} \) is a user input to the program. The final interference drag coefficient \( C_{D_{\text{INF}}} \) is based on capture area per engine. For a given \( \frac{S}{D_j} \) and \( \mathcal{M}_\infty \), \( C_{D_{\text{I}'}} \) is obtained from the table look up for a nozzle pressure ratio of 2.5. To determine the final \( C_{D_{\text{INF}}} \) for any given nozzle pressure ratio and capture area, the following correction is applied:

\[
C_{D_{\text{INF}}} = \left( \frac{2.5}{\text{NPR}} \right) \left( \frac{\text{N_{ENG}} - 1}{\text{N_{ENG}}} \right) \left( \frac{C_{D_{\text{I}'}} \times 2 \times T_E}{Q_A C} \right)
\]

where \( \frac{2.5}{\text{NPR}} \) is a correction for nozzle pressure ratio and \( \text{NPR} \) is input to the program from the engine calculation; \( \frac{\text{N_{ENG}} - 1}{\text{N_{ENG}}} \) is a correction for number of engines since desired output is per engine and \( \text{N_{ENG}} \) is input to the program; and \( T_E \) is gross thrust per engine for the given \( \mathcal{M}_\infty \) and power setting and is input to the program from the engine calculation.

Figure 16 is a plot ACSYNT determined \( C_{D_{\text{INF}}} \) for several values of \( \frac{S}{D_j} \) compared with the data of reference 2. The graph simply shows the accuracy of the table look up procedures while giving an indication of the magnitude and variation of the results with Mach number.

**CONTROL ROUTINE (XINLET)**

This portion of the program controls the sequence of calling the various modules. In addition, it converts all the drag coefficients to the wing reference area and to the proper number of engines, since the values from the various modules are based on capture area per engine.
APPENDIX B

MODULE LISTING

This appendix contains the FORTRAN listing for the Propulsion Installation Calculation (PRINC) module for the ACSYNT program.
SUBROUTINE COBYPA

C COMPUTES THE BYPASS AND BLEED EFFECTS
C THE ADDITIVE DRAG CALCULATION IS FOR THE TOTAL AIRFLOW
C FROZEN THE INLET. USING THIS BOOKKEEPING THE EFFECT OF
C THE BYPASS AND BLEED MUST BE ADDED IN.
C XMO = FREE STREAM MACH NO
C ABYPAC = BYPASS/AC AT FREESTREAM
C ABLEAC = BLEED/AC AT FREESTREAM
C CD = INCREMENTAL DRAG COEF FOR BLEED BASED ON AC
C CDABYPAC = INCREMENTAL DRAG COEF FOR BYPASS BASED ON AC
C PRTOT = INLET TOTAL PRESSURE RECOVERY TO ENGINE FACE
C PINF = FREESTREAM STATIC PRESSURE (PSF)
C PFC = FREESTREAM TOTAL PRESSURE (PSF)
C PTBYP = BYPASS TOTAL PRESSURE RECOVERY (.7*ENGINE FACE PRES REC)
C PTPT = TOTAL PRESSURE RECOVERY (.3*ENGINE FACE PRES REC)

C
C ASSUME EXIT ANGLE FOR BLEED AND BYPASS = 15 DEG
C XM2 = XM*XM
C LSTEM = 1.66
C PTBYP = .7*PTPT
C PTBY = PTBYP/PTU
C PTFIN = PTBYP/PINF
C IF (PTFIN .GT. 1.) GO TO 10
C CDBP = 0.
C GC TO 20
C
C ASSUME THE BYPASS EXIT IS FULLY EXPANDED, FOR A SONIC
C BYPASS NOZZLE SET XM:BY (NOZZLE EXIT MACH NO) = 1.
C XM:BY = SQRT (((PTBYP/PINF) ** .286 - 1.))
C ASSUME A SONIC EXIT

C XM:BY=1.
XMEM0 = (1. + 2.*XM:BY*XMEM1)/(1. + 2.*XM:BY*XM:BY)
CDBP2 = 2.*((1. - (XMEM0*XMEM1)*XMEM0**(-.5)))*COSD)
CDBP = (CDBP2 + (COSD/(.7*XMO2)*XM0/XM:BY*(XMEM0**3)*((1./PTBYP
1 -(XMEM0**3)**(-3.5))))
CDBP = CDBP2 * ABYPAC
C IF (CDBP .LE. 0.) CDBP = 0.
C
C FOR BLEED
C
C 2C FETY = 3.*PTPT
C PTLBL = PTLFL/PT
C PTLFL = PTLFL/PINF
C IF (PTFL .GT. 1.) GO TO 30
C CDE = 0.
C RETURN
C
C ASSUME THE BYPASS EXIT IS FULLY EXPANDED, FOR A SONIC

