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In-Flight Thrust Determination

PREFACE

The growing need for determining in-flight propulsion system thrust employing turbofan and turbojet engines has been apparent for several years. The reasons for requiring this information, as specified in the E-33 charter, are:

- o Determination of aircraft drag
- o Problem rectification if aircraft performance is low
- o Interpolation of measured thrust and aircraft drag over a range of flight conditions by validation and development of analytical models
- o Establishment of a baseline for future aircraft and engine modifications.

In 1972, the Safety Standardization Advisory Committee of the SAE Aerospace Council, working with the SAE Propulsion Division suggested the need for improved knowledge of aircraft propulsion system in-flight thrust. Later, the U.S. Air Force Aeronautical Systems Division independently made a similar suggestion.

The Propulsion Division and the Aerospace Council concluded that the real need was to establish a forum where this subject could be discussed by knowledgeable experts on a technical basis. SAE undertook to do so. Dr. Robert Abernethy was commissioned by George Townsend, Propulsion Division Chairman, to organize the In-Flight Propulsion Measurement Committee E-33. Mr. Gary Adams, representing the Air Force, was also involved in organizing the E-33 Committee at this early stage. The first meeting took place in December, 1978.

PREPARED BY
SAE COMMITTEE E-33
IN-FLIGHT PROPULSION MEASUREMENT

The E-33 Committee endeavored to gather industry-wide expertise in in-flight thrust measurement and uncertainty analysis. The Committee was organized into Subcommittees A/B which concentrated on thrust/drag bookkeeping and thrust determination methodology, and Subcommittee C which addressed the subject of thrust determination uncertainty.

After reviewing the industry state-of-the-art, Committee E-33 determined that it would be appropriate to assemble and publish two companion Aerospace Information Reports. Subcommittee E-33A/B was organized under Chairman John Roberts to produce AIR 1703, "In-Flight Thrust Determination." Subcommittee E-33C, under Chairman Gary Adams, produced AIR 1678, "Uncertainty of In-Flight Thrust Determination." Together these reports provide a comprehensive survey of in-flight thrust determination, beginning with definitions and concluding with guidelines for planning a total program and estimating the measurement errors.

Each member of this committee worked diligently for many years to produce these two reports. We are indebted to them for their extraordinary efforts. Special mention is due the Sponsors, Bill Wallace and Jim Thompson and the Arnold Engineering Development Center that produced the figures. The member sponsoring organizations are to be commended for their support. Finally the British MIDAP Group, the AIAA Thrust-Drag Editorial Board, and our Consultants provided valuable liaison.

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SYMBOLS

Roman	Description	U.S. Common Units	SI Units
A	Area	ft^2	m^2
A_0/A_C	Inlet capture ratio = mass flow ratio	-	-
C_D	Either discharge coefficient or drag coefficient	-	-
C_G	Gross thrust coefficient	-	-
C_L	Lift coefficient	-	-
C_V	Specific thrust coefficient based on ideal flexible convergent-divergent nozzle expansion	-	-
C_X	Specific thrust coefficient based on ideal convergent nozzle expansion	-	-
D	Drag force (for subscripts see ø)	lb	N
D_{AFS}	Aircraft system drag at full-scale reference conditions	lb	N
f( )	Function of argument within ( )	-	-
F	Force or absolute stream force	lb	N
F_{EX}	Excess thrust	lb	N
F_G	Gross thrust and gauge stream force	lb	N
F_G^*	Modified gross thrust at core or bypass nozzle exit	lb	N
F_{IPF}	Installed propulsive force	lb	N
F_M	Measured test stand force	lb	N
F_N	Net thrust between stations 0 and 9 for single-stream engine	lb	N

SYMBOLS (CONTD.)

Roman	Description	U.S. Common Units	SI Units
F'_N	Overall net thrust between stations 0 and 00	lb	N
F^*_N	Modified net thrust	lb	N
$F_{N,\ int}$	Intrinsic net thrust between stations 1 and 9	lb	N
F_R	Ram drag or free stream momentum	lb	N
F_T	Trunnion thrust	lb	N
g	Gravitation constant	ft/s^2	m/s^2
GTP	Gross thrust parameter	-	-
M	Flight Mach number	-	-
N	Engine rotational speed	rpm	rpm
NPR	Nozzle pressure ratio	-	-
NPRA	Area-weighted nozzle pressure ratio	-	-
PLA	Power lever angle	deg	deg
P_s	Static pressure	lb/ft^2	N/m^2
P_{s0}	Free stream static pressure	lb/ft^2	N/m^2
P_{sB}	Nozzle base static pressure (distinguished from P_{s0})	lb/ft^2	N/m^2
P_{se}	Mean static pressure over external surface of engine	lb/ft^2	N/m^2
P_t	Total pressure	lb/ft^2	N/m^2
q	Freestream dynamic pressure	lb/ft^2	N/m^2
R	Gas constant	ft-lb/lb R	J/kg K
RNI	Reynolds number index	-	-
S	Surface or reference area	ft^2	m^2

SYMBOLS (CONTD.)

Roman	Description	U.S. Common Units	SI Units
T_t	Total temperature	R	K
V	Velocity	ft/s	m/s
V_0	Free stream velocity or flight speed	ft/s	m/s
W	Either mass flow rate or aircraft weight	slug/s lb	kg/s N
W_F	Fuel mass flow rate	slug/s	kg/s
W_{RF}	Afterburner fuel mass flow rate	slug/s	kg/s
Greek			
α	Angle of attack	deg	deg
γ	Specific heat ratio = C_p/C_v	-	-
δ	Non-dimensional pressure = P/P_{s0}, SL, STD	-	-
Δ	Parameter incremental change	-	-
η_R	Intake pressure recovery = P_{t2}/P_{t0}	-	-
θ	Non-dimensional temperature = T/T_{s0}, SL, STD	-	-
ϕ	Axial gauge force on a body or stream tube surface	lb	N
ϕ_{AB}	Axial gauge force on external surface of core engine afterbody between stations 19 and 00	lb	N
ϕ_B	Pressure area or buoyancy force in engine test facility	lb	N
ϕ_D	Scrubbing force in engine test facility	lb	N

SYMBOLS (CONTD.)

Greek	Description	U.S. Common Units	SI Units
ϕ_{FB}	Axial gauge force on engine fore-body	lb	N
ϕ_{plug}	Axial gauge force on plug surface downstream of station 9	lb	N
ϕ_{post}	Axial gauge force on post exit streamtube between stations 9 and 00	lb	N
ϕ_{pre}	Axial gauge force on pre-entry streamtube between stations 0 and 1	lb	N
ϕ_{pylon}	Axial gauge force on pylon surface within bypass streamtube	lb	N
ϕ_{s}	Pressure-area and scrubbing forces at ATF slip joint	lb	N

Subscripts	
0,1,2,etc.	Station designations (see Figure 2.1)
a and b	Test cell ambient conditions
act	Actual value (distinguished from "ideal")
AB	Afterbody (i.e., nozzle/afterbody drag)
AC	Aircraft
AVG	Average
con	Convergent ideal nozzle
calc	Calculated
eff	Effective value of V at exit from a con-di nozzle which, when multiplied by W_{act}, gives $F_{G,act}$, or effective flow area
EXH	Exhaust

SYMBOLS (CONTD.)

Subscripts	Description	U.S. Common Units	SI Units
flex-con-di	Flexible convergent-divergent ideal nozzle		
H	High-pressure engine spool		
id	Ideal value (distinguished from "actual")		
INL	Inlet		
L	Low-pressure engine spool		
max, MAX	Maximum		
min, MIN	Minimum		
ref, REF	Reference value or conditions		
s	Static conditions		
STD	Standard (temperature or pressure)		
t	Total conditions		
TRIM	Aircraft trim related		

1. INTRODUCTION:

The determination of in-flight thrust can be a complex process. Success depends upon careful planning and meticulous attention to detail throughout the test program. It is important that the participants involved (engine and airframe manufacturers, military service, government agency, etc.) agree at the outset on the definitions and the methods to be used for demonstration. Provision for flexibility and redundant methods is important to provide for unforseen difficulties with a particular method.

There are no industry or government standards for determining in-flight thrust or associated uncertainty at present. The purpose of this AIR is to present information and guidance on the selection of methodologies to predict and assess propulsion system thrust during flight development programs of conventional aircraft employing turbofan or turbojet engines. Methodologies beyond those presented are required in order to evaluate configurations such as vectored thrust or V/STOL aircraft. The document is intended to be used as a technical guide, not as a standard or legal document. A companion document, SAE AIR 1678, presents procedures for the estimation of uncertainty of in-flight thrust determination. Both documents describe comprehensive procedures and tasks for implementing the methodologies. Each program would select those tasks that are appropriate to meet its particular objectives.

This report reviews the major aspects of processes that may be used for the determination of in-flight thrust. It includes discussions of basic definitions, analytical and ground-test methods to predict installed thrust of a given propulsion system, and methods to gather data and calculate thrust of the propulsion system during the flight development program of the aircraft. Much of the treatment is necessarily brief due to space limitations. This document and the British Ministry/Industry Drag Analysis Panel (MIDAP) Guide, which SAE Committee E-33 used as a starting point, can be used to understand the processes and limitations involved in the determination of in-flight thrust.(1.1)* Application to a specific in-flight thrust determination program will require the use of many important assumptions not fully developed herein, and these assumptions must be evaluated during the conduct of the program.

This AIR is organized into nine Sections containing the following major topics:

Definitions and Basic Methodology (Section 2): This section presents the definitions necessary for the determination of in-flight thrust:

- o Gross and net thrust are defined along with the justification for the selection of particular reference planes for the definition of inlet and exhaust momenta. Modified gross thrust is introduced and explained in relation to calibration practice. Details of the force derivation and control volume are presented in Appendix A.

- o A thrust/drag bookkeeping procedure, which establishes baseline reference geometry and operating conditions for the throttle-dependent drag terms, is described and terms defined. All aspects of this bookkeeping may not be essential to all applications of in-flight thrust determination, as pointed out in Section 3 .

*References are listed in Section 9.

Definitions and Basic Methodology (Section 2) (Cont'd.):

- The various method options for determining gross and net thrust are defined, i.e., overall performance, gas path/nozzle, nozzle exit traverse and trunnion.

- The various nozzle coefficients used for determining flow and thrust in the nozzle option method are defined.

Propulsion System Installations (Section 3): This section discusses the application of thrust/drag bookkeeping for both podded and integrated engine configurations. Inlet and nozzle considerations for variable geometry and unusual installations are addressed briefly.

In-Flight Thrust Methods (Section 4): This section describes the various method options available for determining in-flight thrust:

- Overall Performance - These options require no special engine gas-path instrumentation for implementation. They are generally the same as those called "brochure" options in the MIDAP Guide.[1.1] A special case employs a cycle match computer simulation.

- Gas Path/Nozzle - These options require special calibrated instrumentation at various engine stations for implementation. A common example is the use of calibrated instrumentation at the nozzle entry station in conjunction with nozzle thrust and flow coefficients. Flow can also be determined from instrumentation at other convenient engine stations. These methods are called gas generator options in the MIDAP Guide.

- Nozzle Exit Traverse - This method uses in-flight measurements of nozzle exit pressures and temperatures to integrate into thrust and airflow. It is not a commonly used method, although there are instances of its successful implementation.

- Trunnion Method - This method involves the utilization of direct in-flight measurement of force at the engine mounts. The force measured at the mounts is a mixture of force terms that present a new set of bookkeeping problems. It has had limited application. It is the only method option which utilizes a direct force measurement in-flight.

- Examples - Typical examples are presented in the final subsection.

Calibration Techniques (Section 5): Methods for calibrating engines and sub-scale models in ground test facilities are discussed in this Section. Subjects included are:

- Engine testing in a ground level test bed (GLTB)

- Engine testing in an altitude test facility (ATF)

Calibration Techniques (Section 5) (Cont'd.):

- Scale-model nozzle testing relative to facility selection, model design and fabrication
- Correction of model data and extrapolation to the full-scale flight vehicle
- Scale-model afterbody testing required to assess throttle-dependent afterbody forces
- Scale-model inlet testing required to determine throttle-dependent spillage drag
- Propulsion simulator testing to determine throttle-dependent nozzle, inlet and interference forces.

Discussions of facility considerations, test techniques and examples are included.

Data Acquisition (Section 6): A discussion of general concepts and good practice regarding data acquisition is contained in this section. Sensor and data system requirements for both in-flight testing and testing in ground facilities are discussed for the thrust measurement options described in Section 4. Sensor requirements, signal conditioning, data recording, data processing, datum checks, and calibration are addressed.

Test Analysis For Thrust Validation (Section 7): The use of information from ground and flight test programs to isolate and correct errors and improve confidence in the results is described:

- Preflight consistency checks are a vital part of the overall performance evaluation. Individual parameters, coefficients and overall performance are compared among model and full-scale tests and the "math model" using redundant methods. Inconsistencies should be corrected prior to the flight test.
- Additional checks of engine operation and performance data are necessary after obtaining engine operating data at static and flight conditions in the aircraft. Comparisons with ground test and math models will continue with particular emphasis on verifying installation assumptions.
- A further data verification can be obtained when the aerodynamic drag derived from the propulsion measurements produces a logical drag representation of the aircraft. A more detailed discussion of the fundamentals of this process is given in Appendix C.
- Drag determination techniques which are the basis for the aerodynamic consistency checks discussed in Section 7.
- Examples - Typical examples are included in the final subsection.

Test Planning Guidelines (Section 8): The necessity for early involvement of all the participants in the formulation of an integrated test plan is discussed. Specific topics include:

- The task force concept is one method suggested for formulating and implementing a plan. The function of this group is described.
- The factors to be considered in the selection of method(s) for computing in-flight thrust and the impact which the flight test goals and data validity requirements have on the selection process are presented. Reliance on past experience and judgment and use of the uncertainty analysis are emphasized.
- An example of an integrated program plan is presented.

Three Appendices are also included:

- Appendix A - Fundamentals of Thrust/Drag Accounting

  This appendix starts from basic fundamentals to outline the principles of thrust, drag, and force accounting which are the foundation for the equations of Section 2.

- Appendix B - Unsteady Influences on Thrust Determination

  This appendix outlines methods for assessing the magnitude of transient thrust terms, including unsteady engine thermal effects and transient instrumentation measurement errors, that ordinarily are not considered when applying steady-state equations.

- Appendix C - Aerodynamic Characteristics and In-Flight Drag Determination

  This appendix provides an overview of the theoretical aspects of air vehicle aerodynamic characteristics and in-flight drag determination techniques which are the basis for aerodynamic consistency checks discussed in Section 7.

2. DEFINITIONS AND BASIC METHODOLOGY:

The in-flight forces acting on the installed propulsion system are sometimes difficult to define and are often even more difficult to evaluate. In distinguishing between in-flight thrust and drag forces, it is important that the definition of thrust is well understood by all concerned and that there is a proper accounting of all the forces acting on the propulsion system. The following paragraphs discuss the definitions of thrust, thrust/drag accounting, and other basic terms used for evaluating in-flight thrust.

Prior to discussing these terms, it is necessary to introduce a consistent station numbering system to define flow path locations ahead, within and aft of the propulsion system. Figure 2.1 presents a summary. These designations are consistent with SAE's ARP 755A and AS 681C.[2.1, 2.2]

Appendix A provides additional background information for the development of thrust/drag expressions.

2.1 Thrust:

A basic premise in establishing a useful in-flight thrust methodology is to define propulsion system thrust and associated aircraft drag so they are equal in level, non-accelerating flight. The thrust definition should also facilitate the determination of drag from excess thrust measurements in unsteady flight. To this end, the thrust of a propulsion system is defined as a force attributable to the installed engine in the absence of any losses in the external flow. Being a vector quantity, it will generally have components normal to and parallel to the flight path. For the purpose of the discussion which follows, it will be assumed that the thrust axis is parallel to the flight path and that thrust is therefore equal to drag in steady level flight.

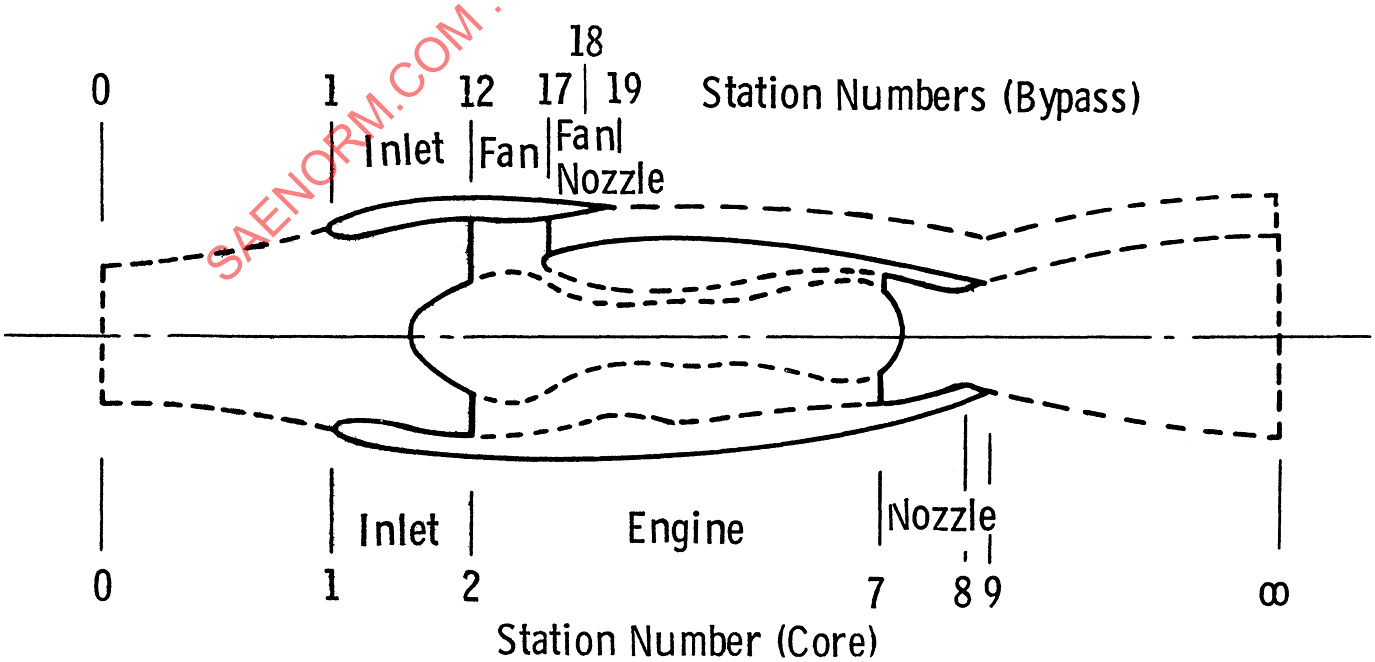


Figure 2.1 - Station Numbering System

2.1 Thrust (cont'd.):

The drag of a non-lifting body is entirely the consequence of energy dissipation in the external flow due to viscous effects resulting from skin friction, flow separation, and the formation of shock waves. In the absence of a propulsive flow, drag may be equated to a momentum defect at a station far downstream of the body where static pressures have returned to the freestream ambient value. It is the sum of drag consistent with this concept and of drag due to the generation of lift which equates to thrust in steady level flight.

A mixing process begins at the conjunction of the internal and external flows at the exhaust nozzle exit. Therefore, neither the thrust nor the drag of a propulsion system can be defined mathematically without either the hypothesis of a frictionless slip surface separating the internal and external flows downstream of the nozzle exit or the involvement of relationships defining the transfer of mass, momentum and energy between the two flows. With the hypothesis of a slip surface, drag may again be equated to a momentum defect at a station far downstream and thrust may be equated to the momentum of the exhaust stream(s) at a station far downstream of the nozzle exit station less the momentum of the internal flow at a station far upstream of the engine inlet. The overall net thrust of a single stream propulsion system is represented by the expression:

$$F_N' = W_9 \, V_{00} - W_0 \, V_0 \tag{2.1}$$

It is the equivalent of the sum of the forces $\emptyset_{pre}$, $F_{N,\ int}$, and $\emptyset_{post}$ exerted on the stream-tube bounding the flow between stations 0 and 00 (see Figure 2.2). The adopted sign convention is that forces exerted by the flow on the stream-tube boundaries are positive in the upstream direction. The intrinsic net thrust, $F_{N,\ int}$, is the force exerted between the inlet and exit stations. The forces $\emptyset_{pre}$ and $\emptyset_{post}$ are implicit in the definition of drag as a force which can be equated to a momentum defect at station 00. Their reactions must be included in the definition of thrust, if thrust is to be considered equal to drag. That is:

$$F_N' = \emptyset_{pre} + F_{N,\ int} + \emptyset_{post} \tag{2.2}$$

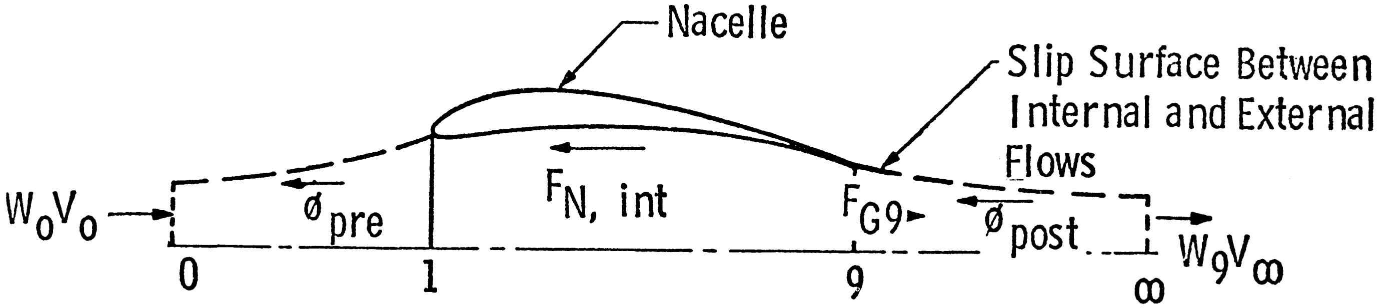


Figure 2.2 - Force Components for Simple Nacelle

2.1 Thrust (cont'd.):

A natural consequence of the definition of thrust as the change in momentum of the internal flow between stations far upstream and downstream of the engine is the automatic inclusion in thrust of all forces exerted by the internal flow on all surfaces wetted by it between those stations, whether they be upstream of the engine inlet or downstream of the nozzle exit stations. Thus for the case of a nacelle with a centerbody, Figure 2-3, the forebody, ϕ_{FB}, and plug, ϕ_{plug}, forces are included in the thrust credited to the engine. For this case, the net thrust is:

$$F_N' = \phi_{pre} - \phi_{FB} + F_{N,\ int} + \phi_{post} + \phi_{plug} \tag{2.3}$$

Figures 2.2 and 2.3 use F_G as the symbol for gauge stream force, which is the sum of momentum and gauge-pressure times area terms.

The expressions for thrust assume the existence of one-dimensional (uniform) flow at station 0 and 00. A rigorous definition applicable to the general case would involve the use of vectors and surface integrals.

An alternate and equivalent definition of thrust is:

$$F_N' = W_9\ V_9 + A_9\ (P_{s9} - P_{s0}) + \phi_{post} + \phi_{plug} - W_0\ V_0 \tag{2.4}$$

Thrust, as defined above, is called overall net thrust by the MIDAP Study Group[1.1] and is dependent on the aerodynamic design of the propulsion system and on free-stream flow effects through their influence on W_9 (and therefore W_0) and V_{00}, or alternatively on W_9, V_9, P_{s9}, ϕ_{plug} and ϕ_{post}. As a consequence, overall net thrust is not compatible with the need for a thrust definition that is independent of the external aerodynamics of the engine installation and the external flow field.

Several alternative expressions for thrust can be adopted and used as models for in-flight thrust determination. These expressions can be related to the overall net thrust, as defined by Equation 2.4, by the elimination of one or more terms. While these expressions do not conform to the definition of thrust in a strict sense, they are adequate for many purposes.

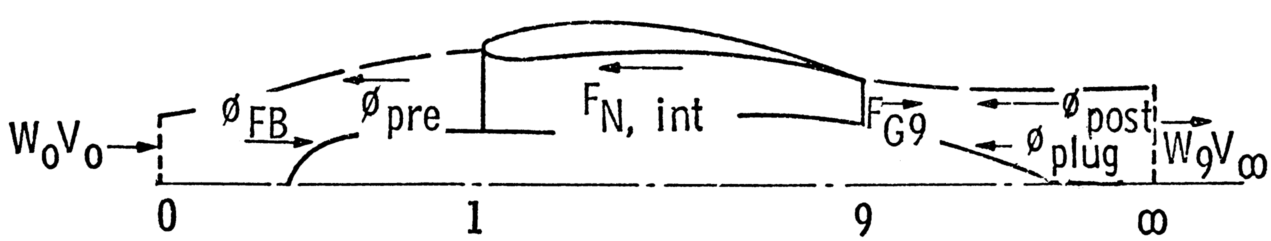


Figure 2.3 - Force Components for Centerbody Nacelle

2.1 Thrust (cont'd.):

An important definition excludes the ϕ_{post} term from net thrust. This deletion neither ignores its existance nor its effect on propulsion. Rather, as a necessity or convenience, it establishes its accounting as a "drag" term.

For the centerbody nacelle, elimination of the ϕ_{post} term from Equation 2.4 results in the expression:

$$F_N^* = W_9 V_9 + A_9 (P_{s9} - P_{s0}) + \phi_{plug} - W_0 V_0 \qquad (2.5)$$

The symbol F_N^* and the term modified standard net thrust have been assigned to this parameter by the MIDAP Study Group. The symbol F_N^* and the name modified net thrust will be used here to avoid any implication that the adoption of this definition as an aerospace standard is intended.

Elimination of the term ϕ_{plug} from the above expression for modified net thrust results in:

$$F_N = W_9 V_9 + A_9 (P_{s9} - P_{s0}) - W_0 V_0 \qquad (2.6)$$

F_N is termed standard net thrust in the MIDAP Report but will be called simply net thrust in this report. Excluding ϕ_{plug} from thrust requires it to be accounted as a drag term. This may be difficult, making modified net thrust a more convenient definition. Net thrust and modified net thrust are the same for an engine without an afterbody within the exhaust streamtube.

Thrust can be assumed to depend on external flow-field effects. Only if the effects of the external flow field are neglected is it possible to define a thrust that can be established by direct force measurement through static testing of the engine and exhaust system in a ground-level or altitude test facility. Partial recognition of the effects of the external flow on thrust may be achieved through model testing (see Section 5.2). Force measurements acquired in the presence of an external flow will invariably include elements of drag or exclude elements of thrust which cannot be determined in a direct manner.

In many instances, post exit thrust and afterbody forces are negligible; in other cases, the relegation of post-exit thrust and/or unaccounted afterbody forces to the drag account may be of no practical consequence. The degree to which external flow-field effects are properly represented in in-flight thrust determination depends entirely on the testing performed to establish installed engine or exhaust system performance characteristics. Precise accounting for external flow-field effects requires "wind-on" testing of the engine installation in a wind tunnel and the application of appropriate accounting methods to break down measured forces into thrust and drag components.

2.1 Thrust (cont'd.):

Numerous interpretations of the term "thrust" are possible. The parties involved in the determination of "thrust" in flight should reach early agreement regarding the definition of "thrust" and corresponding "drag" and of a convenient thrust/drag accounting procedure. The determination of overall net thrust cannot be accomplished without the estimation of the magnitude of one or more forces which cannot be measured directly.

Because of the difficulties in adapting overall net thrust into a practical thrust/drag accounting system, other thrust definitions have been utilized for in-flight thrust determination. Modified net thrust (Equation 2.5) is considered an acceptable definition for most aircraft propulsion systems.

The extension of the above definitions to multiple-flow propulsion systems creates no new problems. If one of the flows totally encloses another, the post exit thrust term of the enclosed flow is cancelled by an equal and opposite force exerted on the enclosing flow. An example of the application of modified net thrust to a high-bypass-ratio turbofan engine, shown in Figure 2.4, is:

$$F_N^* = W_9\ V_9 + A_9\ (P_{s9} - P_{s0}) + \phi_{plug} + W_{19}\ V_{19}$$

$$+ A_{19}\ (P_{s19} - P_{s0}) + \phi_{AB} - W_0\ V_0 \quad (2.7)$$

A common convention is to define net thrust as the vector sum of a ram drag term and an appropriate gross thrust term. Ram drag, F_R, is the free-stream momentum, $W_0\ V_0$. The gross thrust term is the sum of the remaining terms on the right-hand side of the net thrust equation. Thus, for modified net thrust:

$$F_N^* = F_G^* - F_R \quad (2.8)$$

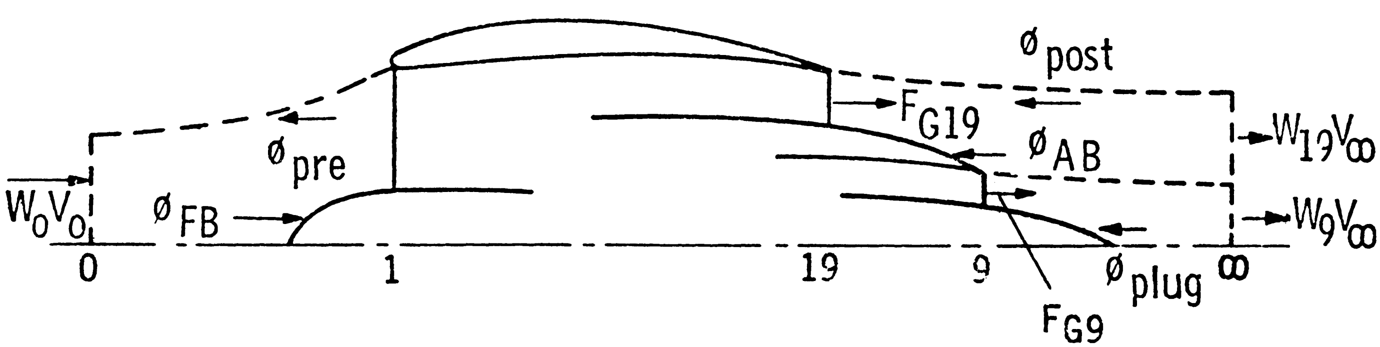


Figure 2.4 - Force Components for Dual Stream Nacelle

2.1 Thrust (cont'd.):

where: Modified gross thrust, F_G^*, for a single-stream engine is:

$$F_G^* = F_{G9} + \phi_{plug} = W_9 V_9 + A_9 (P_{s9} - P_{s0}) + \phi_{plug} \tag{2.9}$$

and for a dual-stream engine is:

$$F_G^* = F_{G9} + \phi_{plug} + F_{G19} + \phi_{AB} = W_9 V_9 + A_9 (P_{s9} - P_{s0}) + \phi_{plug} + W_{19} V_{19} + A_{19} (P_{s19} - P_{s0}) + \phi_{AB} \tag{2.10}$$

The comparable relationship for gross thrust, F_G, for a dual-stream engine is:

$$F_G = W_9 V_9 + A_9 (P_{s9} - P_{s0}) + W_{19} V_{19} + A_{19} (P_{s19} - P_{s0}) \tag{2.11}$$

The above equations are for steady-state flow conditions. Appendix B presents information to substantiate the validity of this assumption for fixed-throttle engine operation during steady-state and quasi-steady-state aircraft maneuvers.

2.2 Thrust/Drag Accounting: The measurement and validation of in-flight thrust requires definition of a thrust/drag accounting system that clearly defines the treatment of all the propulsion-related forces acting on the system[2.3, 2.4]. The variety of actual and possible propulsion systems makes it impractical to specify a single rigorous accounting system. Therefore, to obtain the most accurate and efficient bookkeeping system, the accounting methodology is tailored to the requirements of each particular system.

Neither net nor modified net thrust account for inlet, exhaust, or thrust-moment related forces. However, these significant forces are essential to thrust/drag accounting. They must be accounted for but are uniquely a function of the installation and the engine/aircraft operating conditions. Figure 2.5 illustrates forces involved in the thrust/drag accounting of an isolated, axisymmetric propulsion system. Inlet force, which results from the difference between ϕ_{pre} and ϕ_{cowl}, is a function of engine airflow. The afterbody force, which results from the difference between ϕ_{post} and ϕ_{AB}, is generally a function of nozzle pressure ratio and nozzle area. Therefore, changes in these forces, ΔF_{INL} and ΔF_{EXH}, are propulsion system dependent and must be accounted in the thrust/drag bookkeeping procedure, along with other forces such as trim forces. The above nacelle/aircraft friction and pressure forces (drags) are impractical to separate from the propulsion system forces. Fortunately, this is not mandatory as long as all forces are included either separately or as differences between forces.

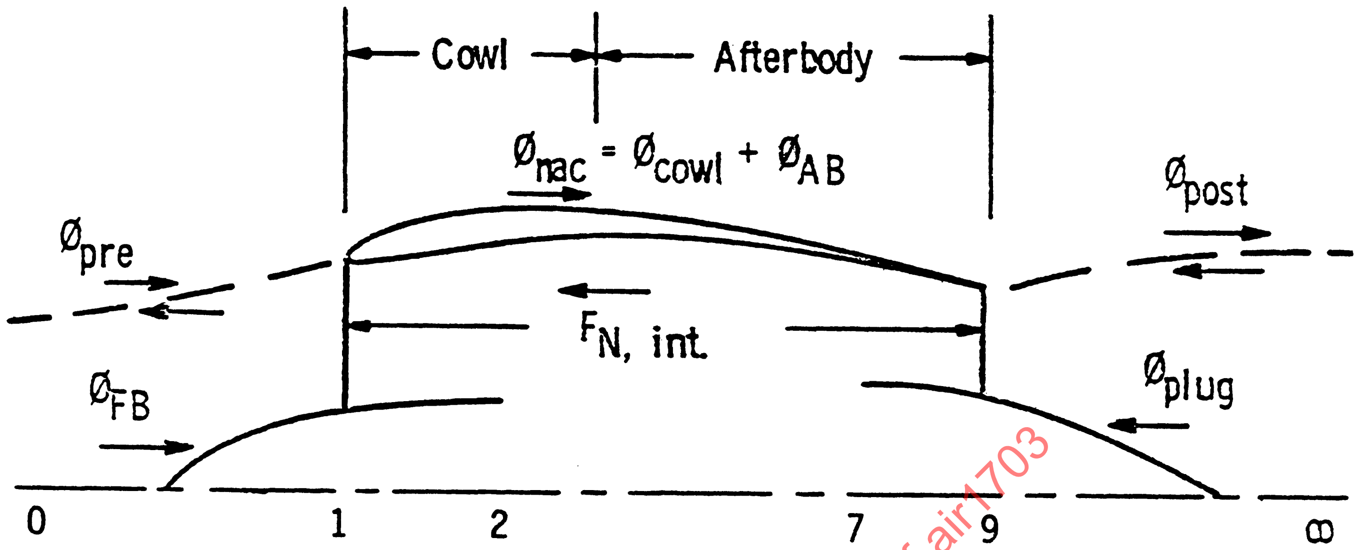


Figure 2.5 - Forces Acting on a Propulsion System

The general approach is to select an aircraft/propulsion-system reference operating condition. Thereby, a reference drag for the aircraft is established that includes reference inlet, exhaust and friction forces. Differences in overall aircraft drag from this reference are established, generally from scale-model testing as a function of the inlet and nozzle operating conditions; and these throttle-dependent forces (drags) are then accounted as changes in propulsive thrust in the thrust/drag accounting procedure. This accounting method permits aircraft performance to be related to engine net thrust and to a reference aircraft drag polar, (Appendix C describes aircraft drag polars).

The following paragraphs discuss the development of a thrust/drag accounting system for a fully integrated (buried) propulsion system to illustrate the overall approach. Considering an aircraft in level flight, the simplified force equation applied in the flight direction takes the form:

$$F_{EX} = F_{IPF} - D_{AFS} \qquad (2.12)$$

where, F_{EX} = Excess thrust (or total unbalanced force) in the flight direction.

F_{IPF} = Installed propulsive force that is obtained from net thrust with adjustments for deviations from full-scale reference conditions.

D_{AFS} = Airframe system drag at full-scale reference conditions as reflected in the aircraft drag polar. (See Appendix C)

2.2 Thrust/Drag Accounting (Cont'd.):

When excess thrust is zero or can be determined, aerodynamic drag relationships can be used to validate the method for determining in-flight thrust. The airframe system drag has several components that correct the aerodynamic drag to the full-scale reference conditions. Appendix C provides additional information.

The installed propulsive force is defined to be equal to the installed modified net thrust at the full-scale reference conditions and accounts for all propulsive forces acting on the aircraft. Additional forces are included in the airframe system drag. For excursions from the full-scale reference conditions, incremental forces must be considered. These throttle-dependent forces are included as adjustments to develop the installed propulsive force:

$$F_{IPF} = F_N^* - \Delta F_{INL} - \Delta F_{EXH} - \Delta F_{TRIM} \tag{2.13}$$

where, F_N^* = Modified net thrust, accounting for installation effects of inlet internal performance, nozzle internal performance, bleed air extractions, and shaft power extractions.

ΔF_{INL} = External force increment between full-scale reference and any given operating condition due to the inlet.

ΔF_{EXH} = External force increment between full-scale reference and any given operating condition due to the exhaust system.

ΔF_{TRIM} = External control-surface-trim force increment associated with operating the propulsion system at other than the chosen propulsion system reference conditions.

The establishment of the full-scale reference conditions requires the selection of several variables, including inlet mass flow ratio, inlet geometry, nozzle pressure ratio, nozzle geometry, secondary airflows, and aircraft trim setting. Since these variables influence the installation drag, a fixed set of reference conditions must be identified. The selection of full-scale reference conditions is influenced by practical considerations, an important factor being the aero-reference model, which is the wind-tunnel scale model used as the basis for determining the aircraft drag polar. The model may be typically of a 5- to 10-percent scale and designed to be as representative as possible of the full-scale aircraft. It may have flow-through nacelles, but it cannot altogether simulate the full-scale engine inlet mass flow ratio, inlet geometry, nozzle geometry, and nozzle pressure ratio. To minimize model complexity, secondary airflows are not normally simulated. Aero-reference conditions, therefore, are usually characterized by a nozzle pressure ratio of about one, a modified and fixed inlet or afterbody geometry, a non-representative inlet mass flow ratio, and no secondary airflows.

2.2 Thrust/Drag Accounting (Cont'd.):

Additional wind tunnel tests are conducted to determine the incremental drag forces attributed to the inlet and afterbody geometry, inlet mass flow ratio, and nozzle pressure ratio. Three models may be used, as illustrated in Figure 2.6. A convenient set of full-scale reference conditions are selected, and the aero-reference drags appropriately incremented to these conditions. The resulting full-scale drag polar is normally characterized by a fixed geometry for each flight speed, a typical operating inlet mass flow ratio and nozzle pressure ratio, and no allowance for secondary flow.

The general approach to thrust/drag accounting is summarized in Figure 2.7. The test models and analyses provide a reference aircraft drag and the throttle-dependent drag increments that are used with net thrust to determine installed propulsive force at any specific engine/aircraft operating condition.

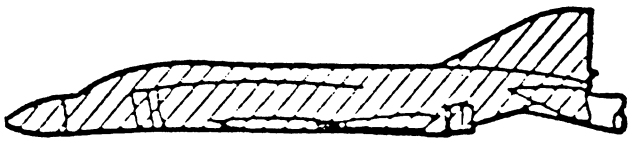
2.2.1 Inlet Force Increment: The inlet force increment can consist of spillage drag, ramp bleed drag, and such secondary flow terms as bypass and ventilation airflow drag. The throttle-dependent inlet spillage force increment (drag) is defined as the change in aircraft drag force resulting from the difference between operating and full-scale reference inlet mass flow ratios. Spillage drag, which is unrecovered inlet additive drag, varies with inlet mass flow, as illustrated in Figure 2.8.

The drag of the aero-reference-model inlet (Point 1 in Figure 2.8) is included in the aero-reference drag. The incremental drag between (1) and the full-scale reference condition (2) represents the scale model to full-scale reference correction to be included in the full-scale drag polar, ΔD_{INL}. Drag differences between engine operating conditions (3) and full-scale engine reference condition (2) are accounted for as propulsive-system-related inlet drag force, ΔF_{INL}.

For subsonic podded installations, spillage drag can be investigated by testing isolated flow-through models with internal geometry variations to alter inlet mass flow ratio. Nacelle cowl geometry should be simulated correctly to account for possible inlet/afterbody coupling effects. An isolated nacelle drag test may be used to determine differences in drag due to mass flow ratio.

An inlet spillage drag test is essential for complete thrust/drag accounting of a fully integrated propulsion system. The inlet and forward fuselage, including any items which might affect inlet flows, should be tested as a unit to assess the spillage drag. A flow-through model with internal duct modifications or an internal flow- metering device could be used to conduct this test. There is generally no requirement to duplicate the afterbody geometry, when inlet and afterbody forces are not closely coupled as with podded installations. For inlets on high speed aircraft, inlet tests must be conducted to determine drag sensitivity to geometry variation as well as inlet mass flow ratio.

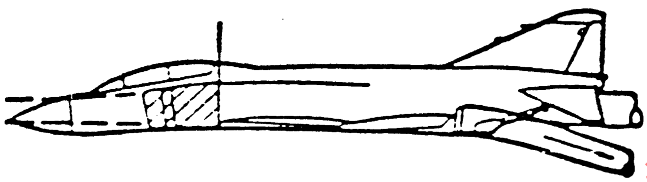
Aero Force and Moment



Characteristics

- Air Flow Normally Limited
- Air Flow Control Usually Limited
- Possible Support Interference and Geometric Distortion
- Complete Model

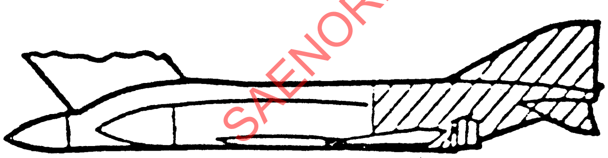
Inlet Drag



Characteristics

- Partial Model
- Wide Range Air Flow Control
- Sting Support

Jet Effects



Characteristics

- Partial Model
- High Pressure Air Supplied from External Source
- Hot or Cold Air May Be Used
- Faired-Over Inlets

Figure 2.6 - Example Test Models

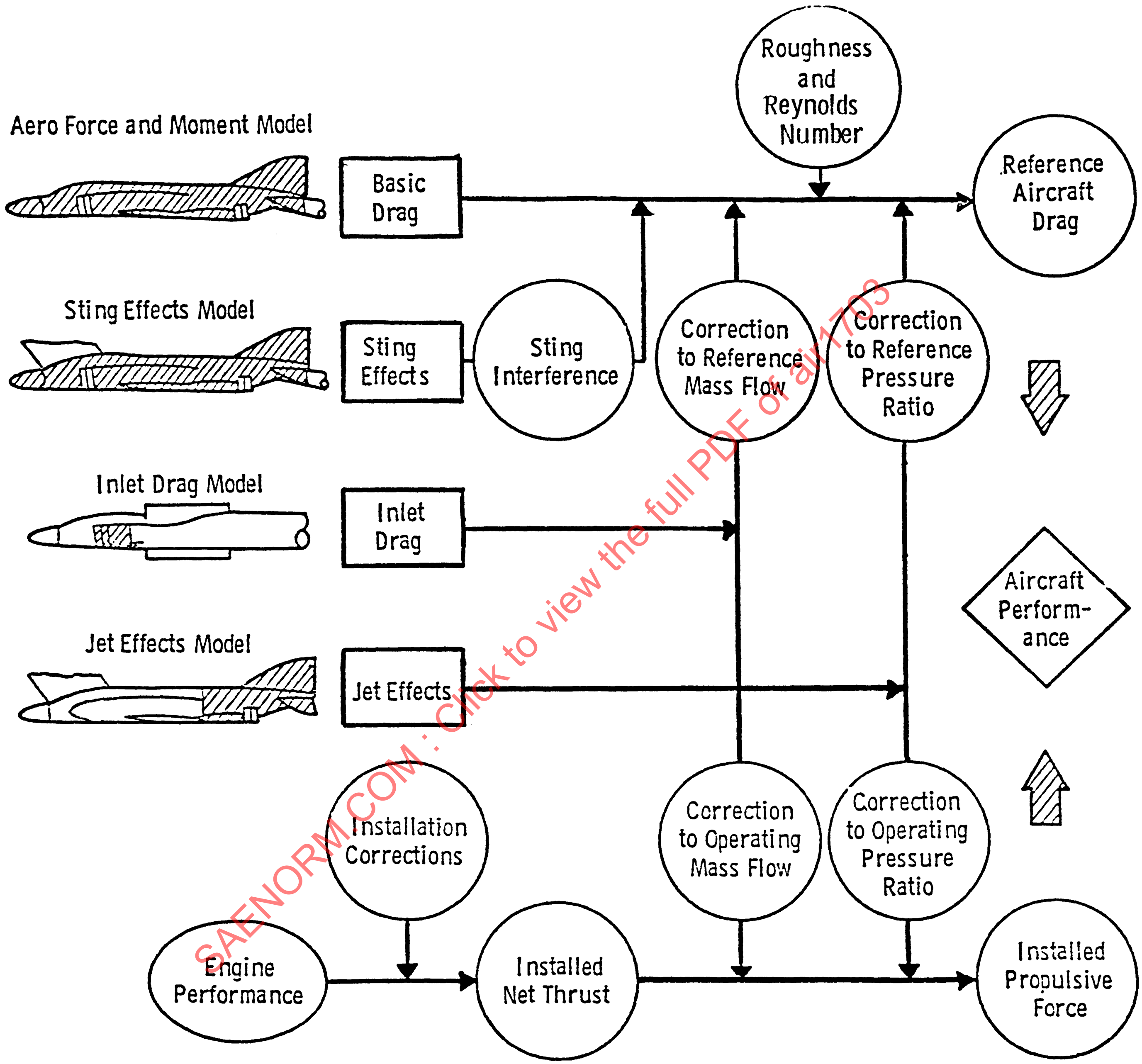


Figure 2.7 - Typical Thrust/Drag Accounting System

2.2.1 Inlet Force Increment (Cont'd.):

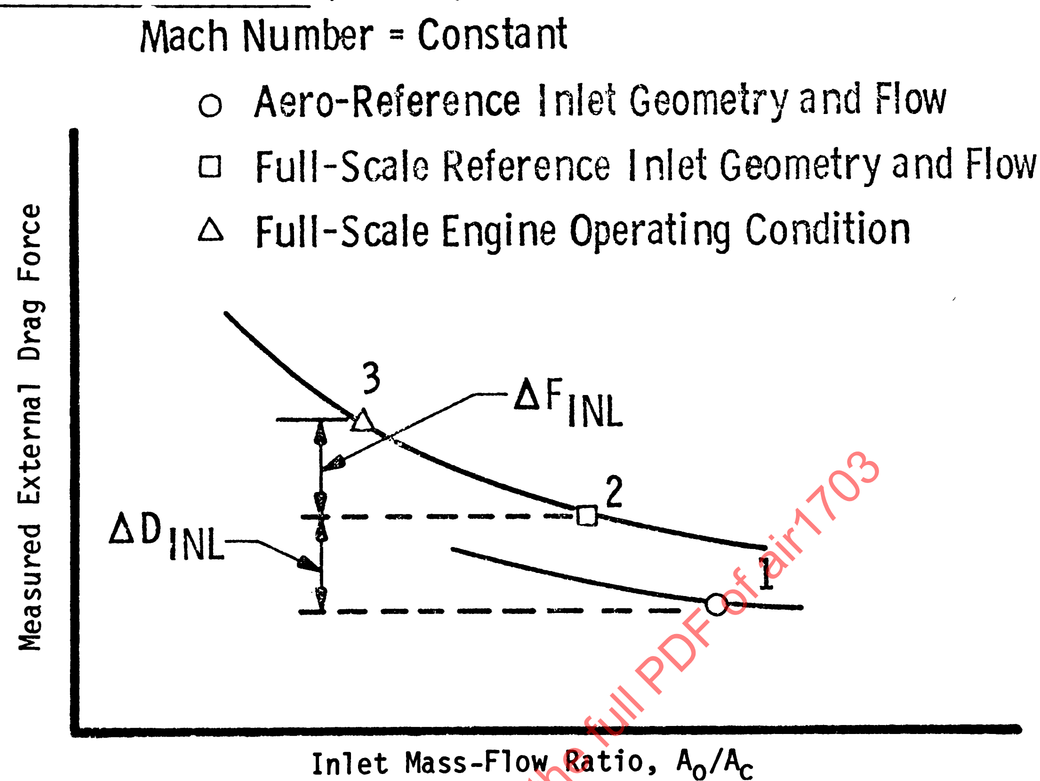


Figure 2.8 - Drag Force Variation With Inlet Mass Flow Ratio

Airflows, such as bypass airflow dump, constitute secondary airflows. Their ram drags (if the airflow is not part of the engine airflow) and thrusts (if the airflow is exhausted through an exit other than the main engine nozzle) must be included in the thrust/drag accounting system. Any impact these flows may have on the operating inlet mass flow and resulting spillage drag must be considered. Since secondary airflows are often throttle related, the net thrust or drag of each represents an additional adjustment to net thrust to obtain installed propulsive force. These additional throttle-dependent terms may be evaluated during inlet drag testing by separate model tests or by analytical calculations.

2.2.2 Exhaust-System Force Increment: The throttle-dependent exhaust-system or interference force increment (drag) is defined as the change in aircraft drag force resulting from the difference between operating and full-scale-reference nozzle pressure ratio and area. For integrated exhaust systems, nozzle pressure ratio influences the afterbody pressure distribution and drag. A change in nozzle area alters the afterbody closure and pressure distribution which likewise affects aircraft drag. For podded installations, nozzle pressure ratio may influence pressure distributions on the nacelle, pylon, wing, or fuselage. The resulting change in drag is called propulsion-system-related interference drag or jet-effects drag. Typical variations in aircraft drag force with nozzle area and pressure ratio are illustrated in Figure 2.9.

2.2.2 Exhaust-System Force Increment (Cont'd.):

Afterbody force tests are conducted to evaluate throttle-dependent exhaust-system drag forces. The drag of the aero-reference-model nozzle (Point 1 in Figure 2.9) is included in the aero-reference drag. The incremental drag between (1) and the full-scale reference condition (2) represents the scale model to full-scale reference correction to be included in the full-scale drag polar, ΔD_{EXH}. Drag force differences between engine operating condition (3) and full-scale engine reference condition (2) are accounted for as throttle-dependent exhaust system force, ΔF_{EXH}. Like the throttle-dependent inlet force increment, ΔF_{EXH} represents an adjustment to net thrust to obtain installed propulsive force.