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SUBROUTINE BLOOZ

C BLEED NOZZLE SET XME3E (BLEED NOZZLE EXIT MACH) = 1.
C XME3E = 50*(1.0 - (PI/2)**2)**286/1.0)
C ASSUME A SONIC EXIT NOZZLE FOR THE BLEED

3C XME3E = 1.
XME3E = (1.0 + 2.0*XME3E*XME3E)/(1.0 + 2.0*XME3E)
CME2 = 2.0*(1.0 - (XME3E/XME3E)**(XME3E**(-5))*COSDE1)
CME1 = CME2 + (COSDE1*(1.0 - XME3E)**(1.0/XME3E)**(XME3E**3)**(1.0/PTBEPT
1 - (XME3E**(-5)))))
CME1 = CME1*XME3E
IF (CME1 LE 0.) CME1 = 0.
IF (CME1 LE 0.) CME1 = 0.
RETURN
END
SUBROUTINE P=SUAS(XMT,PRSU)

C

C COMPUTES THE SUBSONIC DIFFUSER PRESSURE RECOVERY

C

XMT2=XMT*XMT
LPS=.37148*XMT2-.221426*XMT+.6
FRSL=1.-EPS*(1.-1./(1.+2.*XMT2)**3.5)
RETURN
END
SUBROUTINE CSTA(XM0,XNPR,PTMCZ,TTNOZ,AENG,CDBT,WA,AC,AEXIT,BETA) CDBTJ
C
C XM0 = STREAM YACH NUMBER
C XNPR = NOZZLE PRESSURE RATIO
C PTMCZ = NOZZLE EXIT TOTAL PRESSURE
C TTNOZ = NOZZLE EXIT TOTAL TEMPERATURE
C AENG = ENGINE FACE TOTAL AREA, SQ FT
C CTLE = UNATTAIN DRAG PER ENGINE, REFERENCED TO AC
C WA = ENGINE AIR FLOW, L/S/SEC
C AC = INLET CAPTURE AREA, SQ FT
C CDBT = DRAG CORRECTION FACTOR FOR NPR
C AEXIT = NOZZLE EXIT AREA PER ENGINE, SQ FT
C
C KF(XM0) = .52*XM0/(1.12*XM0**3)
C XM0 = SQRT((XNPR**.256-1.)/.74)
C ANTAE = SQRT(PTMCZ*XM0/WMF(XM0))
C AEXIT = .72*XM0EX/(1.**2*XM0EX*XM0EX)**3
C
C IF (ANTAE GT .0) AEXIT = ANUZ/ANTAE
C
C ASSUME
C A CUSTOMER CONNECT = 1.21*AENG
C LNOZ = LIA OF ENG
C
C LIMIT THE MAX EXHAUST DIAMETER TO CUSTOMER CONNECT DIAMETER
C ACC = AREA AT CUSTOMER CONNECT POINT, SQ FT
C
C ATESI = 1.21*AENG
C IF (AEXIT GT ATESI) AEXIT = ATESI
C DENC = 2.*SQRT(AENG/3.14159)
C DCC = 1.1*DENG
C DEXIT = 2.*SQRT(AEXIT/3.14159)
C DEXCC = JEXIT/DCC
C AEXIT = ATESI
C XM0 = XNOZ*DENG
C BETAI = ASIN(((DCC-DEXIT)/(2.4*XNOZ)))
C IF (BETAI = 1.1*L+G) BETAI = 0.
C BETAI = 57.29C7785*BETAI
C TBI = 1.44*TAN(BETAI)
C CONTBI = TBI*(1.-DEXCC*DEXCC)
C IF (XM0 = LE .95) CDBT = 6.102/(.1-XM0**1.5)*BETAI/16.
C IF (XM0 = GE .95) CDBT1 = CONTBI/XM0**1.53
C IF (XM0 = LT .95 OR XM0 GT 1.) GJ = 10
C CDBT2 = CNTBI
C CDBT = CONTBI/.755**.53
C CDBT = GJ + 2*C*(CDBT2-CDBT)*(XM0-.95)
C IF (XM0 = GE 1.1) CDBT = CDBT1
C
C SET ALL CDBT > (CDBT AT M = 1.2) EQUAL TO (CDBT AT M = 1.2)
C THEN CORRECT UNATTAIN DRAG DUE TO XNPR VARIATIONS
C
C CDBTI2 = CONTBI/.2**1.53
C IF (CDBTI2 LT CDBT) CDBT = CDBTI2
C CNPRP = (.5)
C IF (XNPR = GT 3. AND XNPR LT 4.) CNPRP = .35*(XNPR-3.)
C IF (XNPR = GE 4. AND XNPR LT 8.) CNPRP = .01*(XNPR-4.) + .05
C
C CDBTJ
IF (XNPR.GT.0.) CURNPR=.045
CDBT=CDBT-CPNPR
ACC*.75*378*DCC*UCC
CDBT*CST*ACC/AC
IF (CDBT.LT.0.) CDBT=0
RETURN
END
SUBROUTINE CDIV1