Throttle-dependent exhaust-system forces can be investigated using a blown afterbody model or a blown aircraft model. The term "blown" refers to the use of a high-pressure external air source to vary nozzle pressure ratio. The airplane model inlets are faired over. The afterbody model simulates only the aft geometry and, therefore, has no air intake system. The variation in drag due to nozzle pressure ratio and nozzle area changes is measured.

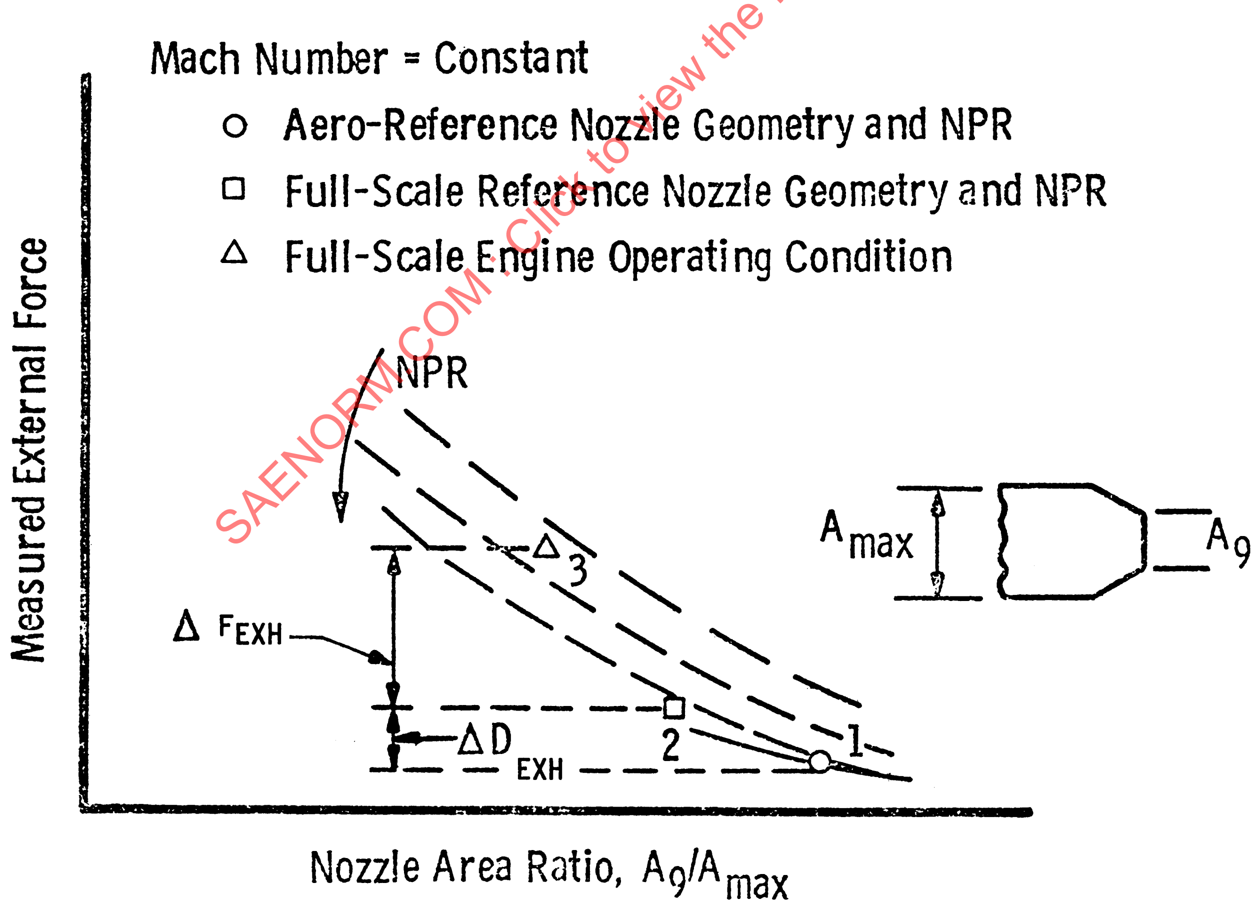


Figure 2.9 - Drag Force Variation With Nozzle Area and Nozzle Pressure Ratio

2.2.2 Exhaust-System Force Increment (Cont'd.):

Afterbody or interference drag force tests are conducted to establish a baseline drag level at full-scale reference nozzle area and pressure ratio which is included in the full-scale drag polar. Incremental changes from this baseline level define the throttle-dependent exhaust-system force adjustments to installed propulsive force.

2.2.3 Trim Force Increment: Aircraft weight, balance and maneuver conditions can require changes in the trim surface positions that affect aircraft drag. These drag differences are normally included in the aero-reference drag. Changes in the propulsion system throttle setting or geometry may also affect aircraft control-surface-trim positions. The drag increment for these items, from the aero-reference condition, is normally charged to the propulsion system and included in the installed propulsion force.

2.3 Thrust Method Options: Four methods for evaluating in-flight thrust are defined and briefly discussed to provide a foundation for detailed development and illustration. These encompass virtually all past and forecasted procedures but are not necessarily inclusive. They can be complementary and jointly utilized for specific aircraft applications.

2.3.1 Overall Performance Methods: An overall performance method consists of a set of dimensional or non-dimensional curves, a set of tables, or a computer program that is used to relate thrust to measureable engine operating parameters. It describes either the average performance of an engine type or the individual performance of a specific engine over a defined area of operating range. The background information for the method ranges from predictions prior to the first run of the engine through to comprehensive calibrations of one or more specific engines of a development or production type.

Overall performance is expressed in terms of the major engine control parameters for defined inlet and exhaust nozzle operating conditions. In the simplest case, prepared performance curves can be entered with one engine parameter measured during flight tests at specific flight operational conditions, and the net thrust obtained as an output. The parameter can be a simple one, such as shaft speed, which is readily available to good accuracy. The overall performance method may be such that it can be utilized with a number of alternative measured parameters. Several independent parameters may be necessary to define the engine operating conditions.

The key distinction of the overall performance method is that it requires no special instrumentation within the propulsion system flow path and involves minimum measurements in flight.

2.3.2 Gas-Path/Nozzle Methods: An engine consists of compressor, combustion and turbine components assembled to provide a gas generator. Internal flow-path measurements within these components may be utilized together with

2.3.2 Gas-Path/Nozzle Methods (Cont'd.):

mass-, momentum- and energy-continuity principles, to calculate flow conditions at various stations within the engine and to project overall engine performance. Inherently, these methods imply more engine instrumentation and in-flight measurements than for the overall performance method, with the expectation of improved accuracy in thrust determination.

The special instrumentation is calibrated and correlated with component and engine performance during ground tests where environmental and operating conditions can be accurately controlled. Where high accuracy is required, it may be necessary to utilize the same equipment during both the calibration and flight testing.

Temperature and pressure measurements at a convenient location or component flow/speed calibrations can be utilized to determine mass flow at that location. Then, engine inlet flow and ram drag may be derived from flow continuity, making allowance for fuel and secondary flows. It may be necessary to calculate flow properties at stations where instrumentation cannot be directly placed; an example being total temperature at the exit of an afterburner. It is good practice to measure sufficient engine parameters to describe the gas properties throughout the engine and check for consistency. Gas-path/nozzle methods normally involve using nozzle performance coefficients derived from model and full-scale experimental data.

Although nozzle methodology is considered a subtopic under the gas-path/nozzle method, it is appropriate to emphasize its importance. The gas generator provides the total pressure, total temperature and mass flow to the nozzle whose function is to convert energy to thrust. Since the nozzle produces losses during this conversion process, thrust cannot be determined solely from the characteristics of the gas generator. Thrust-nozzle methodology describes the procedure utilized to determine exit gross thrust from measurements or calculated conditions at the nozzle inlet. Paragraphs 2.4 and 2.5 discuss this subject.

2.3.3 Swinging Probe Method: A calibrated swinging probe or rake of probes may be used to traverse the nozzle exhaust to measure local total and static pressures, total temperatures and flow directions. An integrated exhaust mass flow and gross thrust can be calculated, provided that the traverse data are representative of the whole cross section. Ideally, special engine calibrations are not required. Fuel and secondary airflows are needed to determine engine inlet flow and net thrust.

2.3.4 Trunnion Thrust Method: This method involves measuring the force transmitted to the airframe via the engine mounting trunnions. This force represents the difference between stream forces at the engine inlet and exit stations taking into account engine nacelle pressure forces. To determine gross or net thrust, measurements of the appropriate interface stream forces are required. If the exhaust system is mounted separately, a similar procedure is required to measure the load on this component.

2.3.4 Trunnion Thrust Method (Cont'd.):

The presence of fire bulkheads, slip joints, ventilation flows, and airframe-to-engine connections can impose additional forces on the trunnions which must be taken into account.

2.4 Ideal Thrust and Normalized Groups: Since the gross thrust, F_G, applies at the nozzle exit plane (Station 9) while practical considerations dictate that nozzle conditions be measured at the nozzle entry plane (Station 7), a convention with several unique definitions has been developed to provide in-flight evaluations of thrust and mass flow. The general procedure is to relate real nozzle performance to that of an ideal nozzle through the use of empirically established coefficients.

2.4.1 Ideal Nozzle: For the ideal nozzle definitions, one-dimensional, isentropic flow is considered to exist within the nozzle downstream of the nozzle entry plane (Station 7). In the past, the additional constraint of constant specific heat ratio, γ, has often been assumed. However, with the widespread use of digital computers, the substitution of real gas properties does not present undue complications and is the preferred procedure. The use of constant specific heat ratio serves a useful purpose in illustrating relationships.

The ideal flow calculations may assume an ideal convergent nozzle where the expansion is limited to sonic (or choked) conditions at the nozzle exit plane. At the critical operating condition, the entry-total-pressure to exit-static-pressure ratio (P_{t7}/P_{s9}) is just sufficient to produce sonic flow at the exit plane via the one-dimensional isentropic flow assumption. At subcritical operating conditions, P_{s9} equals the ambient static pressure, P_{s0}, and the exit Mach number is less than 1.0. At supercritical operating conditions, P_{s9} exceeds P_{s0} due to the limitation of unity Mach number at the exit plane.

An ideal flexible convergent-divergent nozzle is defined as having a conceptual geometry which is infinitely variable or flexible such that the exit static pressure, P_{s9}, remains equal to P_{s0}. At subcritical and critical nozzle operating conditions, the ideal convergent and ideal convergent-divergent nozzles are conceptually identical and have the same ideal thrust performance. At supercritical conditions the ideal convergent-divergent thrust exceeds the ideal convergent thrust by virtue of complete isentropic expansion resulting in higher exit velocity.

Ideal-nozzle total temperature, total pressure and mass flow are assumed to remain constant and equal to the entry values (T_{t7}, P_{t7}, W_7) between the nozzle entry and exit stations. Static temperature, velocity and area (or static pressure) are calculated based on the one-dimensional isentropic-expansion assumption. The minimum area (throat) is designated Station 8 and will be coincident with the exit plane (Station 9) for the convergent nozzle.

2.4.2 Non-Dimensional Groups: Three non-dimensional ideal-nozzle groups are in common use to express the ideal flow and thrust. These are:

$$\left[\frac{W\sqrt{RT_t}}{AP_t}\right]_{id} = \text{Flow Function}$$

$$\left[\frac{F_G}{AP_{s0}}\right]_{id} = \text{Thrust Function}$$

$$\left[\frac{F_G}{W\sqrt{RT_t}}\right]_{id} = \text{Specific Thrust Function}$$

The nozzle area used in forming the thrust function, whether for ideal or actual nozzle performance, is usually the throat area A_8. This convention is adopted for this report. Free stream static pressure, P_{s0}, is used in the thrust function. Alternate derivations using the exit area, A_9, and the total pressure, P_t, may be substituted, where needed.

The ideal groups are related by the expression:

$$\left[\frac{F_G}{AP_{s0}}\right]_{id} = \left[\frac{F_G}{W\sqrt{RT_t}}\right]_{id} \cdot \left[\frac{W\sqrt{RT_t}}{AP_t}\right]_{id} \cdot \left(\frac{P_t}{P_{s0}}\right) \tag{2.14}$$

In ideal one-dimensional flow, the ideal thrust expressions are functions of nozzle pressure ratio, P_{t7}/P_{s0}, and specific heat ratio, γ, as summarized in Table 2.1.

When using constant specific heat ratio calculations, these ideal thrust expressions are helpful in reducing the calculation efforts and lead to the so called $W\sqrt{T}$ and AP thrust calculations.[2.5] The use of the ideal groups has diminished with the direct calculation of real gas properties using digital computers. The so called $W\sqrt{T}$ and PA options provide a choice in the selection of nozzle coefficient.

2.4.3 Referred Parameters: Engine performance is a function of inlet temperature and pressure. Normalizing data to a reference inlet condition removes much of this dependency and tends to collapse component and engine performance parameters to a single-valued function. Sea-level, standard temperature and pressure, T_{STD} and P_{STD}, are selected as the reference condition. The non-dimensional temperature and pressure correction factors are:

$\theta = T/T_{s0,\ SL,\ STD}$ $\qquad$ $\delta = P/P_{s0,\ SL,\ STD}$

Used with the inlet total temperature and pressure, they become:

$\theta_{t2} = T_{t2}/T_{s0,\ SL,\ STD}$ $\qquad$ $\delta_{t2} = P_{t2}/P_{s0,\ SL,\ STD}$

IDEAL PERFORMANCE GROUP	NOZZLE CONDITION	NOZZLE TYPE: CONVERGENT 1)	NOZZLE TYPE: FLEXIBLE CONVERGENT – DIVERGENT
$\frac{F_G}{A_8 P_{s0}}$	UNCHOKED 2)	$\frac{2\gamma}{\gamma-1}\left[\left(\frac{P_t}{P_{s0}}\right)^{\frac{\gamma-1}{\gamma}} -1\right]$	
	CHOKED 3)	$\left[2\left(\frac{2}{\gamma+1}\right)^{\frac{1}{\gamma-1}}\frac{P_t}{P_{s0}}\right]-1$	$\frac{2\gamma}{\sqrt{\gamma^2-1}}\left(\frac{2}{\gamma+1}\right)^{\frac{1}{\gamma-1}}\frac{P_t}{P_{s0}}\sqrt{1-\left(\frac{P_{s0}}{P_t}\right)^{\frac{\gamma-1}{\gamma}}}$
$\frac{W\sqrt{RT_t}}{A_8 P_t}$	UNCHOKED	$\left(\frac{P_{s0}}{P_t}\right)^{\frac{1}{\gamma}}\sqrt{\frac{2\gamma}{\gamma-1}\left[1-\left(\frac{P_{s0}}{P_t}\right)^{\frac{\gamma-1}{\gamma}}\right]}$	
	CHOKED	$\sqrt{\gamma\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}$	
$\frac{F_G}{W\sqrt{RT_t}}$	UNCHOKED	$\sqrt{\frac{2\gamma}{\gamma-1}\left[1-\left(\frac{P_{s0}}{P_t}\right)^{\frac{\gamma-1}{\gamma}}\right]}$	
	CHOKED	$\sqrt{\frac{2(\gamma+1)}{\gamma}} - \frac{P_{s0}}{P_t}\sqrt{\frac{1}{\gamma}\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma+1}{\gamma-1}}}$	SAME AS UNCHOKED.

NOTES: 1) $A_8 = A_9$

2) $\frac{P_t}{P_{s0}} < \left(\frac{\gamma+1}{2}\right)^{\frac{\gamma}{\gamma-1}}$

3) $\frac{P_t}{P_{s0}} \geq \left(\frac{\gamma+1}{2}\right)^{\frac{\gamma}{\gamma-1}}$

Table 2.1 - Ideal Nozzle Performance Groups

2.4.3 Referred Parameters (Cont'd.):

Examples of referred (or corrected) engine-performance-parameters are:

net thrust, F_N/δ_{t2}, and inlet airflow, $W_2\sqrt{\theta_{t2}}/\delta_{t2}$.

2.5 Nozzle Coefficients: It is essential that a consistent set of coefficients be defined to relate actual to ideal nozzle performance in a manner that is practical for in-flight thrust determination.

2.5.1 Flow Coefficient, C_D: It is convenient to define the flow (or discharge) coefficient such that it applies to both the convergent (con) and the flexible convergent-divergent (flex-con-di) nozzle. The coefficient is based on the throat area (Station 8) in the case of the convergent-divergent nozzle. The notation is such that P_t and T_t represent suitable mean values of nozzle entry total pressure and temperature, defined at a reference plane located upstream of the nozzle, and that W_{act} denotes the actual mass flow passing through the nozzle of actual (geometric) area, A_{act}.

The flow (or discharge) coefficient, C_D, is defined as the ratio of actual to ideal mass flow for a given nozzle geometry and pressure ratio. It may equally be defined as the ratio of effective to actual (or geometric) throat area required to pass the actual mass flow. Hence,

$$C_D = \frac{W_{act}}{W_{id}} = \frac{A_{8,\ eff}}{A_{8,\ act}} \qquad (2.15)$$

Using the non-dimensional groups, the flow coefficient may also be defined as:

$$C_D = \left(\frac{W\sqrt{RT_t}}{A_8 P_t}\right)_{act} \Big/ \left[\frac{W\sqrt{RT_t}}{A_8 P_t}\right]_{id} \qquad (2.16)$$

And:

$$A_{8,\ eff} = C_D \cdot A_{8,\ act}$$

$$= \left(\frac{W\sqrt{RT_t}}{P_t}\right)_{act} \Big/ \left[\frac{W\sqrt{RT_t}}{A_8 P_t}\right]_{id} \qquad (2.17)$$

2.5.2 Specific Thrust Coefficients, C_V and C_X: The specific thrust coefficient, C_V, is defined as the ratio of actual specific thrust to ideal specific thrust obtainable from an ideal flexible convergent-divergent nozzle at a given pressure ratio. This coefficient can be expressed as the ratio of an effective discharge velocity to the ideal velocity obtainable with an ideal flexible convergent-divergent nozzle. Thus,

$$C_V = \frac{(F_G/W)_{act}}{(F_G/W)_{id,\ flex\text{-}con\text{-}di}} = \frac{V_{9,\ eff}}{V_{9,\ id,\ flex\text{-}con\text{-}di}} \qquad (2.18)$$

2.5.2 Specific Thrust Coefficients, C_V and C_X (Cont'd.):

where,

$$V_{9,\ eff} = (F_G/W)_{act} \tag{2.19}$$

The fact that F_G/W may be interpreted as an effective velocity has led some to call C_V a velocity coefficient rather than a specific thrust coefficient.

Using the non-dimensional groups, the specific thrust coefficient, C_V, may then be written:

$$C_V = \left(\frac{F_G}{W\sqrt{RT_t}}\right)_{act} \Bigg/ \left[\frac{F_G}{W\sqrt{RT_t}}\right]_{id,\ flex\text{-}con\text{-}di} \tag{2.20}$$

The resulting general thrust Equation is:

$$F_{G,\ act} = C_V \cdot W_{act} \cdot \sqrt{RT_t} \cdot \left[\frac{F_G}{W\sqrt{RT_t}}\right]_{id,\ flex\text{-}con\text{-}di} \tag{2.21}$$

$$= C_V \cdot W_{act} \cdot V_{9,\ id,\ flex\text{-}con\text{-}di} \tag{2.22}$$

Another thrust coefficient, C_X, is defined similar to C_V except that the ideal nozzle parameters are based upon an ideal convergent nozzle. Thus,

$$C_X = \frac{(F_G/W)_{act}}{(F_G/W)_{id,\ con}} = \left(\frac{F_G}{W\sqrt{RT_t}}\right)_{act} \Bigg/ \left[\frac{F_G}{W\sqrt{RT_t}}\right]_{id,\ con} \tag{2.23}$$

2.5.3 Gross Thrust Coefficient, C_G: The gross thrust coefficient, C_G, is defined as the ratio of actual thrust to ideal thrust obtained from an ideal nozzle at a given pressure ratio. It may also be expressed as the product of the flow and specific thrust coefficients for a given nozzle geometry and pressure ratio. Using the ideal convergent nozzle as the basis for coefficient derivation, the gross thrust coefficient is:

$$C_{G,\ con} = C_D \cdot C_X \tag{2.24}$$

$$= \left(\frac{F_G}{A_8\ P_{s0}}\right)_{act} \Bigg/ \left[\frac{F_G}{A_8\ P_{s0}}\right]_{id,\ con} \tag{2.25}$$

2.5.3 Gross Thrust Coefficient, C_G(Cont'd.):

The notation $C_{G,con}$ refers to a gross thrust coefficient based on ideal convergent nozzle performance characteristics regardless of type of actual nozzle being represented.

Using the ideal flexible-convergent-divergent nozzle as the basis for coefficient derivation, the gross thrust coefficient is:

$$C_{G,\text{flex-con-di}} = C_D \cdot C_V \tag{2.26}$$

$$= \left(\frac{F_G}{A_8\ P_{s0}}\right)_{act} \Bigg/ \left[\frac{F_G}{A_8\ P_{s0}}\right]_{\text{id, flex-con-di}} \tag{2.27}$$

Both of the thrust coefficients presented above may be used to represent the performance of actual nozzles which are either convergent or convergent-divergent. However, for convergent-divergent nozzles, some care is necessary in the area of the PA term. Difficulty will be avoided if the flow and thrust coefficients are both based on A_8.

The resulting general thrust Equation, using the coefficient based upon the ideal flexible-convergent-divergent nozzle, is:

$$F_{G,\text{ act}} = C_{G,\text{ flex-con-di}} \cdot A_{8,\text{ act}} \cdot P_{s0} \cdot \left[\frac{F_G}{A_8\ P_{s0}}\right]_{\text{id, flex-con-di}} \tag{2.28}$$

$$= C_{G,\text{ flex-con-di}} \cdot W_{9,\text{ id}} \cdot V_{9,\text{ id, flex-con-di}} \tag{2.29}$$

The preceding equations define the most commonly used nozzle-performance coefficients. However, other forms may be derived to satisfy specific requirements.

3. PROPULSION SYSTEM INSTALLATIONS:

This Section extends the previously defined concepts and definitions to the force accounting of various types of nacelle and integrated propulsion systems. The net thrust and installed propulsive force equations are discussed to emphasize the main performance bookkeeping requirements.

Aircraft engine installations differ widely. Powerplant configurations include high-bypass-ratio commercial turbofans, which have fixed-geometry long- and short-duct nacelles, and highly integrated turbojet and low-bypass-ratio turbofans, which have variable-geometry inlets and exhaust nozzles. Each propulsion system will have its own bookkeeping requirements and procedures. Despite this variety, force accounting procedures have much in common. For example, most will employ a well defined set of reference conditions, utilize wind-off engine calibrations, seek to achieve as unique a drag polar as possible, etc. The form of the basic propulsive force equation will be the same, although the terms may differ in important detail.

For most systems, the installed net propulsive force, F_{IPF}, may be written as previously shown in Equation 2.13. This definition includes the modified net thrust, F_N*, and other terms or "deltas" that represent throttle-dependent forces (Section 2). The individual terms differ in form and magnitude among the various propulsion systems and their operating conditions. At aircraft/propulsion system reference conditions, the "deltas" are zero, and F_{IPF} equals F_N* which in turn equals the aircraft reference drag, D_{AFS}, in steady non-accelerating flight.

The equation is valid in the general case and is applicable to any conventional turbojet/turbofan powered aircraft whether the engine installation is highly integrated or a nacelle/pod. Since performance accounting systems are intended to cover a wide range of operating conditions including highly integrated military engine installations, the evaluation of throttle-dependent force increments will require considerable wind-on and wind-off model testing. In some cases, the ΔF_{INL}, ΔF_{EXH}, and ΔF_{TRIM} terms may be ignored, and less complex propulsion system performance accounting methods may then be adequate. Whether they may be ignored or not depends on the range of aircraft and engine operating conditions, the aerodynamic design of the aircraft and engine installation, the mission, and the objectives of the flight test program.

The purpose of including the force "deltas" in the F_{IPF} term is to produce an aircraft drag polar that is power independent. Any throttle-related terms either excluded, or improperly accounted, will result in the particular force element being transferred from F_{IPF} to the aircraft drag polar; that is, the force element will have been moved to the aircraft (drag) side of the performance bookkeeping account. This section illustrates for selected types of powerplants the relative significance of the several possible throttle-dependent forces so that the relationship of F_N* to F_{IPF} can be clarified.

3. PROPULSION SYSTEM INSTALLATIONS (Cont'd.):

The general examples in Paragraph 3.1 encompass the podded nacelle installation of a turbojet or mixed-flow and compound-flow turbofan, the long-duct turbofan, and the intermediate or short-duct turbofan. Examples of integrated installations are provided in Paragraph 3.2 together with special inlet and nozzle/afterbody force accounting considerations.

3.1 Nacelle Installations: In the following examples, the nacelles are regarded substantially as isolated from the airframe fuselage, wing or other major structure. This simplification implies that the flow field of the airframe has little influence on the propulsion system and that the flow through and about the powerplant has little influence on the aircraft drag. Essentially, nacelles and airframe are considered to be decoupled. This idealized assumption permits the retention of the simplified F_{IPF} equation without the introduction of additional terms that may be required to describe nacelle/airframe interference forces.

The gross thrust and ram drag are evaluated by any appropriate thrust and airflow method option described in Section 4. One of the referred thrust and flow groups, and nozzle coefficients defined in Paragraph 2.2 would be used, if a gas-path/nozzle option were adopted. The application of the net thrust equation is straight forward.

At full-scale reference conditions, the incremental force terms are zero by definition so that, for steady non-accelerating flight, Equations 2.12 and 2.13 yield:

$$F_N{}^* = D_{AFS} \text{ (at full-scale reference conditions)} \qquad (3.1)$$

where, D_{AFS} is the full-scale airframe drag per engine including the nacelle drag at flight conditions.

At other operating conditions Equation 2.13 will in general apply, and F_{IPF} equals the full-scale airframe drag per engine excluding the nacelle force increments at those flight conditions.

For the subsonic isolated-nacelle installation, the incremental nacelle forces ΔF_{INL}, ΔF_{EXH}, and ΔF_{TRIM} and any airframe force changes may be small and may not need to be identified separately. As stated earlier, any non-zero terms could then be included in the aircraft drag polar to acceptable accuracy. Then, $F_N{}^*$ equals (approximately) the full-scale airframe drag per engine.

3.1 Nacelle Installations (Cont'd.):

Generally, the aerodynamic force model includes the inlet flowing at or close to nominal (cruise) mass flow ratio (MFR). With careful attention to inlet lip and nacelle forebody design, sufficient pre-entry cancellation force can be generated to a low inlet flow. This is particularly valid for the majority of aircraft performance test conditions. Only at far off-design conditions may the drag polar not generalize to an acceptable degree so that explicit accounting for the inlet force increment, ΔF_{INL}, would be needed. Even so, practicalities may dictate that special model and flight tests are not warranted to define the incremental inlet drag for these limited number of conditions. Each program will have its own requirements.

Driving factors in producing an exhaust-nozzle afterbody force increment, ΔF_{EXH}, from the reference condition are nozzle area and pressure ratio. With the assumption of a fixed nozzle, the former is eliminated; and for subsonic aircraft, the excursion of nozzle pressure ratio from the full-scale reference pressure ratio may be small at the most important flight conditions. Post-exit thrust effects would be insignificant. Any effects of nozzle flow on adjacent airframe structure and power-related effects may not be significant, if the nacelles are "far-coupled". If so, the explicit accounting of ΔF_{EXH} may be avoided.

The change in control-surface trim forces, ΔF_{TRIM}, which are associated with operating the propulsion system at other than the reference operating conditions, are generally quite small and may be neglected for the subsonic non-afterburning aircraft. If accounting for its influence is required, it will usually be derived from force and moment increments obtained from aircraft model tests.

The net thrust, F_N^*, is often termed the "wind-off" thrust, as it is derived using calibrations of engines in ground test facilities. It is common practice to utilize calibrations in the form of non-dimensional groups or nozzle coefficients. It is these groups or coefficients that are assumed invariant from wind-off (ground facility) to wind-on (aircraft) conditions. The input parameters, such as pressures for the in-flight thrust calculation procedure, are measured in flight. For the simple turbojet example with choked nozzle, non-dimensional performance is independent of wind-off/-on effects, e.g., rematching effects. This would not be so at unchoked nozzle conditions, and corrections to nozzle flow characteristics would be required.

3.1.1 Turbojet Nacelles: The simplest installation example is a non-afterburning engine installed in an isolated nacelle having neither variable inlet nor exit geometry and operating with a choked nozzle in a subsonic environment. In Equation 2.13, the terms ΔF_{INL}, ΔF_{EXH} and ΔF_{TRIM} are generally assumed to be zero for this special case. This is shown schematically in Figure 3.1, and also applies equally to mixed-flow and compound-flow turbofans. The engine attachment pylon does not penetrate into the flow stream and, therefore, produces no additional force in the thrust equation.

3.1.2 Long-Duct Turbofan Nacelles: The turbofan engine develops two separate airflow streams, viz., that produced in the fan bypass duct and that produced in the engine core. In some turbofan designs, the two streams are expanded through separate coannular, coplanar nozzles, i.e., the exit planes are essentially at the same axial location, requiring a long-duct nacelle, as illustrated in Figure 3.2(a). Compound-flow and mixed-flow turbofans have one exhaust nozzle and are also installed in long-duct nacelles, as illustrated schematically in Figure 3.2(b) and Figure 3.3, respectively.

For these long-duct nacelle installations, the overall force accounting rationale, assumptions and bookkeeping procedures remain essentially unchanged from that of the isolated turbojet nacelle. In particular, the propulsion system force equations are the same except that, in the case of the dual-exhaust and compound turbofan, the exit gross thrust is normally calculated separately for the two flow streams (see Equation 2.11).

In the event that a plug nozzle forms part of the long-duct powerplant configuration, it is customary to include the plug force term in with the gross thrust (see also Paragraph 3.1.3).

3.1.3 Intermediate or Short-Duct Turbofan Nacelles: An intermediate or short-duct nacelle, which is accomplished by reducing the length of the outer fan duct, is typical of existing high-bypass-ratio turbofans. Figure 3.4 schematically presents this arrangement and the forces acting on the system. The external nacelle force chargeable to the engine thrust has been separated into the external core-afterbody force and the external primary exhaust plug force. The external core afterbody force includes the bypass scrubbing force on that portion of the pylon that protrudes through the fan-exit streamtube.

The separate assessment of all the forces shown in Figure 3.4 is a difficult task. Experience has shown that it is more convenient and accurate to define modified gross thrust terms which incorporate the external core afterbody and plug forces. Equation 2.7 applies.

The methodology for determining F_N* is selected from those discussed in Section 4. F_N* is the "wind-off" modified net thrust.

Quantifying the incremental force terms, ΔF_{INL} and ΔF_{EXH}, becomes more complex when the fan cowl and aircraft wing are close coupled. Wind-off to wind-on "external" effects are usually bookkept in the incremental terms. In the event that fan and core nozzles are unchoked, rematching of the engine components can occur and produce further flow-field interaction and propulsion-system-force redistribution.

3.2 Integrated Installations: An integrated installation is defined as one in which the propulsion system is buried within or closely coupled aero-dynamically to adjacent aircraft structure (wing/fuselage/nacelle interference). In such installations, the engine inlet airflow and exhaust jet will influence the flow around the airframe and so influence the

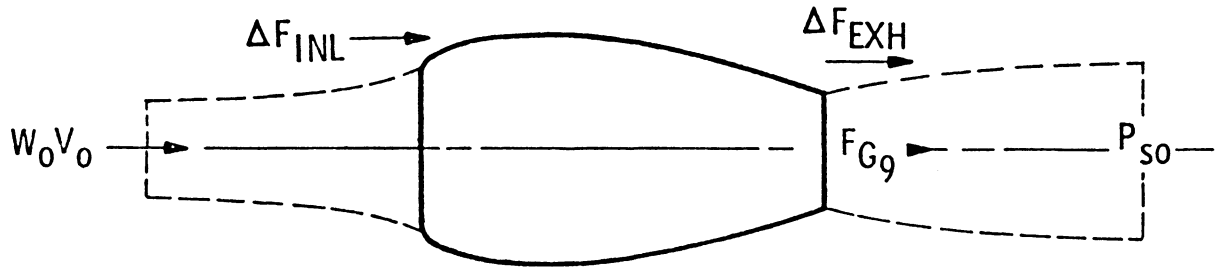


Figure 3.1 - Isolated Nacelle Installation

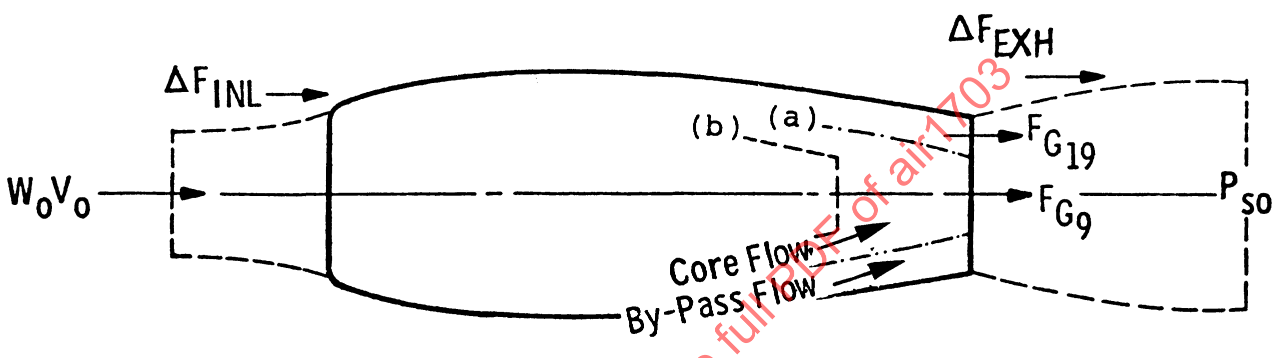


Figure 3.2 - Long-Duct Turbofans

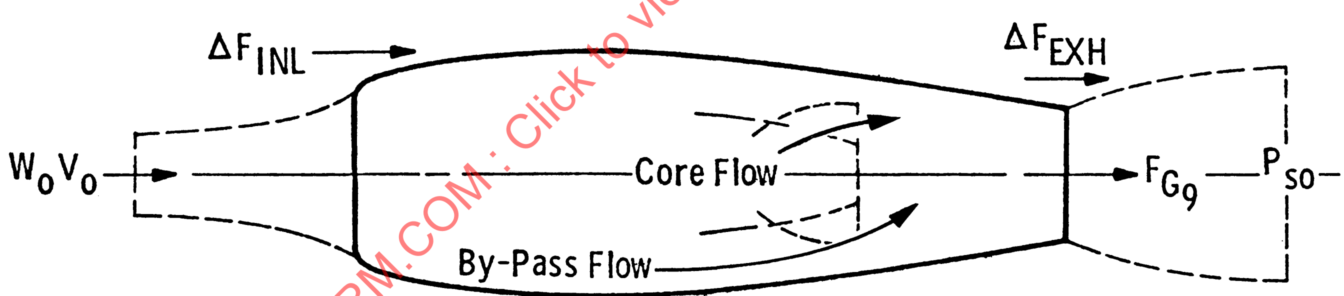


Figure 3.3 - Mixed-Flow Turbofan

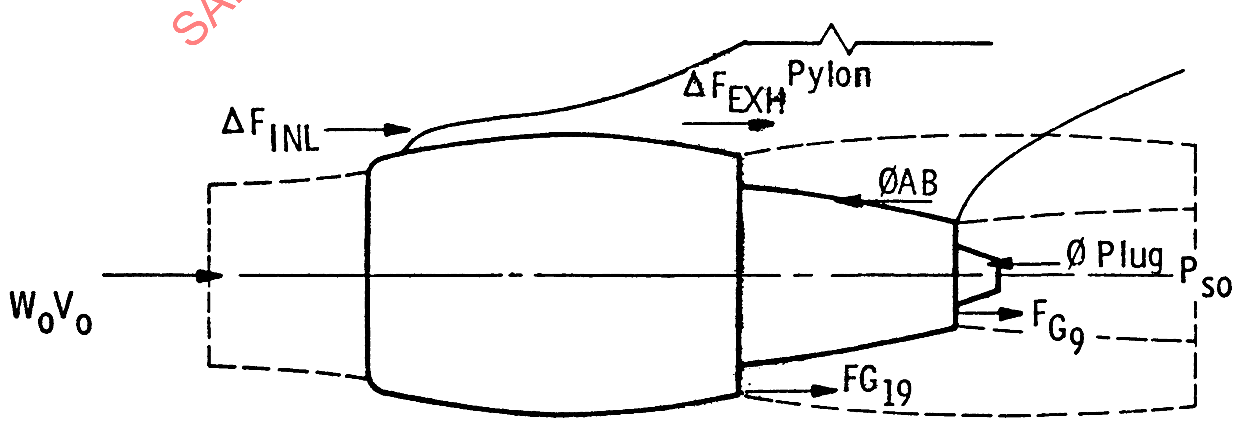


Figure 3.4 - Intermediate or Short Duct Turbofan

3.2 Integrated Installations (Cont'd.):

aircraft drag. A typical installation is illustrated in Figure 3.5. Equation 2.13 applies, and the throttle-dependent incremental forces cannot be ignored. In order to obtain a general aircraft polar necessary for aircraft performance evaluation, these throttle-dependent forces and the associated bookkeeping procedure described in Paragraph 2.2 should be adopted. A unique polar, which is independent of power, may be difficult to achieve, and more than one reference condition may be chosen, e.g., subsonic and supersonic. In calculation of ΔF_{INL} and ΔF_{EXH}, care must be taken to ensure inclusion of the effects of secondary airflows and the effects of propulsion system variable geometry.

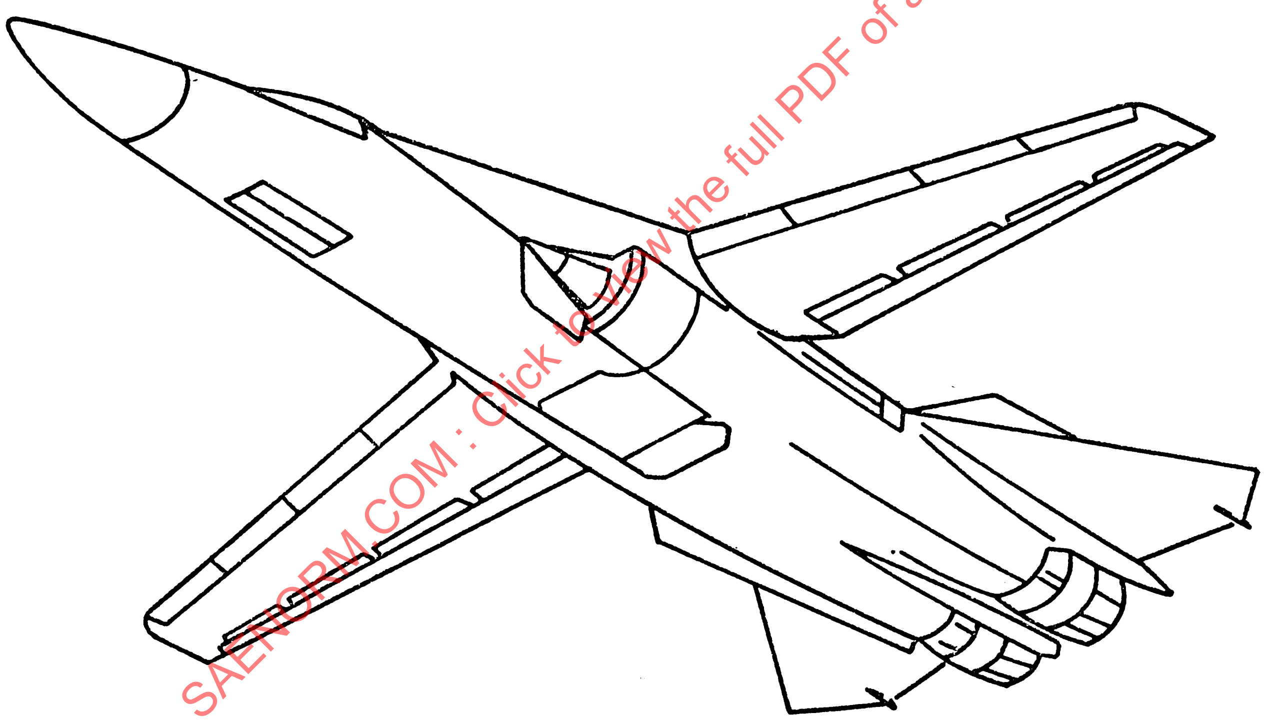


Figure 3.5 - Typical Integrated Propulsion System

3.2.1 Secondary Flows: Buried installations may feature one or more secondary flows within or external to the engine flow stream that are exhausted either separately or in the vicinity of the engine propulsion nozzle. Examples are inlet diverter and bleed systems, and ejector nozzles. Their flow characteristics may depend upon engine power setting. A typical example, shown in Figures 3.6 and 3.7, illustrates the non-afterburning and afterburning cases, respectively. Inclusion of appropriate force terms in F_{IPF} is historically less well established than the terms discussed previously, but they must be addressed and be accounted for in the thrust/drag bookkeeping equation.

Some inlets and ejector nozzles receive secondary air from the airframe local flow field that, in a closely integrated system, can include significant quantities of airframe boundary-layer air. Considerable care must be exercised to properly account for the separate forces without "double accounting," as separation of individual force contributions may either not be possible or, at the least, most difficult.

3.2.2 Inlet Considerations: The inlet spillage drag cannot be directly measured in flight; either it is determined from inlet force model tests or else determined purely analytically. Inlet systems, which are closely integrated with the aircraft, require inlet model tests to develop ΔF_{INL}. These model tests are described in Paragraph 5.4. This process is complicated often by the addition of inlet variable geometry, as illustrated by Figure 3.8. In model tests, the overall force is measured and the momentum forces are subtracted to obtain the inlet external drag force. The concept and definition of ΔF_{INL} was presented in Paragraph 2.2.1, and the resultant is the incremental spillage drag. Proper modeling techniques are necessary to obtain representative values. Model scale effects become particularly significant in closely coupled installations because boundary-layer flow tends not to follow scaling rules. Determination of boundary-layer behavioral differences between model and full scale is extremely difficult, but fortunately from a performance point of view, these influences tend to affect inlet/engine compatibility more than spillage drag. Spillage drag is influenced by airflow and angles of attack and sideslip, thus requiring these measurements in both model and flight tests.

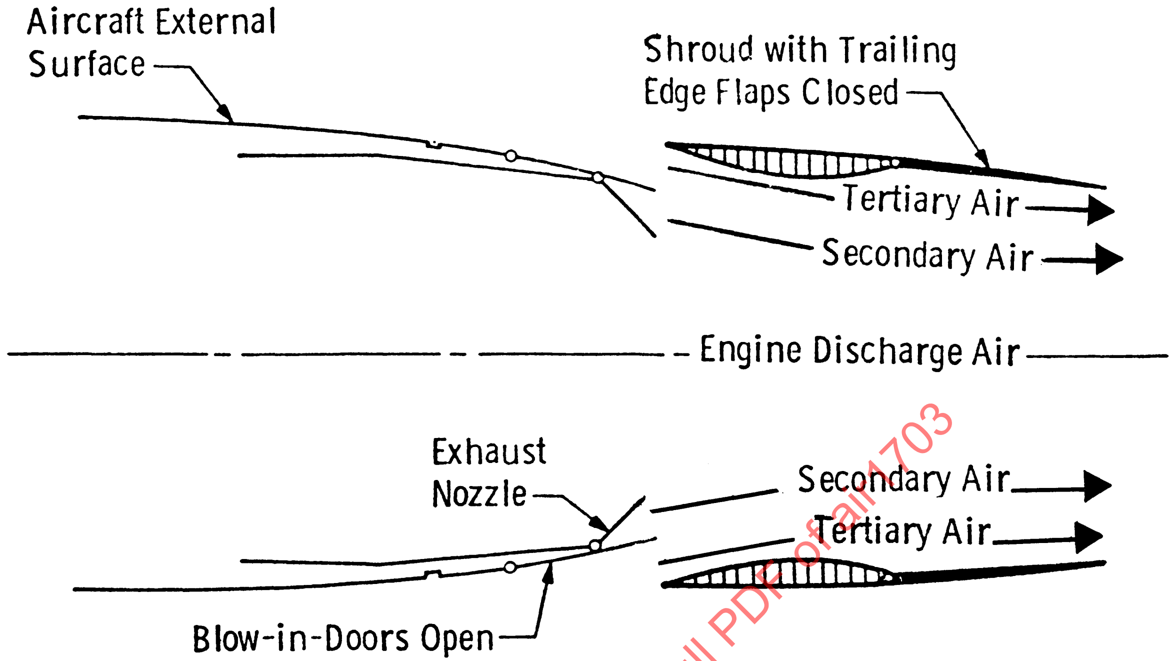


Figure 3.6 - Typical Ejector Nozzle, Non-Afterburning

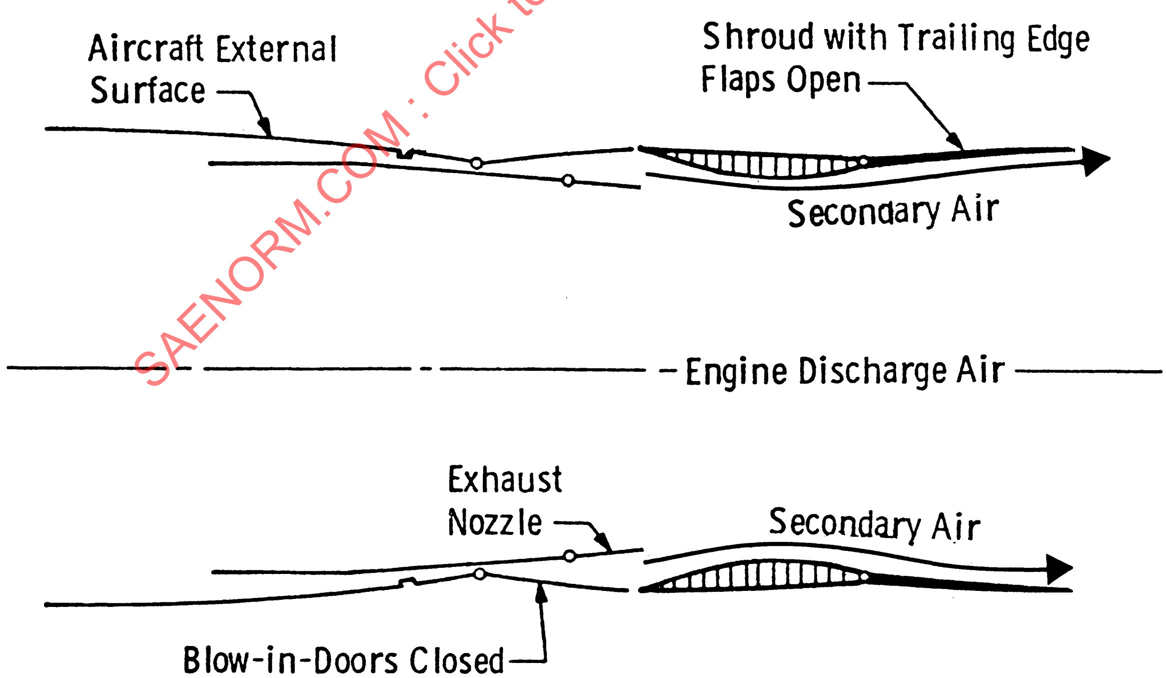


Figure 3.7 - Typical Ejector Nozzle, Afterburning

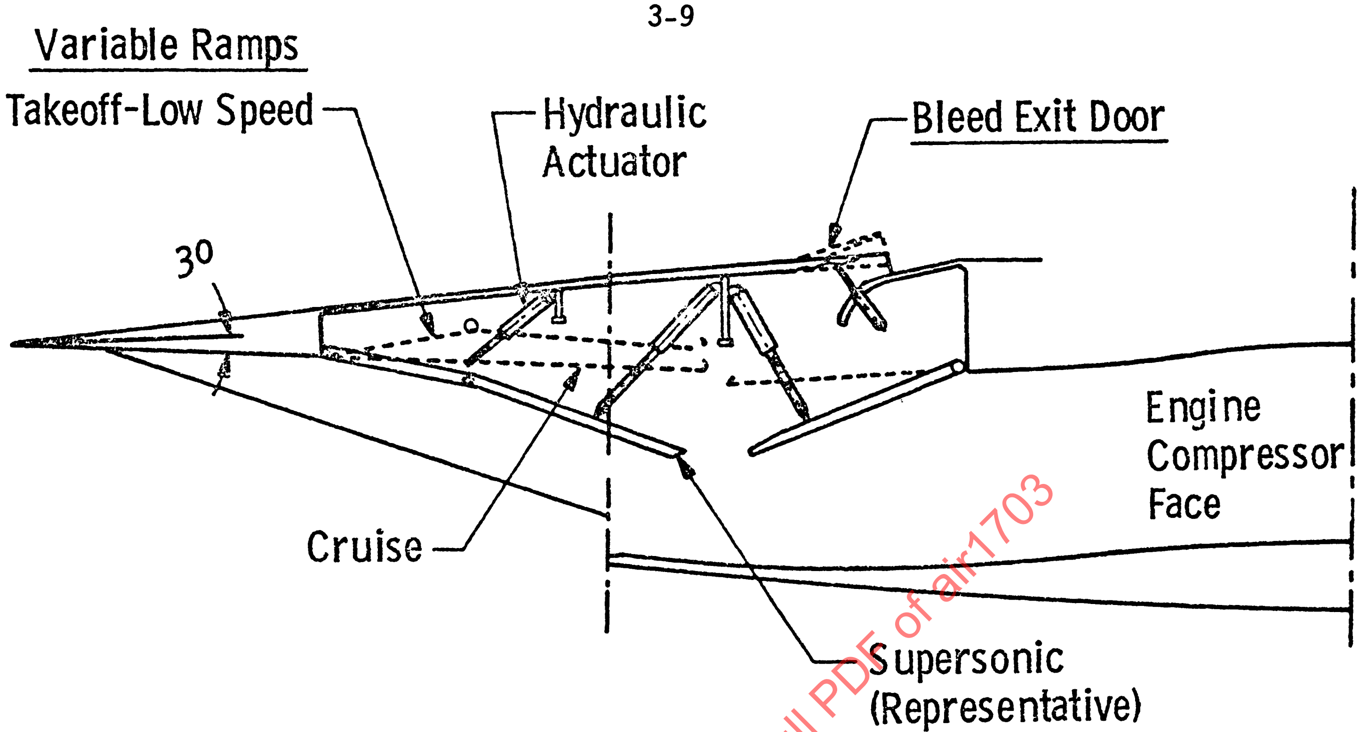


Figure 3.8 - Typical 2-D Inlet With Ramps

3.2.3 Nozzle/Afterbody Considerations: The nozzle/afterbody incremental force, ΔF_{EXH}, is determined mostly through scale-model tests. Pressure measurements have been made on full-scale aircraft in flight; these have generally been used in the thrust validation process. They may reveal problem areas such as severe flow separation and consequent sources of drag. Results from these pressure-measurement tests can lead to re-shaping the aircraft surfaces.