C
C FILL DATA AT M=2. IN G/D HBK PG 3.8.4.2
C ASSUME DIVERTER HEIGHT = .2
C FIT DATA AT M=9 IN INT AERU MANUAL (H/A) PG 3-24
C THDIV = DIVERTER INCLUDED ANGLE
C AREA = AREA OF DIVERTER WEDGE DIVIDED BY AC
C
THDIV=2.0
CDIV=0.0
IF (XMGT.5.0.AND.XM.1.95) CDIV1=(XM-5.0)/3.0*4.0+THDIV/20.
IF (XM.0.95.AND.XM.1.55) CDIV1=.2495*THDIV
IF (XM.1.55) CDIV1=0.0*THDIV/(XM*X5)
CDIV=CDIV1*AWENG
RETURN
END
SUBROUTINE SIZI(N, AEF, XMEF, XMT, PRDES, AC, XMES, PRSUB, SFBEP)

SUBROUTINE TL SIZE INLETS

NOTE: BLEED AND BYPASS SCHEDULES FOR INLET
NOTE THAT THESE SCHEDULES MUST BE COMPATIBLE WITH THE
SIZI N AND BYPASS SCHEDULES IN SUBROUTINE COSHL

DEF = ENGINE FACE FLOW AREA, FT^2
XMT = THROAT MACH MR
PM = SUP ASYM INLET DIFF. P
PRSC = SUB ASYM INLET DIFF. P
PRDS = DESIGN THROAT FLOW AREA, FT^2
IT = THROAT FLOW AREA, FT^2
AL = INLET COMPARE AREA, FT^2
XDES = INLET DESIGN MACH
ITPR = TOTAL PRESS RED. TO EF
SFBEP = SCALE FACTOR FOR INLET BLEED DRAG
SFPO = SCALE FACTOR FOR INLET BYPASS DRAG

WFF(XM) = X^2/(1 + 2*X*XM*XM)***3
ALACS = 1*(XMES/2.4)***3*SFBEP
ALAcC = 2.4
WFFXMT = WFF(XMT)
IT = DEF*(AFF(XMEF)/WFFXMT)*PRSUB*(1 + ALACD)
ALACD = WFF(XMES)/WFF(XMES))*PRDES*(1 + ABLACD)
IF (AC<=IT) AC = IT
RETURN
END

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SLVR bbl'l _ k ENGCOI

ENGCO0
ENGCO3C
ENGCO3C
ENGCO3C
ENGCO3C
ENGCO3C
ENGCO3C
ENGCO3C
ENGCO04
ENGCO05
ENGCO05
ENGCO05
ENGCO05
CALL TANT(SD85,CD85,SD85,CDL,4,1,NERR,DMUNL)
CALL TANT(SD75,CD75,SD75,CDL,4,1,NERR,DMUNL)
DFTAB = .1
IF = XM0 = .75
G0 TO 90

XM0 BETWEEN .85 AND .95

CALL TANT(SD95,CD95,SD95,CD85,4,1,NERR,DMUNL)
CALL TANT(SD85,CD85,SD85,CDL,4,1,NERR,DMUNL)
DFTAB = .1
DM = XM0 = .85
G0 TO 90

XM0 BETWEEN .95 AND 1.0

CALL TANT(SD10,CD10,SD85,CDL,4,1,NERR,DMUNL)
CALL TANT(SD95,CD95,SD85,CDL,4,1,NERR,DMUNL)
DFTAB = .95
DM = XM0 = .95
G0 TO 90

XM0 BETWEEN 1.0 AND 1.2

CDU = .19
CALL TANT(SD10,CD10,SD85,CDL,4,1,NERR,DMUNL)
DFTAB = .2
DF = XM0 = .1

CD11 = CDL + (CDU - CDL) * .04 / DFTAB

DETERMINE CD1 FOR THE ENGINES

CDT = (2.5 / XM0) * CD11 * 2.0 * FIP / (Q * AC)
CDT = (EN - 1.0) * CD11 / EN
RETURN
END

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SUBROUTINE: CADGDI(ATAC,XM,J,PY,XYM,XMDES,PTU,TT,CO,CDAD,CDADS,
2 APY,PAC,ALFAC,ADAC,WA,SFBPP,SBPP)

C
C AT = INLET THROAT FLOW AREA, FT*FT
C AC = INLET CAPTURE AREA, FT*FT
C XM = FREE STREAM MACH
C PF = SUPERSONIC DIFFUSER TOTAL PRESSURE RECOVERY
C XFT = INLET THROAT MACH NO
C XM5 = INLET DESIGN MACH NO
C TTC = FREE STREAM TOTAL PRESS
C TTC = FREE STREAM TOTAL TEMP
C CDAD = SUPERSONIC SPILL ADDITIVE DRAG BASED ON AC
C CDADS = SUBSONIC SPILL ADDITIVE DRAG BASED ON AC
C ADAC = AD/AC
C ALFAC = AL/AC
C APY = APY/AC
C WP = ENGINE AIRFLOW, LBS/SEC
C XMNC = CONE SURFACE MACH NUMBER
C CPFN = CONE SURFACE PRESSURE RECOVERY
C PSPAN = STATIC PRESS RATIO ACROSS N.S. AT CONE SURF MACH
C
C FTPD1(XM)=1.0+2*X*X-3.5+4*X**3+5.5*X**5
C W(XM)=.2*X/X-(1.0+2*X*X*X+3+4*X**3+5.5)
C CGP(XM)=1.0/7+7*X*X*X
C PSNPC(XM)=0.076*X*X*X
C WFXMO=APY(XM)
C ADAC=WA*SQRT(TT)/((FFXMO*PTU+AC)
C ALAC=ADAC
C CS=0.
C XMDC=XMDC
C IF (XMDC<=XT*1.) GO TO 10
C CLAC=C.
C CDAD=3.
C APY=APY.
C ALFAC=1.
C IF (XMDC<=XT*1.) GO TO 10
C
C 10 CAM=1.4
C THETA=2C.
C THET1=THETA/57.2957795
C CSUM=1.
C
C THE NEXT CARDS ARE THE BLEED AND BYPASS SCHEDULES FOR THE INLET
C THESE SHOULD BE MADE COMPATIBLE WITH THE INLET DESIGN POINT
C VALUES IN SUBROUTINE SIZIN
C
C ALAC=1.*SFBPP*(XMDES/XM)*3*(XM-1.)/(XMDES-1.)
C APY=APY*(3.*SFBPP*(1.-ADAC)
C IF (XPJ*XT<=1.) GO TO 20
C CLAC=C.
C ALAC=0.
C XMSUM=(XMDC+XTM)/2.
C PSNP=1.

20 IF (ADAC+XT*1.) APY=APY.
C ALAC=CC
C WCAP=AC*WFXMO*PTU/SQRT(TT)
C
36

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INLET GEOMETRY

\[ H = H_0 + \left( \frac{1}{2} \rho \frac{v^2}{\gamma} \right) \]
\[ A_1 = A_0 \left( 1 + \frac{x}{C} \right) \]
\[ v = v_0 \left( 1 + \frac{x}{C} \right) \]
\[ C = C_0 \left( 1 + \frac{x}{C} \right) \]

NOTE THAT THE INLET GEOMETRY HAS BEEN SPECIFIED BY ASSUMING THE
THroat Mach Number AND THE ANGLE OF THE CONE OR RAMP
A_1 = INLET THROAT LOW AREA, FT*FT
A_S = FRONTAL AREA OF INLET C/B AT INLET THROAT, FT*FT
CONE IF CONE AT INLET THROAT, FT
XACONE = DISTANCE FROM CONE TIP TO INLET THROAT, FT
Y_C = RADIUS CORRESPONDING TO INLET CAPTURE AREA, FT
XCYC = XXCYC/FYC