Very careful attention must be paid to the scaling of all the geometrical surfaces, particularly in complicated designs such as illustrated in Figures 3.6 and 3.7. Designing the model flow passages requires careful accounting for thermal expansion. Consideration of scale and boundry-layer effects on the force field is advisable. Any differences between scale-model and full-scale reference areas or pressure ratios must be accounted.

Air is often used to simulate exhaust nozzle flow. If this is done, adjustment for the effects of the different gas properties (γ, R, T, P) must be made analytically. Use of hot gases minimizes such corrections; however, when mixtures, such as combusted hydrogen peroxide are used, the gas properties will still differ from those of the actual engine exhaust gas.

The desired model test result is an incremental force, ΔF_{EXH}, as defined in Paragraph 2.2.2. In reality, several models have to be tested. Paragraph 5.3 describes the tests.

3.2.4 Ejector Installations: The ejector nozzle requires special consideration in arriving at a definition of thrust/drag terms, particularly if individual components are required to be identified for analysis purposes. Careful accounting of forces generated by the secondary and tertiary flows at the nozzle exit is required to determine modified gross thrust. Similarly, the ram drag must include the introduction of the secondary flow.

Secondary flow will influence both the gross thrust coefficient and the suppression effects of the external flow field. The gross thrust determination of an integrated ejector nozzle requires adjustment of uninstalled wind-off thrust coefficients, to obtain the installed thrust coefficient. In the installed configuration, the external flow field is generally asymmetric, further complicating the analysis. The prime concern is that all forces are accounted - but only once.

4. IN-FLIGHT THRUST METHODS:

The fundamental problem of in-flight thrust measurement is that the methodology employed is indirect; that is, measurements are made of indicators of thrust, rather than thrust itself. This section discusses separately each of four general method options used for in-flight thrust determination. Subsequently, method logic charts are presented to illustrate a few examples of the use of these methods.

4.1 Overall Performance Options: The key distinction of an overall performance method for the determination of in-flight net thrust or modified net thrust, which distinguishes it from other method options, is that it requires no special instrumentation within the engine gas flow path; so in-flight thrust is determined from minimum measurements encompassing aircraft flight and engine power-setting conditions. Overall performance methods are used extensively in their own right or in conjunction with other methods. In-service thrust setting systems may be essentially overall performance methods.

For many installations, a properly constructed overall performance method can enable in-flight thrust to be determined to the required accuracy. However, overall performance methods can suffer from certain intrinsic difficulties, some of which can be overcome by covering a wide range of flight conditions during engine calibration and others which cannot readily be overcome. Difficulties in the former category typically involve accounting for environmental effects (inlet distortion or nozzle pressure ratio effects) on engine performance, the increasing complexity of calibration data for engines with multiple degrees of freedom, etc. Typical difficulties in the latter category include engine deterioration between calibration and flight test, external flow (wind-on/-off) effects, sensitivity to engine build standards, etc. It is for such reasons that gas-path/nozzle methods utilizing gas path instrumentation to determine propulsion system component performance came to be employed (Paragraph 4.2). Even when using a gas-path/nozzle method, an overall performance method may be utilized as a secondary or supplementary method.

The following paragraphs discuss overall performance methods under two general topics. First, the general methodology is presented, giving the underlying theory and form of the traditional expressions. These can be quantified directly from overall engine performance tests without the need to interpret internal or component gas turbine performance. The second general topic is computer model simulations. These can be looked at as advanced overall performance methods that, with the aid of high speed computers, assemble much more complete numerical models of the propulsion systems and provide an enhanced capability to determine performance, including in-flight thrust.

4.1.1 General Methodology: An overall-performance thrust method involves the use of curves or tables that describe average engine performance or the performance of each specific flight test engine which has been calibrated in the ground-level test bed (GLTB) or altitude test facility (ATF). The performance is expressed in terms of engine operating parameter or parameters for defined engine inlet and exhaust-nozzle environmental operating conditions. For example, in the simple case of the choked-exit turbojet engine, the procedure can be entered with one engine parameter measured at specified in-flight conditions to yield net thrust directly as an output. The parameter can be a simple one, such as shaft speed, or one that is less directly obtained, such as fuel flow. Ideally, the parameter should accurately describe thrust regardless of the "health" of the engine. For redundancy and added confidence, the procedure may be structured to input a number of independent control parameters that are directly related to power setting and thrust. Other engines require more than one independent control parameter to define engine operating conditions and hence thrust, e.g., a variable-geometry turbojet or turbofan in afterburning.

Engines that are calibrated in the GLTB require their performance to be extrapolated to flight conditions. This involves correcting for changes in engine matching due to the effect of ram pressure ratio, inlet total pressure and temperature, and true specific heat variations from sea level to altitude; and accounting for nozzle thrust coefficient changes, leakage flows, customer power extraction, etc. Some of these corrections, such as those due to altitude, may be based on experience with the engine type rather than the individual flight engine. All engines calibrated in the ATF provide quiescent air (wind-off) thrust at a number of specific flight conditions and usually over a range of engine power settings. It may also be necessary to account for changes in engine airflow and thrust resulting from wind-on effects on engine matching.

Overall performance data are often presented in non-dimensional or referred forms using the thrust and airflow groups, as defined in Section 2. Gross thrust and airflow data may be expressed separately or may be combined to provide net thrust.

For example, using the ideal thrust expressions in line 2 of Table 2.1 and the definition of gross thrust coefficient, the gross-thrust group for the choked, fixed-nozzle turbojet or mixed-flow turbofan engine operating dry can be related to a representative spool speed by the implicit expression:

$$\frac{1}{2}\left(\frac{\gamma+1}{2}\right)^{\frac{1}{\gamma-1}}\left[1+\frac{1}{C_{G,\,con}}\left(\frac{F_G}{A_8\,P_{s0}}\right)_{act}\right]\frac{P_{s0}}{P_{t2}}=\frac{P_{t7}}{P_{t2}}=f\left(\frac{N}{\sqrt{\theta_{t2}}}\right) \tag{4.1}$$

4.1.1 General Methodology (Cont'd.):

This may be generalized for illustration purposes to yield the functional relationship:

$$\frac{F_{G,\,act}}{P_{s0}} = f\left(\frac{P_{t2}}{P_{s0}}, \frac{N}{\sqrt{\theta_{t2}}}\right) \tag{4.2}$$

Referred airflow may be expressed in the functional form:

$$\frac{W_{act}\sqrt{\theta_{t2}}}{\delta_{t2}} = f\left(\frac{P_{t2}}{P_{s0}}, \frac{N}{\sqrt{\theta_{t2}}}\right) \tag{4.3}$$

Hence, the corrected net thrust:

$$\frac{F_{N,\,act}}{P_{s0}} = f\left(\frac{P_{t2}}{P_{s0}}, \frac{N}{\sqrt{\theta_{t2}}}\right) \tag{4.4}$$

Iterative procedures may be used to evaluate subsidiary correction terms for airflow and nozzle thrust due to Reynolds number variations, real gas effects, actual nozzle pressure ratio, etc. Such supplementary data may be based on comprehensive GLTB and ATF engine calibration data and model nozzle data.

For a twin-spool engine, the referred speed of either the high- or low-pressure compressor may be used. The selected independent engine parameter may be different, e.g., fuel flow could be used.

The separate-exhaust turbofan engine can be treated as above, utilizing separate expressions for core and bypass gross thrusts, overall gross thrust, or net thrust, as appropriate.

The above describe single-degree-of-freedom cases that apply irrespective of the number or arrangement of spools within the engine. An engine with a single variable final nozzle operating in the dry mode requires a second independent variable in order to define engine performance. The nozzle area itself or a more convenient parameter may be used, thus:

$$\frac{F_{N,\,act}}{P_{s0}} = f\left(\frac{P_{t2}}{P_{s0}}, \frac{N}{\sqrt{\theta_{t2}}}, A_9\right) \tag{4.5}$$

4.1.1 General Methodology (Cont'd.):

In the case of a two-spool engine with variable geometry, the two referred spool speeds may be used as inputs to the overall performance method.

Engines operating in the afterburning mode require a third independent variable, such as afterburner fuel flow, to specify nozzle entry conditions and hence gross thrust. Engine performance up to the low-pressure-turbine exit plane may be expressed in terms of two independent variables, with an afterburner parameter providing the nozzle entry conditions, e.g.:

$$\frac{F_{N,\ act}}{P_{s0}} = f\left(\frac{P_{t2}}{P_{s0}}, \frac{N_L}{\sqrt{\theta_{t2}}}, \frac{N_H}{\sqrt{\theta_{t2}}}, \frac{W_{RF}}{\delta_{t2} \quad \theta_{t2}^{EXP}}\right) \qquad (4.6)$$

A common feature of all the overall performance methods outlined above is that flight thrust data can be generated without recourse to instrumentation within the engine gas path provided that the engine-entry-to-ambient-static pressure ratio, P_{t2}/P_{s0}, is defined and that the data provided for main and supplementary performance curves or tabulations are sufficiently comprehensive to account for aerothermodynamic changes which occur throughout the engine. In this respect, these overall performance methods differ from gas-path/nozzle methods, in that they can be constructed to yield calibrated-net-thrust data largely irrespective of turbomachinery performance. The methods treat the flange-to-flange engine and final nozzle as one assembly.

4.1.2 Computer-Model Simulations: The preceding section described overall performance methods for in-flight thrust determination that are obtained directly from engine tests in a GLTB or ATF. Other overall performance methods use engine digital-computer simulations to determine, in essence calculate, in-flight thrust. These methods also use available engine test data to reduce the uncertainty associated with the computer-model simulation; however, they are much more than a table-look-up/interpolation scheme for retrieving stored test data.

Computer-model simulations incorporate individual engine-component and nozzle performance linked together by an executive program logic that will satisfy continuity, energy, and momentum requirements to "match" the components at any specific engine operating condition. Thermodynamic subroutines must handle the real gas properties of air and air/combustion products for the temperatures of interest. Control functions and limitations are incorporated as necessary. Alternate match options should also be available in order to select specific independent variables, e.g., power lever angle, net thrust, fuel flow, rotor speed, or turbine inlet temperature. The general requirements for these digital-computer simulations are contained in SAE publication, AS 681C.(2.2)

4.1.2 Computer-Model Simulations (Cont'd.):

To obtain in-flight thrust, the inlet performance, nacelle secondary airflows, and installation losses (shaft horsepower and bleed flow) must also be specified. The computer program output may be limited to the net thrust and fuel flow or, depending upon the test program scope, may include detailed component performance at each simulated flight condition.

The accuracy of the simulation is dependent upon the quality of the component data available for the computer program. Initially in a development program, these data may be estimated from past experience with similar components and engines. Subsequently, component rigs and/or heavily-instrumented engines may be utilized to update each component performance estimate. If hardware changes are incorporated, they will necessitate corresponding changes to the computer model. By this process, the computer model is continually refined as the engine program progresses through the design, development, preliminary flight test, certification/qualification, and production phases.

There may be several different computer models that simulate a specific engine or a group of engines, as well as satisfy a specific intended use. These could reflect minimum engine performance as defined by contractual or other criteria, average performance, or any other special definition. For in-flight thrust determination, an average model which has been adjusted to particular hardware performance is desired.

The computer model may be used as the only thrust determination method or in conjunction with any of the other methods described in this report. In any case, all available supporting data should be utilized to improve confidence in the computer simulation. Prior to the flight test, all rig and engine test data should be correlated with the model prediction (both component and overall performance parameters) to refine the model of the particular test hardware. Unless test data are available from an ATF, the prediction must rely entirely on GLTB calibrations and component performance maps to calculate the engine performance at inlet and exhaust-pressure-ratio conditions encountered in flight. For such cases, it is imperative that the component test data cover the full range of operating conditions to which each component will be subjected. Component performance maps should properly account for engine-inlet pressure and temperature distortion and component interactions; such as radial distortion, inlet swirl, boundary layer effects, etc.

During and after flight test, confidence in the model simulation can be further improved by additional correlations of available flight test data. Even when the computer model is the single thrust-determination method and no special internal engine instrumentation has been added, some useful data may be available. Examples include fuel flow, rotor speeds (other than the thrust-setting parameter for multi-spool engines), and interturbine or exhaust-gas temperature.

4.1.2 Computer-Model Simulations (Cont'd.):

Where more than an overall performance method is planned for in-flight thrust determination, more information (gas path measurements) will be available. Correlation of these measurements with the computer simulation will add confidence in all methods being utilized. Conversely, a lack of correlation should be pursued to uncover difficulties and resolve differences. Thus, the computer-model simulation is a powerful performance determination method that has a role in most in-flight performance evaluation programs.

4.2 Gas-Path/Nozzle Options: Gas-path/nozzle methods utilize calibrated measurements of gas-generator-flow properties at various stations within the engine. Many method options are available for accomplishing this, but a particular propulsion system will have a preferred option or a minimal set of options.

The basis of virtually all methods is the separate determination of nozzle gross thrust or thrusts, and ram drag. Net thrust expressions for propulsion systems having single and dual-stream exhausts are presented in Section 2.1.

Gross thrust calculations require a knowledge of the nozzle geometry, entry conditions, operating pressure ratio, and nozzle performance. Ram drag calculations require a determination of the total engine-inlet airflow. Various options exist for obtaining mass flow at different locations within the gas generator flowpath. The engine-inlet air-mass-flow differs from the nozzle-exit gas-mass-flow by virtue of fuel addition, service bleed, leakage, etc. Thus, the determination of ram drag using a nozzle flow measurement requires accounting for secondary flows.

All gas-path/nozzle methods require determination of ambient pressure, P_{s0}; free-stream total pressure, P_{t0}; and inlet total temperature, T_{t2}. Other variables which may require determination include:

- o Inlet total pressure, P_{t2}
- o Nozzle mass flows, W_9 and W_{19}
- o Engine inlet mass flow, W_2
- o Nozzle areas, A_8, A_9, A_{18} and A_{19}
- o Nozzle entry temperatures, T_{t7} and T_{t17}
- o Nozzle entry pressures, P_{t7}, P_{s7}, P_{t17} and P_{s17}

The particular variables to be measured or calculated will depend on the gas-path/nozzle method option adopted. In many cases, aircraft and calibrated flight development engines are instrumented to provide more information than may be required to meet the need for in-flight thrust determination. Notable examples are: accurate fuel flow measurement to determine aircraft specific range and engine specific fuel consumption, and data to compare engine component behavior with projected component performance or to confirm control functions. Various stations throughout

4.2 Gas-Path/Nozzle Options (Cont'd.):

the engine can be used for evaluating the flow quantities needed for in-flight thrust determination.

A universal methodology applicable to all propulsion systems cannot be considered feasible. Optional choices are highly configuration dependent and involve clear identification of engine/airframe interfaces within an agreed thrust/drag bookkeeping system for a particular configuration.

The number of combinations of flow-path options is large.[4.1, 4.2, 4.3, 4.4] It is not the intent herein to evaluate them in depth for the many engine configurations that exist in service or are under development. It is sufficient to note that options do exist and that it is important to assess them for each individual case. Preferred options should be identified for each powerplant early in its development. To facilitate this, an uncertainty analysis should be used to discard the less desirable options. Since circumstances may change during the course of engine development, various preferred options should be kept open. The number of options should be related to thrust accuracy requirements.

4.2.1 Mass Flow Determination: Some of the more important methods for in-flight determination of mass flow are discussed. Experience with various methods, backed by an uncertainty analysis, will lead to the choice of preferred method(s) for a particular application. Comparison of post-test ground and flight data should be consistent for each method. Paragraph 5.1.5.2 discusses airflow measurements in engine test facilities. These facility measurements should be used to calibrate an in-flight method or, at least, to check consistency.

4.2.1.1 Calibrated Flight Inlet: For some flight inlet configurations, it is practical to add instrumentation that will permit the flow area to be used much like a venturi flow-measuring device. The flight inlet must be calibrated using facility flow measurements or other engine mass-flow-determination methods. When the flight inlet is calibrated or used at pressures substantially lower than the sea-level standard, adjustment of the measured flow may be required to account for boundary layer differences. Consistency of pressure instrumentation and spinner geometry, etc. must be maintained.

4.2.1.2 Compressor Flow Capacity: Compressor flow capacity may be used to calculate core-engine airflow. The pressure ratio and referred speed are measured in the engine, and the flow determined from the compressor performance map. The measurements of the compressor inlet temperature and pressure, and exit pressure must be comparable to those made to construct the component map. Adjustments for circumferential and radial profiles, Reynolds number, radial tip clearances, and humidity may be required. Adjustment for interference effects resulting from engine compressors operating in series, rather than singularly, as in a rig, may be necessary. If the compressor is equipped with variable stators, vane-angle correlations will also be necessary.

4.2.1.2 Compressor Flow Capacity (Cont'd.):

The general method need not require a component performance map. By using facility measured airflow from an engine test, the compressor (and engine) flow capacity can be correlated to engine referred speed and compressor pressure ratio. For some systems, ram pressure ratio, P_{t2}/P_{s0}, or some other parameter may be used as the second correlating parameter, thereby reducing the instrumentation requirements.

For a turbofan engine, total inlet airflow may be determined in a similar manner using the fan flow capacity.

4.2.1.3 Turbine Flow Parameter: The turbine flow parameter, $W\sqrt{T_t}/P_t$, is especially useful for determining core-engine airflow because it is very nearly constant over a wide range of engine operating conditions. The choked value may be determined from rig tests and modified for the turbine-nozzle area in the test engine. It can be calibrated during an engine ground test using engine instrumentation and airflow determined by another method. Corrections to the flow parameter are required as functions of turbine pressure ratio (or an equivalent parameter), referred speed, and gas properties. Other adjustments may be necessary for the effects of thermal expansion, etc.

Turbine-nozzle inlet total temperature and pressure are needed to calculate gas mass flow from the flow parameter. These are normally obtained by measuring compressor discharge temperature and pressure and applying calculated corrections for the combustor temperature rise and pressure loss. The combustor calculations will require fuel flow and airflow, and will be iterative.

4.2.1.4 Temperature and Pressure Rakes: Mass flow can be obtained from engine internal static-pressure and total-pressure and -temperature instrumentation. After installation, the flow correlation should be calibrated in an engine test using flow determined from another method. Potential flow-path locations which may accommodate the required instrumentation rakes include the compressor exit, the bypass duct, and the low-pressure turbine exit. Sufficient instrumentation (numbers of probes and rakes) is required to obtain representative average pressures and temperatures over the intended range of engine operating conditions. These diverse conditions may introduce significantly different radial and circumferential profiles at the measurement plane. The large amount of instrumentation required for this method may be a major disadvantage.

4.2.1.5 Thrust Nozzle: Mass flow from the thrust nozzle can be calculated if the nozzle-entry total pressure and temperature, nozzle pressure ratio, nozzle area, and flow coefficient are known. If entry conditions incorporate large profiles or swirl, their effects should be included in the nozzle performance determination. The instrumentation is essentially the same as required for nozzle specific thrust determination, discussed in Paragraphs 2.5.2 and 4.2.2. The thrust nozzle method of mass flow determination is especially useful because

4.2.1.5 Thrust Nozzle (Cont'd.):

the "linked methodology" of using the same airflow method for gross thrust and ram drag cancels some errors in the resulting net thrust.[1.1, 4.1, 4.2]

4.2.1.6 Energy Balance: For turbofan engines, the above methods, when used singularly, determine either inlet total airflow or core-engine flow, but not bypass ratio. Energy-balance calculations using temperature rise in the core engine and bypass duct can be used to determine the flow split between the two streams. Fuel flow, T_{t2}, T_{t7}, and T_{t17} measurements are required.

4.2.2 Nozzle Thrust Determination: Nozzle thrust determination requires a knowledge of nozzle size, entry conditions, pressure ratio, and nozzle performance. Nozzle thrust may be described as either gross thrust or specific gross thrust, which must be multiplied by nozzle mass flow to obtain gross thrust. In-flight, the above option may lead to a choice of measuring either nozzle area (gross thrust) or nozzle inlet temperature (specific thrust). Other measurements will also be required.

The selected method may correlate actual thrust directly with the measured parameters or may correlate measurements to a nozzle coefficient. The thrust coefficient relates real nozzle performance to that of an ideal nozzle, as described in Paragraph 2.5. The use of nozzle coefficients permits a direct "visualization" of nozzle performance, which is separated from the gas generator performance, and is an excellent technique to ensure credible and consistent performance representations.

For a separate-stream turbofan engine, the fan-duct (bypass) thrust and airflow can be determined by subtracting the core-engine thrust and airflow from the total thrust and airflow that are measured in the GLTB/ATF. Normally, the core (primary) thrust will be determined using scale-model-derived coefficients and full-scale measured values of nozzle-inlet temperature and pressure and nozzle pressure ratio. Fan-duct nozzle thrust and flow coefficients can be calculated from the full-scale fan-duct thrust and airflow, and the ideal bypass-nozzle thrust using the nozzle entry conditions and pressure ratio. Any error in the primary nozzle coefficients would affect the level, and possibly the consistency, of the bypass-nozzle coefficients. Such errors will not significantly impact total thrust (or airflow) if the same procedure is used throughout the ground- and flight-test program.

The nozzle throat area can also be used in a calculation of flow continuity to verify the afterburning nozzle-inlet total temperature.

The following paragraphs discuss the several parameters required for in-flight nozzle thrust determination.

4.2.2.1 Nozzle Area: The in-flight nozzle area is required when the gross-thrust (or PA) option is selected. For fixed area systems, this can be obtained from physical measurement of the flight test hardware with an adjustment for the operating metal temperature. For variable geometry systems, some form of in-flight measurement will be required.

4.2.2.1 (Continued):

The nozzle area can also be used in a calculation of flow continuity to verify the afterburning nozzle-inlet total temperature.

4.2.2.2 Nozzle-Inlet Total Temperature: The nozzle-inlet total temperature is required when the specific-thrust (or $W\sqrt{T}$) option is selected. Nozzle mass flow will also be required. The same temperature may be used in conjunction with a nozzle flow coefficient to calculate this flow. The temperature may be measured or determined from a correlation with other measured parameters such as speed or pressure ratios.

For non-afterburning engines, direct measurement of nozzle entry temperature is accomplished by placing thermocouple rakes downstream of the LP turbine and in the bypass duct. The temperature measurement in the engine will not ordinarily provide the true average temperature due to profiles and the practical limit placed on the number of thermocouples. The average temperature will differ from scale-model tests used to evaluate nozzle performance, and the magnitude of the discrepancy can vary with flight condition and from one engine to another.

For afterburning engines, direct in-flight measurement of nozzle entry temperature is not generally feasible. Inlet temperature to the afterburner is usually measured and corrected for the heat addition to obtain the nozzle inlet temperature. Other methods of calculating nozzle inlet temperature are also possible using heat balance procedures.

4.2.2.3 Nozzle-Inlet Total Pressure: All gas-path/nozzle methods require nozzle-inlet total pressure. For non-afterburning engines, the most common approach is to place total pressure probes downstream of the LP turbine and in the bypass duct. The effect of pressure profiles on the "average" pressure inferred from the individual measurements must be carefully evaluated, as discussed above for the nozzle-inlet total temperature.

For afterburning engines, direct pitot-probe measurements at the nozzle inlet may be feasible using water-cooled probes. A more common practice is to place the pitot rakes ahead of the afterburner and correct the pressure using a calculated value of pressure loss to the nozzle inlet.

For some in-flight applications it is possible to infer average nozzle-inlet total pressure from static pressure measurements, if the total-to-static pressure correlations have been predetermined during calibration tests.

4.2.2.4 Nozzle Operating Pressure Ratio: Nozzle environmental (base) static pressure, P_{sB}, at the nozzle exit plane may differ significantly from the ambient static pressure, P_{s0}. The relationship between base and ambient pressures may vary between wind-on and wind-off conditions and with engine power setting. A clear distinction between nozzle exhaust pressure ratio, P_{t7}/P_{s0}, and nozzle applied pressure ratio, P_{t7}/P_{sB}, must be made. Gas-path/nozzle thrust method options can be

4.2.2.4 Nozzle Operating Pressure Ratio (Cont'd.):

constructed using either P_{t7}/P_{s0} or P_{t7}/P_{sB} (though in some cases the measurement of P_{sB} to sufficient accuracy may be impractical). The options formulated for a single nozzle may be illustrated by the functional relationships of Table 4.1.