IF RAMP IF CONE STATIC PRESSURE USE AVERAGE OF
FREE STREAM AND THROAT STATIC PRESSURE.

\[ P_{IN} = \frac{P_{ST}}{P_{TH}} \]
\[ CP_{CS} = \frac{P_{IN}}{P_{IN,Th}} \]
\[ PSIN = \frac{P_{IN,Th}}{P_{IN}} \]

IF \( (X \cdot 10^2 \leq 1) \) \( \leq 30 \)

CONE SURFACE PRESSURE RATIO

\[ X_{M} = X_{M}/1 - 1 \]
\[ CP_{CS} = CP_{CS}/(X_{MO1}) \]
\[ ALG = \frac{A_1 - A_S}{A_1} \]
\[ \frac{\Delta a_1}{\Delta a_1} = \frac{\theta_{A1} A_1 + A_S}{A_1} \]
\[ A_{X} = X_{M} \cdot A_1 \]
\[ a_{X} = \frac{a_{X1}}{a_{X1} + a_{X2}} \]
\[ X_{M} = X_{M}/1 - 1 \]
\[ CP_{CS} = CP_{CS}/(X_{MO1}) \]

IF \( (X \cdot 10^2 \leq 4) \) \( \leq 40 \)

\[ \text{CLAO} \]
\[ \text{CDAL} \]

RETURN

4. \( CD = 0.2 \times (\text{GAM} \cdot X_{MO1}) \)
\[ CE = A_{X1} \cdot F1P*(\text{GAM} \cdot X_{M} \cdot X_{MT1} + 1) \cdot C_{D} \]
\[ C = \text{INLET} \text{ ADDITIVE DRAG} \text{ CAL WHICH CAUSES THE ADDITIVE DRAG} \text{ TO BE NON ZERO AT AURAC} = 1.0 \text{ THE NEXT CARDS INTRODUCE A} \]
\[ \text{CORRECTION TO THE ADDITIVE DRAG TO COMPENSATE FOR THIS ERROR} \]
C
AL*AC
FLF*FF*NL/FF(XMT)/PR*AE1
FSI*AC*AC*ACOSLAM
C\(d_{AC} = (\alpha_{AC} + \pi) / \text{SPIN} - 1, -\text{GAM} \times \text{XM0}1\)
C\(d_{AC} = \text{AC} \times \text{P1PC} \times (\text{GAM} \times \text{XMT} \times \text{XMT} + 1) \times \text{COSLAM}\)
C\(\text{CLAC} = \text{CDA} \times (\text{CLAD} + \text{CLADS})\)
IF (YMO*L = 1., GO TO 50)
\(xK = \frac{1.5 \times (0.1 - 1.1.492555 \times \text{XM0}2 + 2.1842}\)
C
C
\(F(x) = \text{LAC} \text{FOR} \text{MINIMUM SUPERCRITICAL SPILLAGE}\)
C\(F(x) = \text{FCN}(\text{XCOYC}, \text{XM0}, \text{TTHETA})\)
C\(1.8 = \text{PRESENT USE APPROP VALUE FOR 20 DEG CUNE AND XM0 = 1.4}\)
C
C = \text{CUNE SURFACE MACH NUMBER}
C\(B1 = X \times C1 \times \text{CPCS}\)
C\(c1 = \text{C}\)31 + 1.
C\(c2 = C132 + 1.\)
C\(c3 = C134 + 1.\)
C\(\sqrt{(1 + X \times C1 \times \text{CPCS} \times \text{B3})} / (1 + Y \times C1 \times \text{CPCS} \times \text{B3})\)
C\(\text{IF} (\text{XMCON} \leq 1, X \times \text{CUNE} = 1,\)
C\(\text{PNSPC} = \text{PNSPC} \times (\text{XMCONE})\)
C\(\text{CS} = \text{(2} \times (\text{GAM} \times \text{XM0}1)) \times (\text{AS} - \text{AY}) / \text{AC} \times (\text{PNSPC} - 1.0) \times \text{SPIN}\)
C\(\text{CEAC} = \text{CUA} \times (\text{CDAC} + \text{CDAZ}) + \text{CDOP}\)
C
C
\(\text{WANG AND ANDERSON FORMULATION IF ADDITIVE DRAG}\)
C\(\text{PEF NASA TN D-7445}\)
C
VIV = (1.2 \times \text{XMT} \times \text{XM0}) \times \sqrt{((1 + 2 \times \text{XM0}) / (1 + \text{XMT} \times \text{XMT}))}
C\(\text{CPST1} = \text{(P1PC} \times 1) \times \text{INP1NC} \times \text{XM0})\)
C\(\text{CCAD1} = 2.0 \times \text{AGAC} \times (\text{VIV} \times \text{COSLAM} - 1.) \times \text{CPST1} \times \text{ATGUSLAM} \times \text{AC} + \text{CPCS} \times \text{AS} / \text{AC}\)
C\(\text{CCAD} = \text{CDADZ} + \text{CDUCUR}\)
C\(\text{AWS} = \text{CS}\)
C\(\text{IF} (\text{AGAC} \times \text{LT} = 1.) \text{GO TO 50}\)
C\(\text{CLAD} = \text{C}\)
C\(\text{CDADS} = 3.0,\)
C\(\text{RETURN}\)
C\(\text{IF} (\text{CDAD} \times \text{LT} = 5.) \text{CDAD} = 0.\)
C\(\text{IF} (\text{CDADS} \times \text{LT} = 5.) \text{CDADS} = 0.\)
C\(\text{RETURN}\)
C
END
CORRECTING XINLET