Nozzle Pressure Ratio	$\frac{P_{t7}}{P_{s0}}$ OPTION (Exhaust Pressure Ratio)	$\frac{P_{t7}}{P_{sB}}$ OPTION (Applied Pressure Ratio)
Nozzle Discharge Coefficient	$C_D = f\left(\frac{P_{t7}}{P_{s0}}, \text{geometry}, M\right)$	$C_D = f\left(\frac{P_{t7}}{P_{sB}}, \text{geometry, residual effects}, M\right)$
Nozzle Gross Thrust Coefficient	$C_G = f\left(\frac{P_{t7}}{P_{s0}}, \text{geometry}, M\right)$	$C_G = f\left(\frac{P_{t7}}{P_{sB}}, \text{geometry residual effects}, M\right)$

Table 4.1 - Nozzle Pressure Ratio Options

In some cases when nozzle thrust coefficients are only weakly dependent upon external flow, it may be appropriate and expedient to utilize calibrated static or wind-off values.

4.2.2.5 Thrust Coefficients: Nozzle thrust coefficients, together with nozzle flow coefficients, are the prime variables that account for departures of actual gross thrust from the ideal value. They are functions of nozzle pressure ratio and nozzle geometry. Actual flow effects that may be accounted for in the thrust coefficients include:

- o Three-dimensional nature of flow in the nozzle
- o Corrections for real gas effects which may arise in the application of model nozzle test data to full scale, particularly at high-pressure low-temperature conditions
- o Nonuniformity of pressure and temperature profiles across the duct at the nozzle entry plane
- o The coverage (number and location) of the pressure and temperature probes, which will not in general give mean values
- o Non-axial flow at the nozzle inlet and exit, including swirl
- o Value of γ used for thermodynamic functions
- o Dissociation of real gases at high temperature and energy mode fixation during rapid nozzle expansion

4.2.2.5 Thrust Coefficients (Cont'd.):

- o Errors in estimation of pressure losses between the measurement plane and the nozzle entry, particularly with afterburning

- o Mass flow leakage from the nozzle.

Additional terms may be included in the thrust and flow coefficients, e.g., for separately exhausted dual stream nozzles, wind- off cowl scrubbing forces may be lumped with Station 19 coefficients and wind-off core-flow plug forces may be lumped together with Station 9 coefficients.

Nozzle flow and thrust coefficients are determined by testing nozzles in which mass flow, nozzle-entry total pressure and temperature, and nozzle-exit static pressure or ambient pressure are measured together with nozzle thrust over the nozzle operating pressure ratio range. Theoretical estimates may be made.

Coefficients are obtained from representative subscale and full-scale isolated nozzle or installed nozzle-plus-afterbody assemblies tested in quiescent air (wind-off) conditions. Nozzle coefficients may be derived from installed subscale-nozzle wind-on tests when the nozzle assembly is metric. Nozzle coefficients may be derived from full-scale quiescent-air tests of the engine on the GLTB and in the ATF. Coefficients established from engine-plus-nozzle tests include the effect of nozzle-entry flow profiles and can be referenced to the instrumentation to be used in flight for a defined nozzle entry plane. Any differences between the GLTB/ATF and aircraft installations that could change the nozzle-entry profile should be evaluated.

For engines that operate with choked nozzle(s) at the high-power ground-level-static conditions, nozzle coefficients may be based on engine GLTB and scale-model nozzle testing, as illustrated in Figure 4.1. The model coefficient curve shape is used to extrapolate the full-scale nozzle coefficients to cruise nozzle pressure ratios. As illustrated by the cross-hatched area in Figure 4.1, the uncertainty associated with the extrapolation is relatively small, since it is assumed the precision (random) errors are reduced by obtaining sufficient data.

A high-bypass-ratio turbofan will have unchoked nozzle(s) during ground-level-static operation. The engine match, fan operating lines, and pressure and temperature profiles at the fan nozzle entry station will differ between high-speed cruise and static operation. This results in increased uncertainty when the model coefficient curve shape is used to extrapolate nozzle coefficients to cruise nozzle pressure ratios, as illustrated in Figure 4.2. Nozzle coefficients derived from ATF engine testing can be used to reduce the uncertainty band (Figure 4.2).

A detailed discussion of how nozzle thrust and flow coefficients are measured is contained in Section 5.

Notes:
1. Choked nozzles result in similar operating lines at sea level and cruise
2. Extrapolation based on scale model shape and measurement error
3. Legend: TO = Typical Takeoff Power
   CR = Typical Cruise Power
   0 = SLS Outdoor Testing

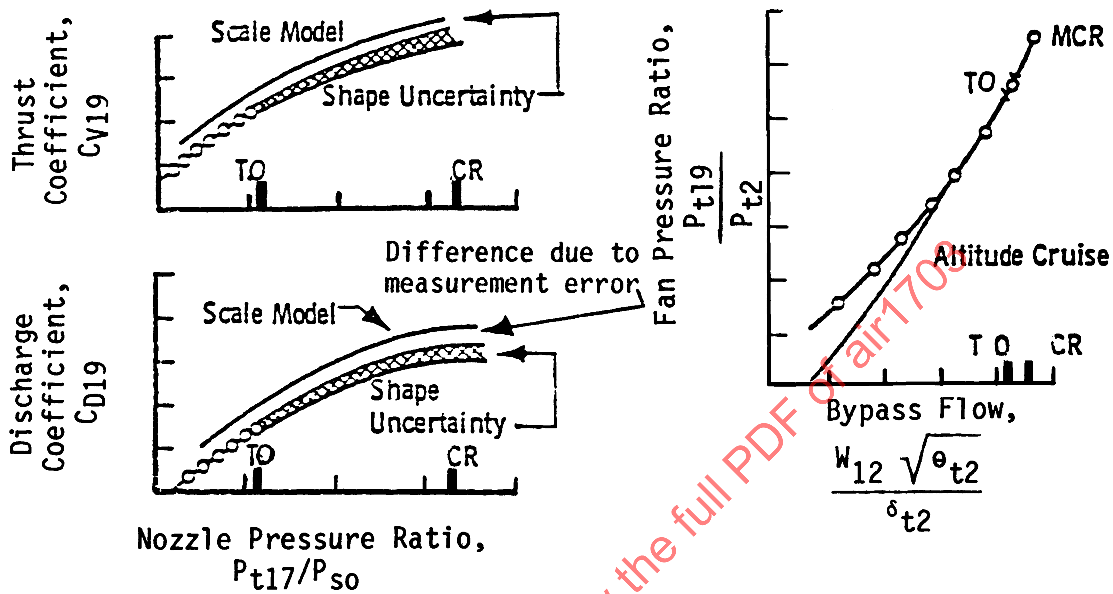


Figure 4.1 - Nozzle Coefficient Extrapolation for Low-Bypass-Ratio Engine

Notes:
1. Operating lines are different at sea level and cruise
2. Greater uncertainty in extrapolation of measurement error
3. Legend: o SLS Outdoor Testing
   □ Altitude Test Data

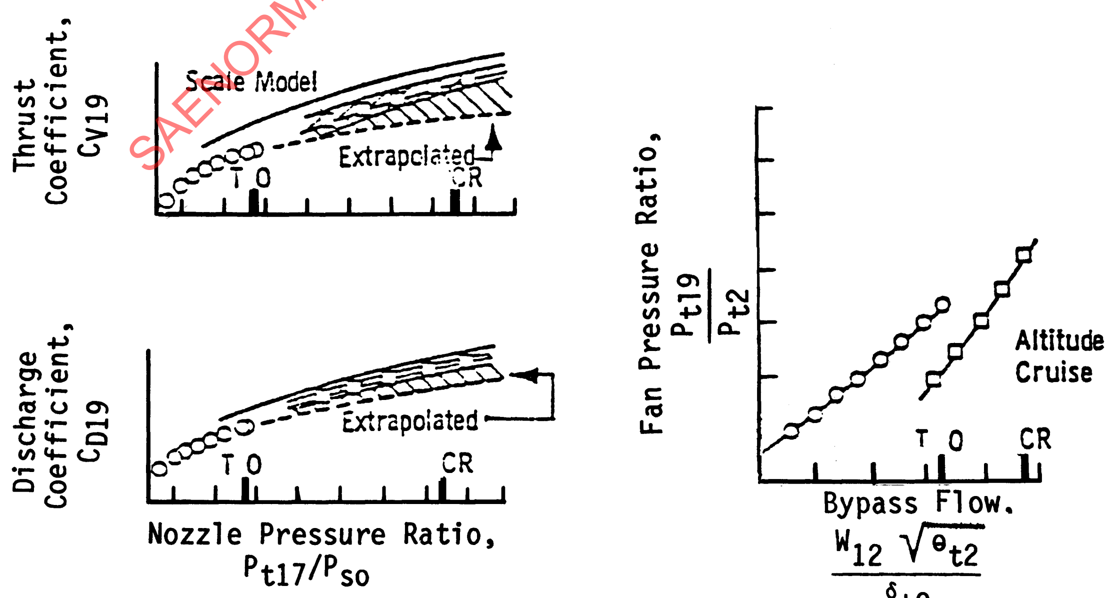


Figure 4.2 - Nozzle Coefficient Extrapolation for High-Bypass-Ratio Engine

4.2.2.5 Thrust Coefficients (Cont'd.):

In order for model and full-scale coefficients to be compatible, adjustments must be made for differences in Reynolds number, surface roughness, geometry detail such as steps and gaps and acoustic treatment, leakage, instrumentation loss, real-gas effects, flow mixing, profiles, and swirl. Some of the required adjustments are easily made; others are more difficult and peculiar to a particular installation. Part of the method application process to ensure accurate in-flight thrust determination is to reconcile differences between exhaust nozzle coefficients derived from model and full-scale tests.

4.2.3 External Flow Effects: Engine gross and net thrust may be affected by flow conditions in the nozzle operating environment and airflow quality at the aircraft inlet/engine interface. For some propulsion systems, e.g., the short cowl turbofan and the integrated nozzle/afterbody boattail system, it may be impractical to separate the forces acting on the surfaces external to the nozzle exit plane from the quiescent (wind-off) calibrated thrust. Significant shifts in matched engine operating conditions, thrust and external forces can occur between wind-off and wind-on conditions. A clear thrust/drag bookkeeping methodology is particularly important in such cases.

4.2.3.1 Nozzle Environment: Existing engine GLTBs and ATFs do not provide a capability for acquiring thrust calibration data in wind-on conditions. Moreover, significant local external-flow effects can arise due to the proximity of adjacent wing and fuselage surfaces. In such cases, extensive use of propulsion-system wind-tunnel model tests may be required to correct wind-off data for the effects of forward speed and account for all thrust, force, and drag terms needed to establish installed propulsive force. A detailed discussion is presented in Section 5.

Subscale tests provide preflight information of external-flow effects on nozzle flow coefficients and, hence, a preliminary guide to engine rematching in flight. They also provide a supplementary aid for diagnosing engine performance changes signaled by the gas generator flight instrumentation. Wind-tunnel model data provide quantitative guidelines on the relationships between P_{sB}, P_{s0}, nozzle operating pressure ratios, thrust and flow coefficients, and flight speed at simulated flight conditions.

4.2.3.2 Engine Inlet Conditions: Engine gross thrust and airflow at a given aircraft and engine operating condition depend on the engine-inlet total pressure and temperature. Engine-inlet total pressure needs to be measured to compare measured and predicted engine performance and flight thrust, to set up GLTB and ATF flight-engine calibration tests, and to expedite some in-flight gas-path/nozzle method options, e.g., those involving airflow determination from inlet or compressor normalized flows. Other method options, such as those employing calibrated

4.2.3.2 Engine Inlet Conditions (Cont'd.):

instrumentation internal to the engine, do not require knowledge of the inlet pressure directly.

Engine-inlet average total pressure and inlet total-pressure-recovery factor should be defined and agreed upon for a particular propulsion system development. The definition should be used consistently through all phases of the propulsion system development.

4.3 Nozzle Exit Traverse: One theoretical approach to the determination of in-flight thrust is to measure the gas properties at the nozzle exit and integrate across the area to calculate the exhaust velocity, pressure-area force, and mass flow. The traversing rake is a tool that has been used to accomplish this.

The traversing rake measures total pressure, static pressure, and total temperature across the flow field. This can be done rather quickly to minimize exposure time to the hostile environment; however, the scan rate must be slow enough to provide time to acquire representative data. The scan rate must be selected for the particular installation, considering data acquisition requirements and the drive system.

Velocity and mass-flow-per-unit-area are calculated from the pressure and temperature measurements along the arcs travelled by the instrumentation, as illustrated in Figure 4.3. Along with the measured static pressure, these must be integrated in a manner representative of the exhaust stream geometry to yield overall mass flow, momentum, and pressure-area force.

The use of a traversing rake for in-flight thrust determination has had limited application, because of the relative complexity of the installation coupled with the difficulty of correlating associated installation effects.

4.3.1 Rake Design and Installation:

4.3.1.1 Number of Measurements: The location and number of individual measurements required is highly installation sensitive and must be developed for each application. For example, a single-stream turbojet would tend to have more uniform exit flow properties than a dual-stream turbofan and, therefore, would require fewer measurement locations. The task is eased if the exhaust nozzle system is symmetric, so that measurements can be made from one pivot location and applied to the "mirror image" second side. Comparison of data from different rake pivot locations enables the determination of the degree of exit flow symmetry.

Ground tests to evaluate the number of measurements required, and the rake geometry and pivot location are mandatory to establish a satisfactory rake design and installation. An ATF evaluation of the system capability to accurately determine thrust is recommended.

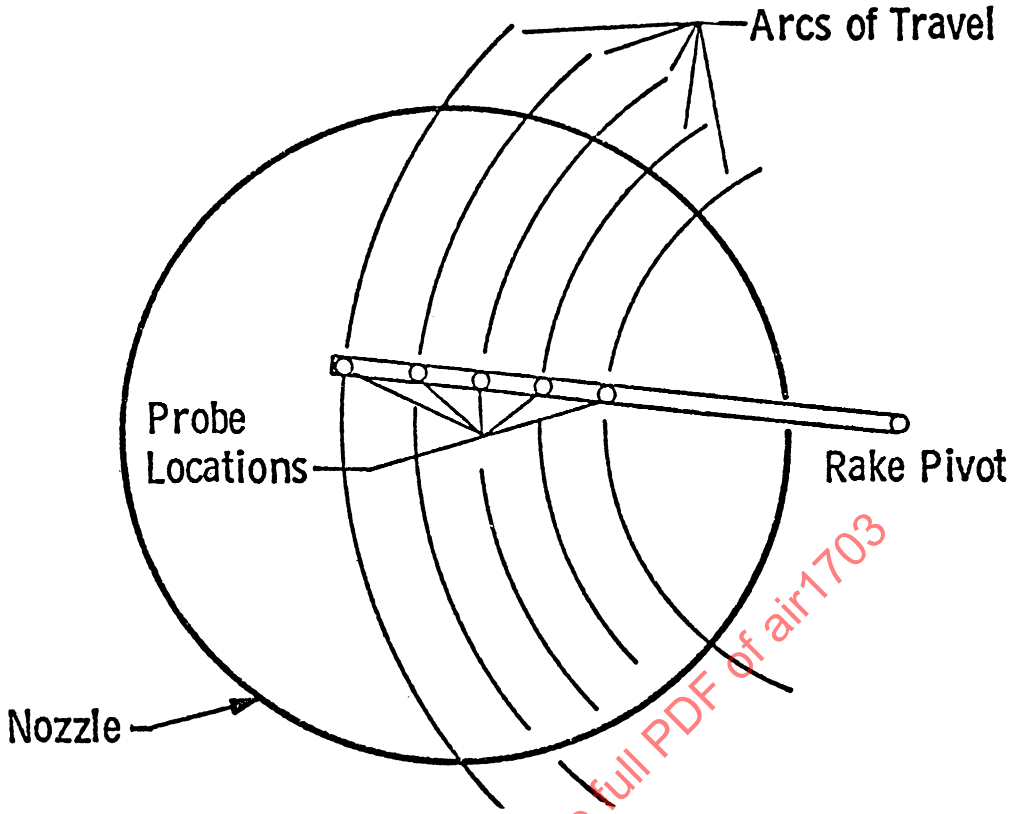


Figure 4.3 - Traversing Rake Arrangement

4.3.1.2 Rake Mounting: The rake size and pivot location must permit traversing the nozzle exit plane to provide sufficient temperature and pressure data. A major consideration is to find a satisfactory part of the aircraft upon which to mount the rake. Accessible aircraft structure is required, which may not be available for all propulsion installations.

The rake and its mounting hardware influence the aircraft area distribution at the location where they are installed. This can be a critical area from the aircraft drag viewpoint, and the rake itself can influence airplane performance. The determination of airplane performance requires accounting for the drag of the rake installation. Different methods can be used. One method is to build a model of the rake to the scale of the aerodynamic force model and develop a rake tare in the wind tunnel. Another approach is to develop a generalized set of thrust curves from the flight data for the installation using the rake and then repeated stabilized maneuvers without the rake. The thrust differential between rake-on and rake-off tests are used to determine the rake drag increment. This requires care to assure similarity of flight conditions and to correct the thrust for ambient temperature conditions.

4.3.1.3 Environmental Considerations: The rake operates in a hostile environment, with temperature and sonic fatigue being major life-limiting concerns. Temperature is far more extreme in an afterburning engine. The adverse effects of temperature are reduced by cooling and, 4.3.1.3 Environmental Considerations (Cont'd.):

most significantly, by decreasing the exposure time to the high temperature. This brief exposure period is the major reason for employing a traversing rake rather than installing a fixed set of probes. Also, sonic fatigue effects are reduced by operating the rake in the hostile environment for short periods only. The speed at which the rake is traversed across the flow stream is a compromise between requirements for reducing the exposure to the hostile environment and providing sufficient time for accurate temperature measurement.

4.3.2 Data Analysis: The exhaust gas velocity and mass flow are calculated by integrating the traversing-rake pressure and temperature profiles over the cross-sectional area of the internal stream tube. Integration limits, in terms of radial distance, are defined by the condition that total and static pressure existing at the edges of the stream tube are constant. A weighted area must be defined for each probe, as a function of angular rake position, and used in the integration calculation. Initially, this would be accomplished by assuming symmetry of the exhaust stream. Moving the rake pivot and probe locations for identical operating conditions permits verification of this assumption.

Use of the traversing rake for measuring thrust in flight requires a careful ground calibration. The time constant of the temperature probes must be determined. Temperature probe correction factors can be developed by comparison of rake data taken at selected scan rates with known temperature data obtained from a fixed rake at similar power settings.[4.5]

4.4 Trunnion Method: The trunnion method is based on evaluating net thrust from the measurement of reaction forces of the engine and nozzle assembly on the engine airframe mounts, or trunnions, together with appropriate stream thrust and integrated pressure force terms.[4.6, 4.7] For the simplified illustration shown in Figure 4-4 where none of the external nacelle surface forces act through the engine mounts, the trunnion force, F_T, is given by:

$$F_T = F_{G9} - WV_2 - A_2 (P_{s2} - P_{s0}) + (A_2 - A_9) (P_{se} - P_{s0}) \qquad (4.7)$$

The pressure/area terms represent appropriate integrals or mean values, e.g., P_{se} is the average static pressure over the external surface of the engine. Thus, knowledge of internal flow variables and external engine-surface pressure forces is necessary. Hence, the net thrust is given by:

$$F_N = F_T + W(V_2 - V_0) + A_2 (P_{s2} - P_{s0}) - (A_2 - A_9) \ (P_{se} - P_{s0}) \qquad (4.8)$$

The above equation for the simplified installation shows that F_T approximates the net thrust at low subsonic flight speeds. As the flight speed increases, the momentum- and pressure-difference terms become large relative to F_T, so the net thrust is sensitive to correctly assessing and measuring these terms. Care must be exercised in measuring the integrated average pressures and computing the engine stream thrust in flight at representative

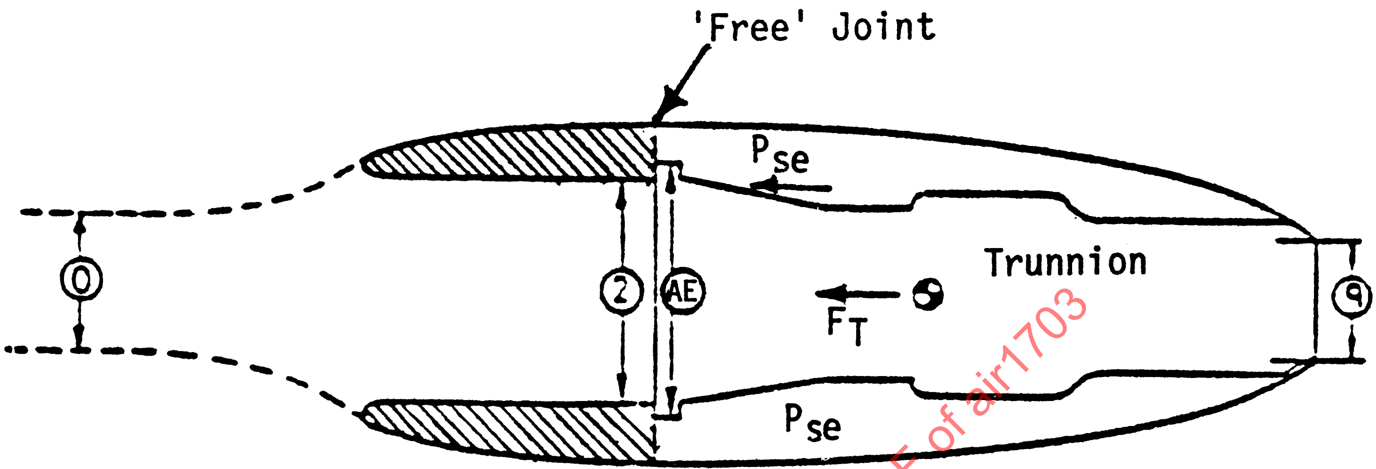


Figure 4.4 - Simplified Trunnion Thrust Method

4.4 Trunnion Method (Cont'd.):

mass flow rates. Accurate measurement of the trunnion loads requires that close attention be given to the design and installation of the instrumentation. Seal friction forces and tare loads from airframe to engine connections must be included in the force accounting. For more complex installations, such as a podded turbofan, the trunnion measured force would include some external nacelle forces and drags. When the nozzle assembly is mounted separately from the flange-to-flange engine, the evaluation of net thrust requires that loads on the nozzle mounting points be measured.

The direct force measurement system may be extended to podded installations by employing pylon-mounted load cells. If interference forces are negligible, the thrust-minus-drag of the pod would be determined and could be used directly, or corrected for drag terms, to yield in-flight net thrust.

The use of the trunnion method for measuring in-flight thrust has had limited application to date. At least in part, this has been because of the difficulties of instrumenting the mounts, recording the loads, and assessing the in-flight tare forces between the airframe and engine.

4.5 Examples of In-Flight Thrust Methods: Several simplified examples are discussed to illustrate the use of in-flight thrust methods. Logic charts (schematics) are presented to depict the particular measurement and calculation process. Such charts should be prepared early in the planning process for each candidate method. They depict the in-flight measurements

4.5 Examples of In-Flight Thrust Methods (Cont'd.):

required and the data correlations that must be developed prior to the flight test.

4.5.1 Single-Spool Fixed-Nozzle Turbojet: The use of an overall performance method for a simple turbojet is illustrated in Figure 4.5. Measurements are designated within a rectangle and calculated parameters within a parenthesis.

The gross thrust relationship, which is used to correlate in-flight thrust, is essentially that given previously in Equation 4.1; while the airflow relationship, which is needed for ram drag, is based upon the compressor speed/flow operating line. Secondary parameters are shown to correct for Reynolds number and real gas effects. All of these relationships must be developed through GLTB and ATF engine calibrations.

All in-flight thrust methods require a knowledge of P_{s0}, T_{t2}, and P_{t2} (or P_{t1}) at each data point. The simple overall performance method presented in Figure 4.5 requires only one additional measured value (N) as input to the calculation procedure.

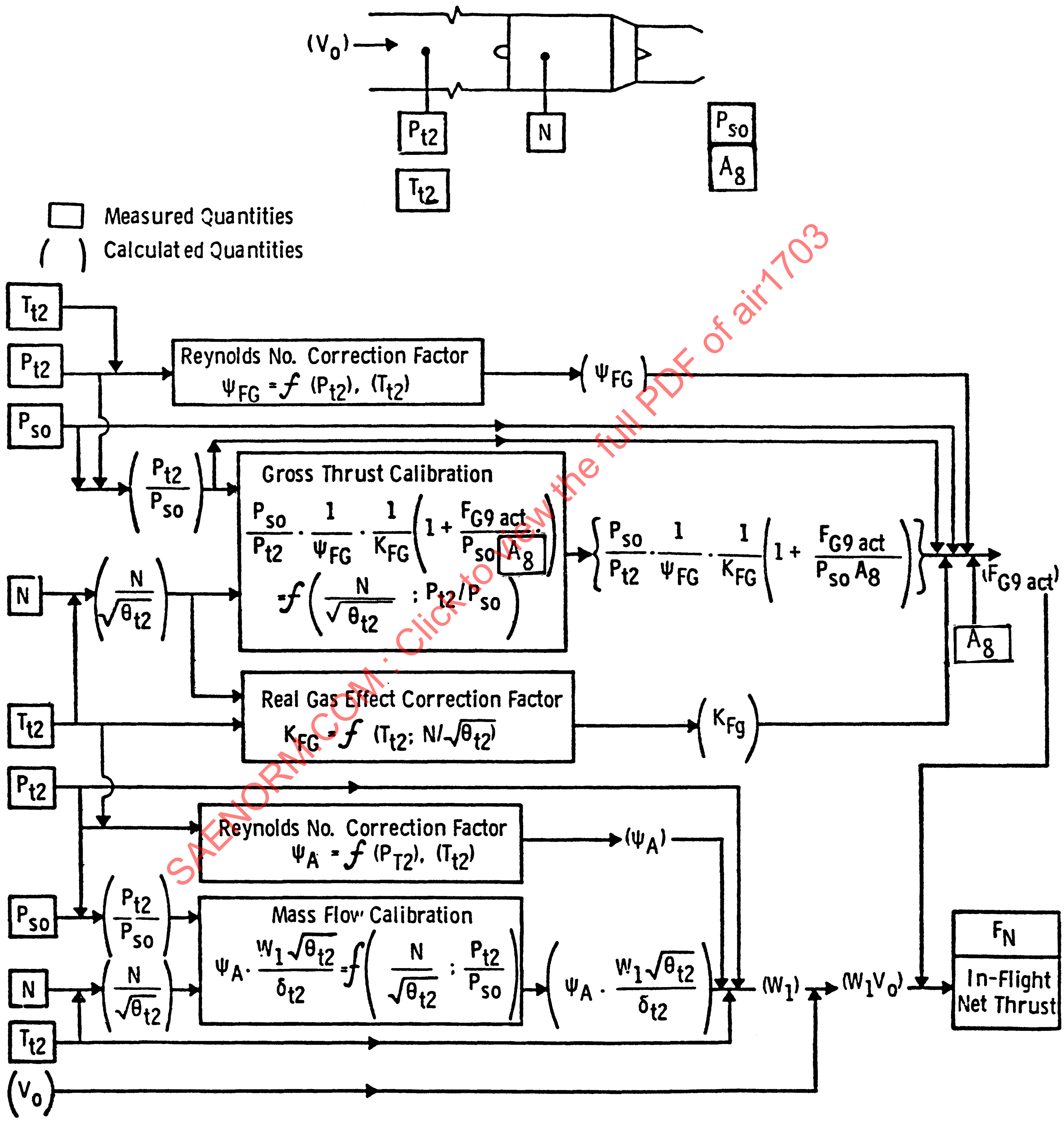
4.5.2 Single-Exhaust Turbofan: Three different options are presented including both simple gas-path/nozzle and overall performance methods. The use of multiple methods to provide redundancy is a desirable feature for any test program.

All three of the examples make use of a fan-speed correlation to determine total airflow and ram drag. Thus, the methods are not totally independent.

Figure 4.6a illustrates a gas-path/nozzle method for in-flight thrust determination. A correlation of referred gross thrust, F_{G9}/δ_{t2}, versus area-weighted nozzle pressure ratio, NRPA, and ram pressure ratio, P_{t2}/P_{s0}, is used based upon ATF measurements. In-flight measurements of core and bypass nozzle-entry pressures are required, in addition to low rotor speed, for the airflow calculation.

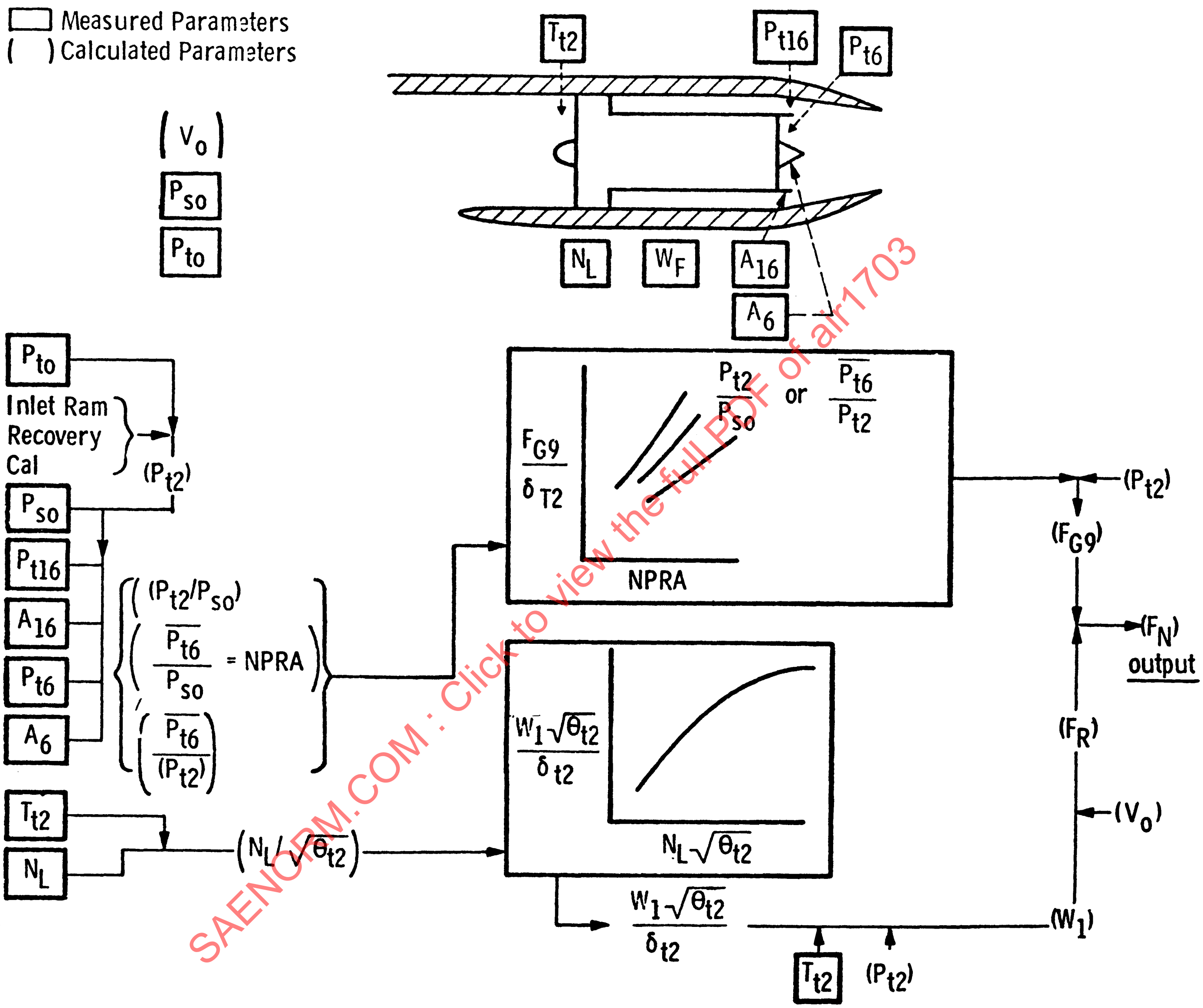
Figure 4.6b and c illustrate in-flight thrust methods which do not require gas path instrumentation and are therefore classified as overall performance methods. Most flight test programs require the measurement of fuel flow. From ATF calibrations, referred gross thrust can be correlated in terms of referred fuel flow, $W_F/\theta_{t2}^{\,exp}\ \delta_{t2}$, and ram pressure ratio (or an alternate gas-generator parameter). The exponent in referred fuel flow by simple dimensional analysis is 0.5 but actually may be higher due to real gas properties.

Single Spool, Fixed Nozzle Turbojet ~ In-Flight Thrust Evaluation Using Overall Performance Method



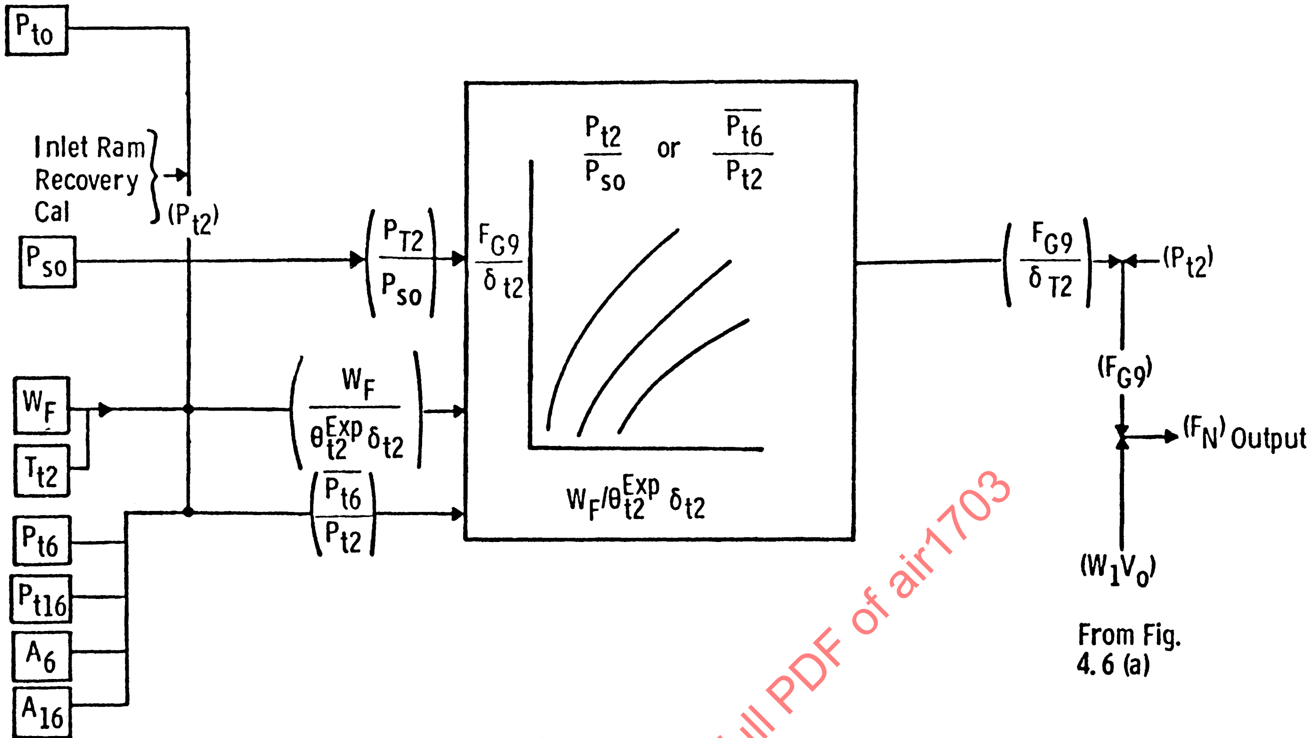
Note: Loops for Customer Bleed and Power Off-Take Omitted for Clarity

Figure 4.5 - Overall Performance Method for Single-Spool Fixed-Nozzle Turbojet

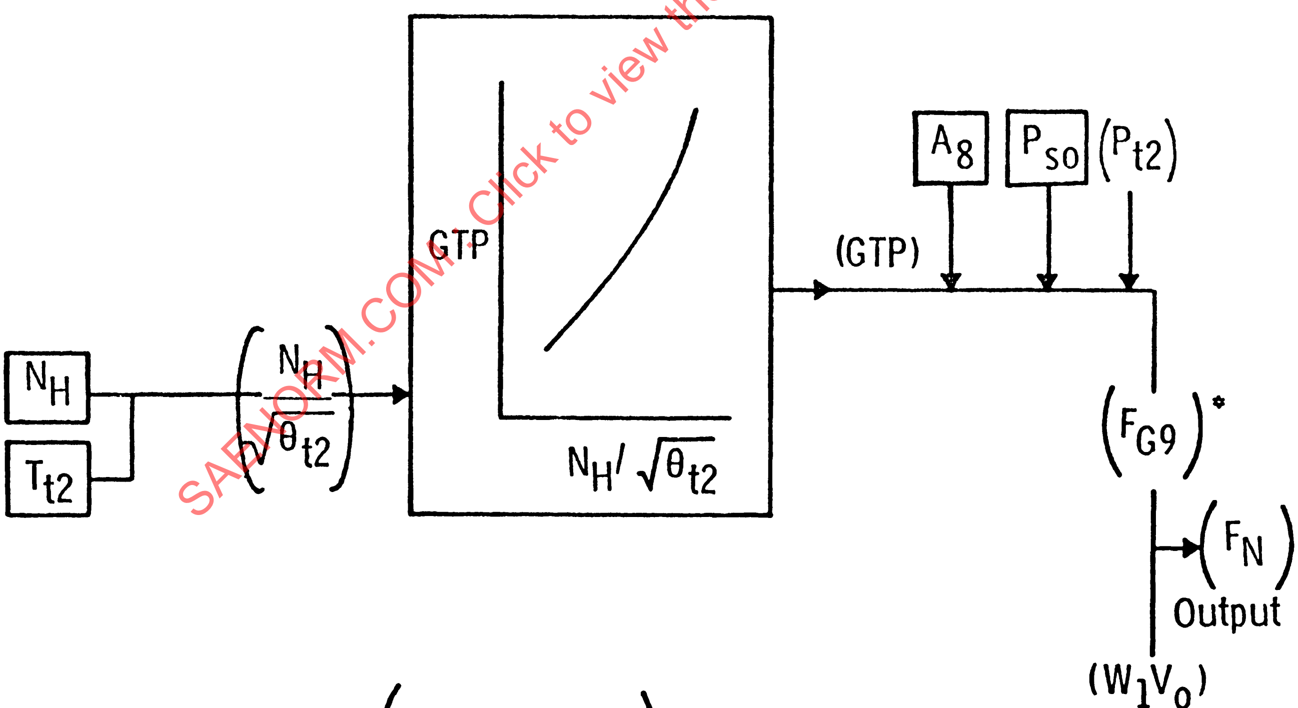


a. Area-Weighted Nozzle Pressure Ratio Method

Figure 4.6 - In-Flight Thrust Evaluation for Single-Exhaust Turbofan



b. Gross Thrust/Fuel Flow Correlation Method



$$* \quad F_{G9} = \left([GTP]\; A_8\, P_{t2} \right) - A_8\, P_{so}$$

From Figure 4.6 (a)

c. Gross Thrust Parameter Method

Figure 4.6 (Cont'd) - In-Flight Thrust Evaluation for Single-Exhaust Turbofan

4.5.2 Single-Exhaust Turbofan (Cont'd.):

The third in-flight thrust method (Figure 4-6c) uses a correlation of gross thrust parameter, GTP, versus referred rotor speed. From ATF measurements, the GTP is calculated:

$$GTP = \left(\frac{F_{G9}}{A_8 \, P_{s0}} + 1\right) \frac{P_{s0}}{P_{t2}} \tag{4.9}$$

This correlation (essentially the same as shown in Figure 4.5) is insensitive to Mach number when the thrust nozzle is choked.

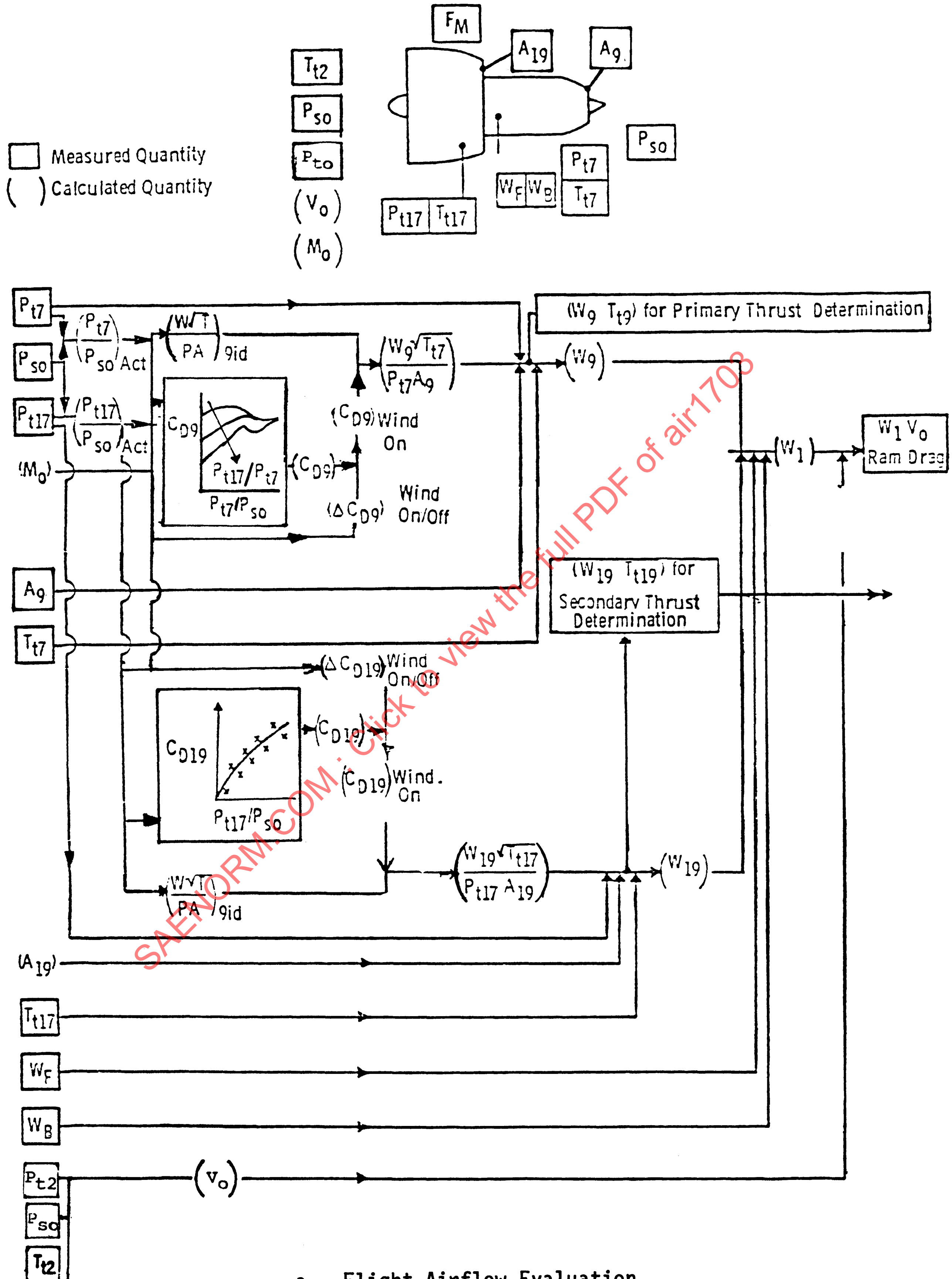
4.5.3 Separate-Stream Turbofan: A gas-path/nozzle method for in-flight thrust determination of a separate-stream turbofan is illustrated in Figure 4.7a and b.

Primary-stream thrust and airflow are determined using nozzle-entry pressure and temperature measurements and nozzle thrust and flow coefficients. The nozzle thrust coefficient is based on scale-model coefficients adjusted for engine measurement sampling errors and other effects, determined by analysis of full-scale test data. The nozzle flow coefficient is based upon data from the ATF engine test. The primary coefficients are the same as those used during the ATF calibration to determine fan coefficients, except for adjustments to account for free-stream suppression.

Fan thrust and airflow are determined using fan-nozzle pressure and temperature measurements and nozzle thrust and flow coefficients determined from the ATF test. The fan coefficients are adjusted for the effect of free-stream suppression. Ram drag is calculated from inlet total-airflow determined by adding the fan- and primary-nozzle gas flows and adjusting for fuel flow and air bleeds.

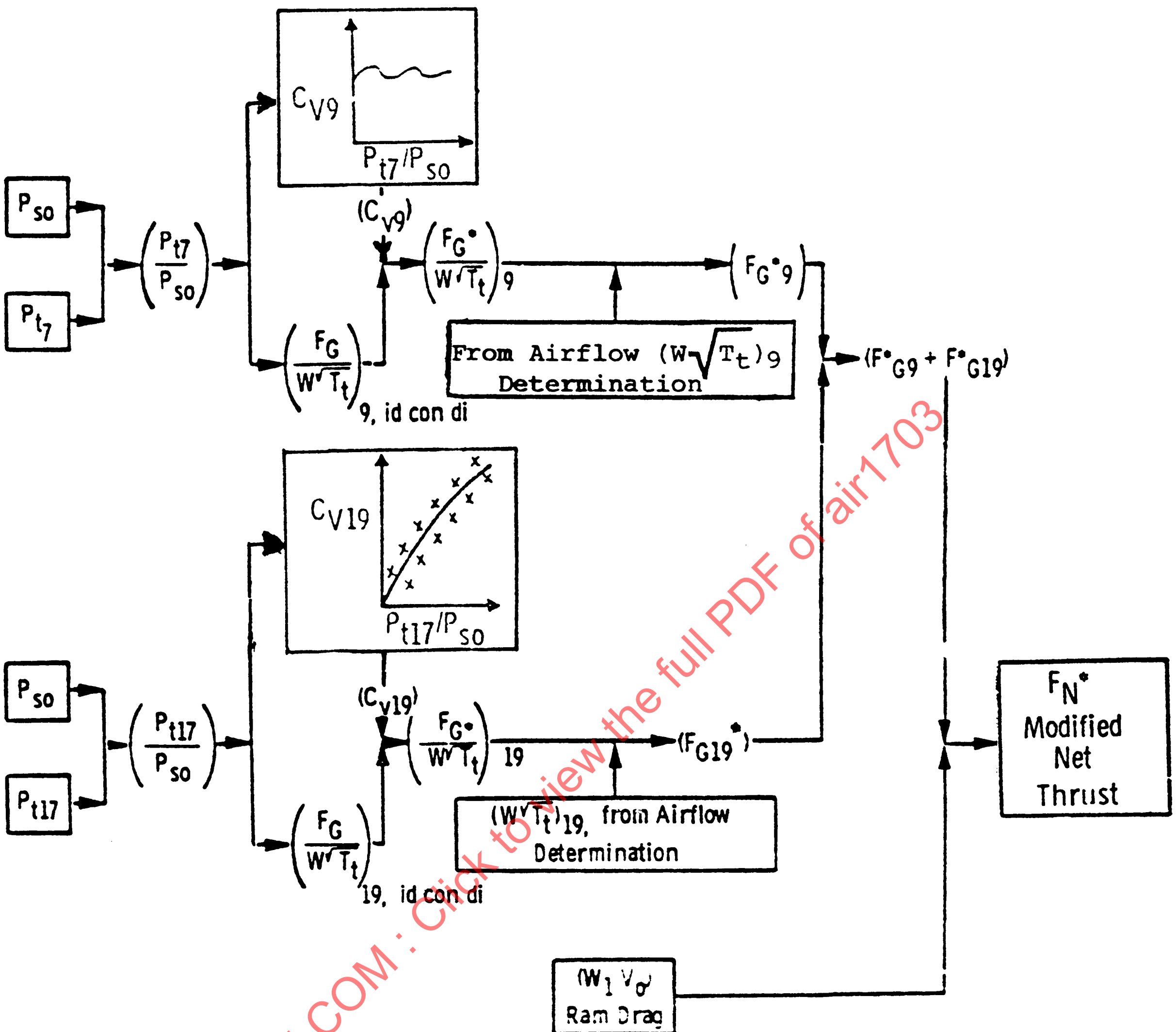
The method uses the same measurements and calculated airflow for both gross thrust and ram drag. This has the advantage of lessening the error in net thrust due to any error in airflow. The choice of preferred in-flight thrust methods should be based upon the results of an uncertainty analysis, as discussed in AIR 1678.