C

CLADP=0.
CLACP=0.
CLAFP=0.
CLCFP=0.
CLCPF=0.
CLCFIP=0.
CLINP=0.
CLINFP=0.
CLINSP=0.
CL=N=0.
IF (KEYZ.EQ.2) GO TO 10

C

THE INLET IS ASSUMED TO HAVE GEOMETRY THAT CAN BE VARIED IN
SUCH A MANNER TO KEEP THE THROAT MACH NUMBER AT THE INPUT
VALUE. THE INLET MODEL IS (FOR THE PRESENT) FOR AN EXTERNAL
CLFDISSION AXISYMMETRIC INLET. IF IT IS DESIRED TO MODIFY THE CODE
IN THE MANNER TO MODIFY THE THROAT FLAT AREA SHOULD BE SET
EQUAL TO THE DESIGN VALUE AND THE SUBSONIC DIFFUSER PRESSURE
RECOVERY SHOULD BE COMPUTED FROM CONTINUITY.

C

IF (IPK.EQ.1) CALL PRSJB(XMT,PRSUB)
CALL PRINL(XMT,DELPY,PR,IPR,PRTJ,PRSUD,XMPRI,APRI,PRDES,XMDES)
IF (IPR.LT.5) RETURN
/L = .SSO7U*AC*PTO*PRTJ/SQRT(TTJ)
CALL SIMN(AFM,XMF,XMT,PRDES,AC,XMDES,PRSUB,SBEP)
CALL CAOUC1(AMC,AC,XMOP,PR,XTUM,XMDES,PTJ,TTJ,CDAD,CDADS,ABYPAC,
1 AMC,AC,ABAD,AC,W,SBEP,SBEG)
CALL CDIPACX(XMO,ABYPAC,ABAC,CE,CD,P,PTJ,T,PINF,PT0)
CALL COAUXI(AUENG,COAUX)
CALL COAIUI(XMOP,AENG,COALIV)
IF (KEYZ.EQ.1) RETURN

10 CALL COAIUI(XMOP,AENG,COALIV)
CALL ENCOI(XMOP,SGDG,CDI,FIP,Q,AC,XMOP)
RATIO=AL*ENG*PCDFAC/S
G
CLADP=CDAD*RATIO*SBADP
CLACSP=CDADSP*RATIO*SFADSP
CLCFP=CDCFP*RATIO
CCBF=CF*RATIO
CLAEUP=CALEY*RATIO*SFUXP
CDDIFP=CDIFP*RATIO*SFDP
CLCFIP=CFI*RATIO*SFIP
CLALF=CDADP+CDADSP+CDAPP+CDAPP+CDAUXP+CDDIFP
CDAFTP=CDFTP+CDIP
CLUDISP=(COAILP+CDAFTP)*SFINS
RETURN
END
SUBROUTINE CLAUXI(AUXENG, COAUX)

C FILE: INTERNAL AERODYNAMICS MANUAL, NAV, PP 7-24 SEC 7.8
C CLAUX = AUXILIARY SYSTEMS DRAG
C AUXENG = AREA OF AUXILIARY SYSTEMS DIVIDED BY AC
C
CLAUX = 2 * AUXENG
RETURN
END

ORIGINAL PAGE IS OF POOR QUALITY
SUBROUTINE PRLN(XMU,DELPR,PR,IPK,PR1UT,PR2UT,PR2UP,XMDES,XPR1,PR1L,S,
1 PR161)
C
C SUBROUTINE TO COMPUTE THE INLET TOTAL PRESSURE RECOVERY
C
C IPR = PRESSURE RECUPERATION BRANCH CODE
C = 1, ALA STANDARD
C = 2, MIL SPEC SSCRB
C = 3, NORMAL SHOCK
C = 4, TABLE LOOK-UP, PR VS MACH
C XP = FREE STREAM MACH NUMBER
C DLPR = INCREMENTAL PRESSURE RECOVERY REDUCTION
C = INLET AS A POSITIVE NUMBER
C FR1L = TOTAL PR TO INLET FACE
C
C DIMENSION XPR1(6), XPR2(6)
C COMMON:
C XM2=XMDE*XMDES
C IPR=(IPR.3.E-1) GO TO 10
C IDUM=IPR
C IPR=1.35(IPR)
C .IF (IPR.LT.3.) GO TO 25
C IDUM=IPR
C 
C Ct T: (3,4,5,6,7,8),IPR
C ALL STANDARD
C
C 3C IPR=1. 
C PRDE1=1. 
C IF (XM2.GT.1.) P=1.1-1*(XM2-1.)**1.5 
C IF (XM2.LE.0.001) PRDE1=1.1-1*(XMDES-1.)**1.5 
C GO TO 8.
C
C MIL SPEC 5505B
C 4C IPR=1. 
C PRDES=1.* 
C IF (XM2.GT.1.) P=1.1-.075*(XM2-1.)**1.35 
C IF (XMDES.GT.1.* P=1.1-.075*(XMDES-1.)**1.35 
C GO TO 67.
C
C NORMAL SHOCK
C 5C IPR=1. 
C PRDE1=1.* 
C IF (XM2.GT.1.) P=1.1+(6.*XM2/(XM2-5.))**3.5*(6.*(7.*XM2-1.))**2.5
C IF (XMDES.GT.1.) PRDES=1.1+(6.*XMDES/(XM2-5.))**3.5
C IF (XMDES.GT.1.) P=1.1+(6.*(7.-XM2-2.))**2.5
C PR=PR-DELPR
C PRDE=PRDES-DELPR 
C GO TO 120.
C
C TABLE LOOK-UP
C 7C IPR=XM21(1)
SUB R1, R3

10 X = (XPR2(I) - XPR1(I)) / (XPR2(I) - XPR1(I))

IF (XJ > XJ + XMPRI(I)) Go To 110

DE = I = 1

XJ = XJ + XMPRI(I)

1 = (XJ > XJ + XMPRI(I)) * XPR1(I) * (XJ - XPR1(I))

IF (XPR2(I + 1) - XMPRI(I)) * XPR2(I) * (XPR2(I + 1) - XPR1(I))

IF (XJ > XJ + XMPRI(I)) Go To 110

CONTINUE

P = XMPRI(I) + SLP * (XMPRI(I) - X)

IF (P > XJ) P = XJ

CONTINUE

P = XMPRI(I) + SLP * (XMPRI(I) - X)

IF (P > XJ) P = XJ

RETURN

END.
REFERENCES


Figure 1.- Nomenclature.

Note: A denotes areas

(a) Inlet.
(b) Nozzle.

Figure 1.- Concluded.
Figure 2.- Block diagram of ACSYNT.
Figure 3. - Block diagram of propulsion installation losses subroutine (PRINC); values computed are in the boxes; inputs are in parentheses below each box.
Installation drag coefficients:

1. Additive = \( \int_{II}^{I} (P - P_{\infty}) dA \)

2. Bleed = \( \frac{\dot{m}_{BL} V_\infty}{\rho_\infty A_c} \left( \dot{m}_{EVE} + \dot{m}_{E} (P - P_{\infty}) \right) \)

3. Bypass = \( \frac{\dot{m}_{BP} V_\infty}{\rho_\infty A_c} \left( \dot{m}_{EVE} + \dot{m}_{E} (P - P_{\infty}) \right) \)

4. Boattail = \( \int_{IV}^{III} (P - P_{\infty}) dA \)

5. Interference = \( \int_{IV} (P - P_{\infty}) dA \)

6. Diverter = \( \frac{\dot{m}_{DIV} V_\infty}{\rho_\infty A_c} \)

7. Auxiliary = \( \frac{\dot{m}_{AUX} V_\infty}{\rho_\infty A_c} \)

Accounting method:

\[ F_{net} = T_{INST} - D \]

\( T_{INST} \) = installed thrust

\( D \) = airframe drag

Where:

\[ T_{INST} = T_{UNINST} - (1 + 2 + 3 + 4 + 5 + 6 + 7) \times \rho_\infty A_c \]

Where:

\( T_{UNINST} \) = uninstall thrust (corrected for pressure recovery)

Figure 4.- Accounting method used in PRINC module.
Figure 5. - Supersonic diffuser pressure recovery schedules.
Figure 6.- Mass flow ratio versus Mach number for simulated F-5A with (2) J85-13 engines.

\[ h = 10973 \text{ m (36000 ft)} \]
\[ S_{\text{ref}} = 15.8 \text{ m}^2 (170 \text{ ft}^2) \]
\[ A_c = 0.083 \text{ m}^2 (0.892 \text{ ft}^2) \text{ per engine} \]

Military or maximum A/B power
\[ h = 10973 \text{ m} \ (36000 \text{ ft}^2) \]
\[ S_{\text{ref}} = 15.8 \text{ m}^2 \ (170 \text{ ft}^2) \]
\[ A_c = 0.083 \text{ m}^2 \ (0.892 \text{ ft}^2) \text{ per engine} \]

Figure 7.- Example propulsion installation drag calculated by PRINC for simulated F-5A with (2) J85-13 engines; based on \( S_{\text{ref}} \).
Figure 7. - Concluded.
h = 10973 m (36000 ft)
S_ref = 15.8 m² (170 ft²)
A_c = 0.083 m² (0.892 ft²) per engine

Military power

Figure 8.- Example installation drag versus mass flow ratio calculated by PRINC for simulated F-5A with (2) J35-11 engines; based on S_ref.
CD

h = 10973 m (36000 ft)
S_{ref} = 15.8 m^2 (170 ft^2)
A_c = 0.083 m^2 (0.892 ft^2) per engine
Military power

(a) M_x = 1.2

Figure 8. - Concluded.
Figure 9. - Thrust correlation for simulated F-5A with (2) J85-13 engines; h = 10973 m (36000 ft).

(a) Maximum afterburning.
Figure 9.- Concluded.
Figure 10. - Specific fuel consumption correlation for simulated F-5A with (2) J85-13 engines; $h = 10973$ m (36000 ft).

(a) Maximum afterburning.
(b) Military power.

Figure 10. - Concluded.
$M_a = 1.05$

- Sibulkin (TR 1187), fig. 7(b)
- ref. 3
- PRINC module ($M_{TH} = 0.85$)

$C_{DAD}$ is based on $A_C$

Cone half angle = 20°

$M_{oo} = 1.50$

$C_{DAD}$

Figure 11.- Additive drag correlations.
Figure 12. Pressure recovery and mass flow schedules for a study supersonic transport concept from reference 14.
Figure 13. Correlation of bypass and bleed drag coefficients for sonic exit Mach numbers; based on capture area.
Results from ref. 14

Figure 13.- Concluded.
Diverter wedge half angle = 20°
Diverter/boundary-layer height = 0.5

Figure 14.- Correlation of boundary-layer diverter pressure coefficient.
Nozzle pressure ratio = 2.50
$C_{DBT}$ is referenced to maximum nozzle area of 0.279 m$^2$ (3.00 ft$^2$)

Figure 15. - Correlation of nozzle boattail drag coefficient.

(a) Boattail angle effects.
Based on: $A_{\text{ref}} = 1.88 \text{ m}^2 \ (20.2 \text{ ft}^2)$, $D = 129.5 \text{ cm} \ (51 \text{ in.})$, $L = 127.0 \text{ cm} \ (50 \text{ in.})$, $\beta = 20^\circ$

(b) Nozzle pressure ratio effects.

Figure 15.- Concluded.
Figure 16.- Correlation of nozzle interference drag coefficient; nozzle pressure ratio = 2.50.