4.5.4 Mixed-Flow Afterburning Turbofan: A gas-path/nozzle method for in-flight thrust determination of a mixed-flow afterburning turbofan is illustrated in Figure 4.8. Two alternative options are presented: a F/AP method and a $F/W\sqrt{T}$ method. In the first option, gross thrust is obtained using a C_G nozzle coefficient that is obtained from correlations measured in the ATF and represented as functions of nozzle pressure ratio, P_{t7}/P_{s0}, and throat area, A_8. Nozzle inlet pressure, P_{t7}, is determined by measurement of turbine discharge pressure, P_{t6}, adjusted by calculation for the pressure loss in the afterburner. Nozzle throat area must be measured in flight.



a. Flight Airflow Evaluation

Figure 4.7 - Gas-Path/Nozzle Method for Separate-Stream Turbofan



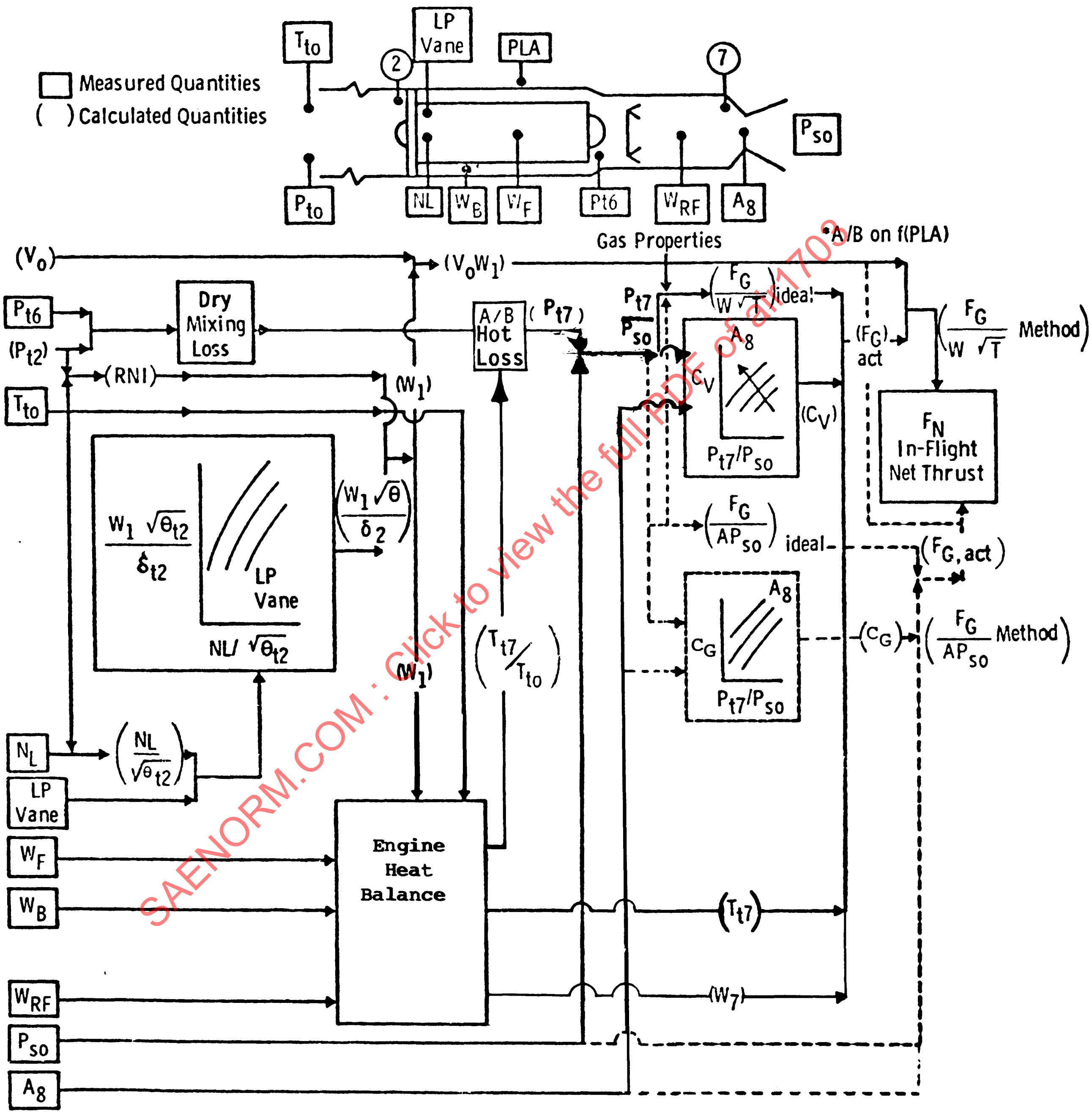
b. Flight Thrust Evaluation

Figure 4.7 (Cont'd) - Gas-Path/Nozzle Method for Separate-Stream Turbofan

4.5.4 <u>Mixed-Flow Afterburning Turbofan</u> (Cont'd.):

In the second option, specific gross thrust is obtained using a C_V nozzle coefficient that is obtained from correlations measured in the ATF and represented as functions of nozzle pressure ratio and throat area. Nozzle inlet temperature is obtained from a heat balance calculation based upon engine inlet temperature, inlet airflow, bleed flow, and core and afterburner fuel flows.

For both options, total airflow is obtained from a correlation of fan referred airflow versus referred LP-rotor speed with adjustment for Reynolds number.



F/AP$_0$ Method Differences Shown in Dashed Lines

Figure 4.8 - Gas-Path Nozzle Method (F/W $\sqrt{T}$) and F/AP$_0$) for Mixed-Flow Afterburning Turbofan

5. CALIBRATION TECHNIQUES:

The purpose, objectives and techniques of full-scale engine tests and scale-model nozzle, afterbody and inlet tests in ground facilities are discussed in their relationship to determining in-flight thrust.

Full-scale engine static tests (no external airflow) in ground facilities and scale-model nozzle tests are used to obtain calibration data for correlating the modified gross thrust and airflow characteristics that are necessary to determine in-flight modified net thrust. These test techniques are described in Sections 5.1 and 5.2.

Aircraft scale-model tests are utilized to determine the remaining propulsion system installation-related forces which complete the linkage between uninstalled and installed thrust in a manner consistent with thrust/drag accounting systems. These test techniques, which account for external-flow effects with changing engine airflow and nozzle area and pressure ratio, are described in Sections 5.3 and 5.4.

Methods described in Sections 5.3 and 5.4 assume that the inlet and nozzle effects are independent, and for many applications, this is acceptable. For cases where the inlet and exhaust effects may be coupled, a propulsion simulator, which can simulate simultaneously airflow and nozzle area and pressure ratio, may be beneficial. This technique is described in Section 5.5.

5.1 Engine Ground Testing:

5.1.1 Objectives and General Requirements: Most in-flight methods of deriving net thrust require some means of determining gross thrust and airflow for calculation of ram drag. The primary objective of calibrating the flight engine(s) in a ground facility is to correlate airflow and thrust with the parameters that can be measured in flight. Correlation data may be based on the calibration of flight engines or similar hardware. Compressor flow maps, turbine nozzle flow parameters, and calibration of other internal stations for determining the engine mass flow and fan-to-core flow split are derived generally from component rig or full-scale engine tests. Exhaust nozzle coefficients may also be derived early in the engine development cycle by a combination of full-scale and model-nozzle tests. Depending on the selected in-flight thrust method requirements, the flight engine calibration would be structured to:

- o Calibrate the existing engine performance model to the flight test hardware
- o Determine the effects of inlet-pressure, inlet-temperature and engine-pressure-ratio conditions on engine performance
- o Generate or confirm the exhaust nozzle coefficients
- o Determine the installation effects of compressor bleed, power extraction and inlet distortion on performance

5.1.1 Objectives and General Requirements (Cont'd.):

- Determine engine performance at specific simulated altitude and flight conditions.

Several versions of Ground Level Test Beds (GLTB) and Altitude Test Facilities (ATF) are available. The following are typical examples:

- Open Air Test Stand
- Ground Level Test Cell
- Altitude Test Cell
- Other Ram Facilities.

Ground facilities are used extensively during the engine development cycle for endurance testing, troubleshooting, production and overhaul acceptance, and evaluation of subsequent engine hardware improvements. When equipped with airflow and thrust measuring capabilities, these facilities may be used for calibrating the flight engines or other engines with representative flight hardware. Of the four facilities mentioned above, the ATF offers the most flexibility since calibrations can be made at the simulated flight conditions that are planned for the in-flight thrust evaluations and the influence of inlet pressure and temperature on engine performance can be defined separately. However, economic considerations and, in some cases, limitations in ATF capability may dictate the use of a GLTB alone.

The type of ground facility and instrumentation used in the calibrations are greatly influenced by the in-flight evaluation option selected, by the accuracy desired, and on the type of engine. For example, for a turbojet engine with choked nozzle, the in-flight net thrust can be determined from relationships of engine rotor speed to airflow and of exhaust nozzle pressure ratio to corrected gross thrust. Similar results may be obtained by deriving the nozzle flow and thrust coefficients. In either case the desired correlation data may be obtained in an open air test stand or a ground level test cell. Since the exhaust nozzle is choked, the profile(s) at the nozzle entry plane will vary only with power setting along the engine operating line experienced in an open air or ground level test stand. In the case where the nozzle(s) are unchoked, the profile(s) at the nozzle entry plane will vary with the multiple engine operating lines as well as the power setting. The error associated with extrapolating the sea level data to higher flight speeds may be relatively small for the choked nozzle. However, for an afterburning turbofan or a high-bypass-ratio turbofan, the calibration may have to be conducted in the ATF, if the accuracy of the simple turbojet is to be achieved. In most engine test facilities, the afterbody flow field cannot be simulated; and the wind-on effects on nozzle performance have to be determined from the wind tunnel tests, either full scale or model.

In the thrust calibration procedures of this report, most flow variables have been expressed as properly averaged one-dimensional parameters. A well designed test calibration process should provide the one-dimensional averages in two general ways:

5.1.1 Objectives and General Requirements (Cont'd.):

- Design the test facility installation configuration so that pressure and temperature gradients are minimized at the control volume boundaries.

- Make sufficient measurements so that a proper average can be determined when gradients may exist.

The application of the AIR 1678 uncertainty methodology recommends quantitative estimates of how well the average flow variables can be determined by a specific instrumentation/measurement system in a particular test facility. Validity checks to identify unanticipated errors in the in-flight thrust determination process are discussed in AIR 1703, Section 7.

Section 6 provides data measurement guidelines that apply to ground facilities and to in-flight measurements. It is recommended that identical pressure and temperature probes and position sensors be used for both the ground calibration and the in-flight thrust evaluation. The environment around the data acquisition components should be monitored and, where practical, should be controlled. Even though the current data acquisition systems are sophisticated, the need for frequent calibrations, some duplication of measurements, uncertainty analysis, and continuous attention to details cannot be overemphasized. Reference 5.1 is a comprehensive guide for the gas turbine measurements and the uncertainty analysis related to the ground facilities.

Ground calibration data are generally acquired at stabilized engine conditions. If, however, the selected in-flight method employs aircraft maneuvers, the flight engine calibration may include a definition of engine performance in the semi-transient environment. Engine parameters that are usually measured include: airflow, thrust, compressor bleed flow, power extraction, inlet pressure and temperature, turbine discharge pressure and temperature, rotor speed(s), fuel flow, ambient pressure and many other potential correlating parameters. If the exhaust-nozzle and the inlet geometries are variable, these too must be measured.

In open air and ground level test stands, an engine calibration can be accomplished relatively quickly, since no facility changes are required between power points. However, when highest quality data are desired, it is good practice to repeat the calibration on a different day. This will provide some insight into the engine operating repeatability (trim limits) and added assurance that the data are valid. ATF calibrations take more time because the engine environment is controlled by a complex facility. Due to the high cost of operating an ATF, sophisticated real-time data reduction systems are used to cross-check the data with predictions. This is done to minimize the need for rerunning tests.

5.1.2 Ground Level Test Beds:

5.1.2.1 Open Air Stand: An open air test stand can provide for essentially a direct measurement of gross thrust. The instrumentation environment remains constant during instrument calibrations and engine testing. Since the calibration data are obtained only at ground level static conditions, the effects of altitude and Mach number on engine performance have to be determined from other sources. The wind velocity/direction and humidity are important considerations. The influence of crystal or liquid water ingestion on engine performance is difficult to assess, therefore performance calibrations should not be run during any type of precipitation.

In an open air test stand, the engine, along with the calibrated bellmouth (flow nozzle) and the supporting structure, is supported by flexures, either from an overhead structure or from a ground platform (Figure 5.1). Axial force (thrust) on the structure is measured by the strain-gage load cell. To prevent hot gas reingestion, some installations have a pivoting mechanism which allows the engine to be pointed into the wind. When the wind speed is zero, the test stand measured force, the engine gross thrust and the net thrust are equal. If engine testing is done at other than zero wind conditions, correction to the measured force may be necessary. For example, the ram drag correction for a 10-knot headwind is equal to about 0.5 lbs per lb/sec of engine airflow. It is accepted practice to test in headwinds but to limit the calibrations to test periods when wind velocity is less than 10 knots and steady.

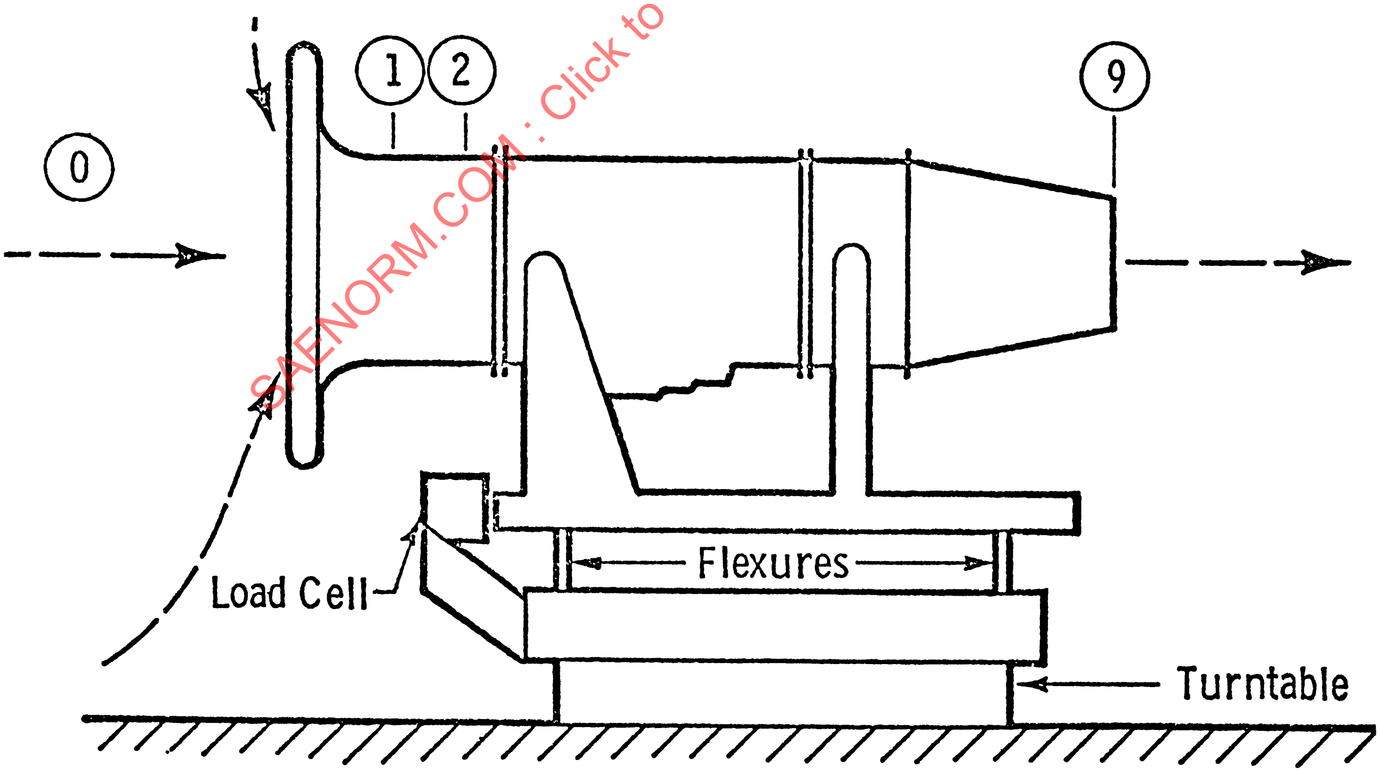


Figure 5.1 - Open-Air Test Stand

5.1.2.2 Ground Level Test Cell: Engine installation and thrust and airflow measurements in the GLTB (Figure 5.2) are similar to the open air test stand except that the gross thrust measurement is more involved, because the physical boundaries of the enclosure may induce additional forces on the engine and the attendant structures. These forces have to be derived either by correlating the test cell measurements to an engine whose performance is well defined(5.2) or by calculations based upon measured velocities and pressures. The magnitude of the correction force is unique to the engine model/installation.

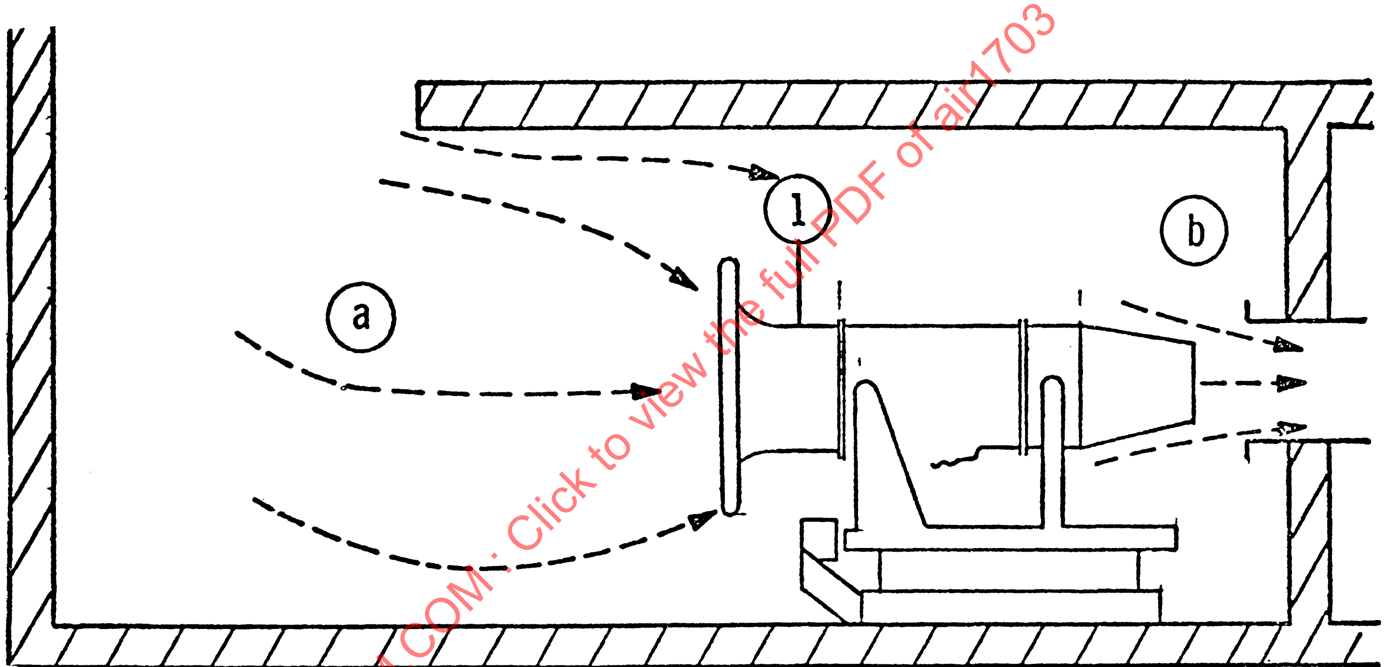


Figure 5.2 - Ground Level Test Cell

Total airflow through the test cell is the sum of engine airflow and secondary flow induced by the exhaust plume entrainment. For a given test cell cross-sectional area, the thrust correction forces increase with an increase in the total cell airflow. Generally, the free stream momentum is the dominant factor. The scrubbing force and pressure area force are directly related to the secondary airflow. Some test cells have a front bulkhead. In this arrangement, the scrubbing forces are negligible, and the total airflow through the forward part of the test cell is reduced. There may be an attendant increase in pressure difference from the front to the back of the test cell.

5.1.2.2 (Cont'd.)

In the GLTB, the gross thrust for a single stream engine is related to the test stand measured force by the following expression:

$$F_{G9} = F_M + W_0V_0 + \phi_B + \phi_D \tag{5.1}$$

where, F_M = measured net test stand force

W_0V_0 = freestream momentum

ϕ_B = A_1 (P_{sa} - P_{sb}) = pressure area or buoyancy force

ϕ_D = scrubbing force on the engine and supports.

The test cell thrust correction factors can be derived by: (1) correlation with calibrated test stand, (2) correlation with a calibrated exhaust nozzle, (3) semi-empirical calculations based on test cell measurements, or (4) the use of a thrust measuring bellmouth plus area and pressure measurements. The first two methods provide only the total correction factor. It is good practice to use a second method to check the cell corrections. This process provides added confidence to the experimentally derived correction factor and provides a technical base for future test cell improvements. For the test cell depicted in Figure 5.2, typical force values that contribute to the overall thrust correction factor are:

- o W_0V_0 less than 5 percent of F_{G9}
- o ϕ_B less than 2 percent of F_{G9}
- o ϕ_D less than 0.5 percent of F_{G9}

Some of the older test cells which were designed for low airflow engines may not be suitable for calibrating the current large turbofan engines.

5.1.3 <u>Altitude Test Facility</u>: Gas turbine engine component performance and rotor speed matching is influenced by the altitude and the flight speed, but engine calibration in the GLTB does not define these effects. The major advantage of the ATF is that the engine can be tested over a wide simulated flight spectrum. Depending on ATF capabilities, the exhaust nozzle coefficients can be determined over the complete nozzle-pressure-ratio operating range, and altitude and inlet condition effects on engine performance can be defined.

5.1.3 Altitude Test Facility (Cont'd.):

An ATF is a complex facility comprised of the air supply system, the altitude test cell, and the exhaust system. Figures 5.3 and 5.4 show typical engine installations in an altitude test cell. The air supply system includes air compressors, temperature control equipment, and pressure/flow control valves. Conditioned air is supplied to the engine through a flow measuring device or a duct section in which the flow is measured.

The inlet duct contains a seal (slip joint) to separate the fixed structure from the free-floating test bed. Engine exhaust flows through an ejector or a gas collector, a gas cooler, flow control valves, and the compressors that discharge the exhaust gases to atmosphere.[5.3]

The cooling/scavenging airflow to the altitude cell is controlled and measured. Leakage through the labyrinth seal is also a known quantity. This enables the scrubbing forces (carcass drag) to be defined as a function of total secondary airflow. Most ATFs have exhaust ejectors to augment exhauster pumping capability. For single exhaust engines, ejector sizing is relatively easy and good test cell scavenging is possible. For dual exhaust turbofan engines this task is more difficult, and additional scavenging air may be needed to eliminate or reduce the exhaust gas recirculation.

Isolation of the thrust balance system at the engine inlet is usually accomplished by means of a slip joint. Labyrinth and pressure balance seals are the most common. Airflow through the labyrinth seal may be in or out of the duct, depending on the static pressure differential across the seal. Airflow through the labyrinth seal is calculated using either an assumed discharge coefficient or a calibration. Some seals may contain an inflatable rubber boot which, when inflated, provides a positive seal. This arrangement, in conjunction with the upstream venturi, may be used to calibrate the airflow through the slip joint. Other slip joint designs may include a cavity in which the pressure can be closely monitored and controlled. When the cavity pressure is matched to the static pressure within the duct, this pressure balance seal prevents the air from flowing in or out of the main engine duct.

Figure 5.3 shows a dual exhaust turbofan engine installation featuring an overhead suspension and thrust measuring system. Airflow is measured with the venturis located upstream of the bellmouth. Generally, some flow straightening arrangement is required downstream of the venturis to provide a uniform flow profile at the bellmouth. Figure 5.4 shows a single exhaust engine installed in an ATF with under-engine thrust sled. Here, the airflow is calculated from the pressure and temperature profiles measured downstream of the slip joint at Station 1. As an alternative, a

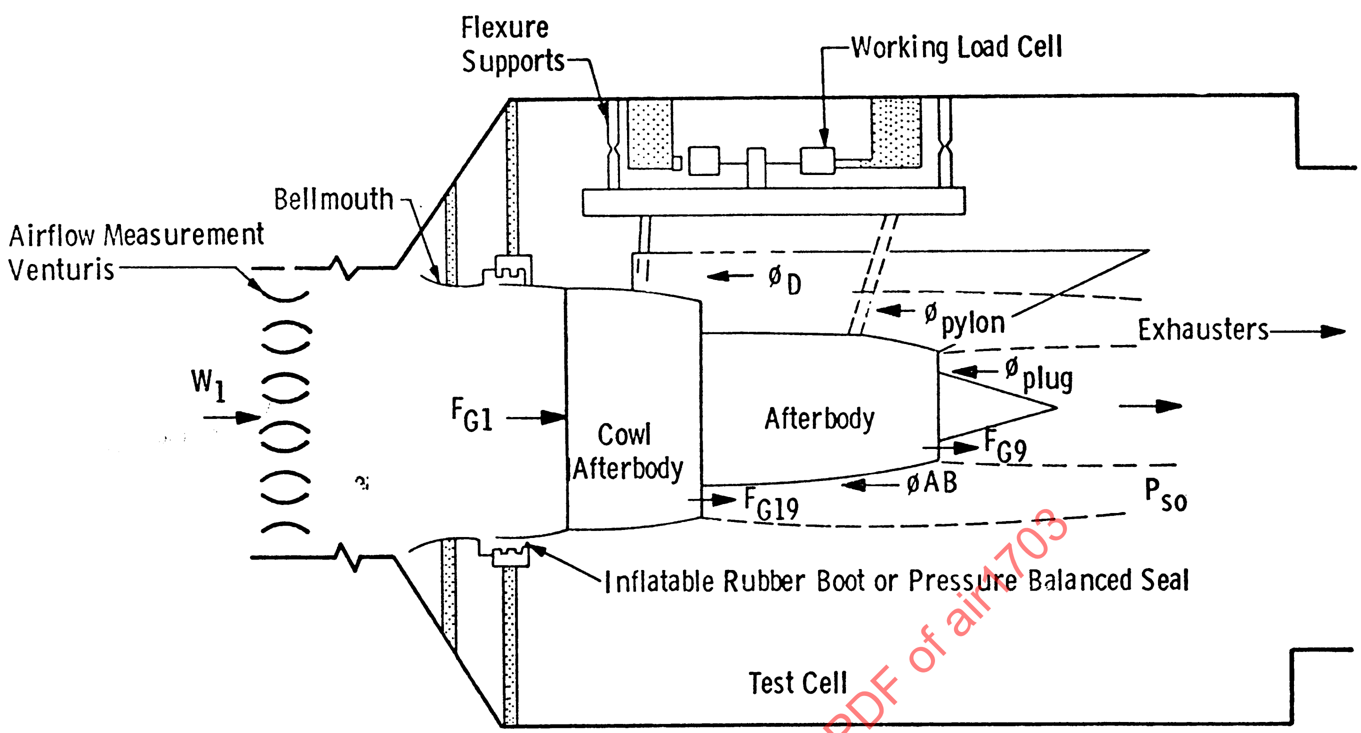


Figure 5.3 - Altitude Test Cell with Overhead Thrust Measuring Systems

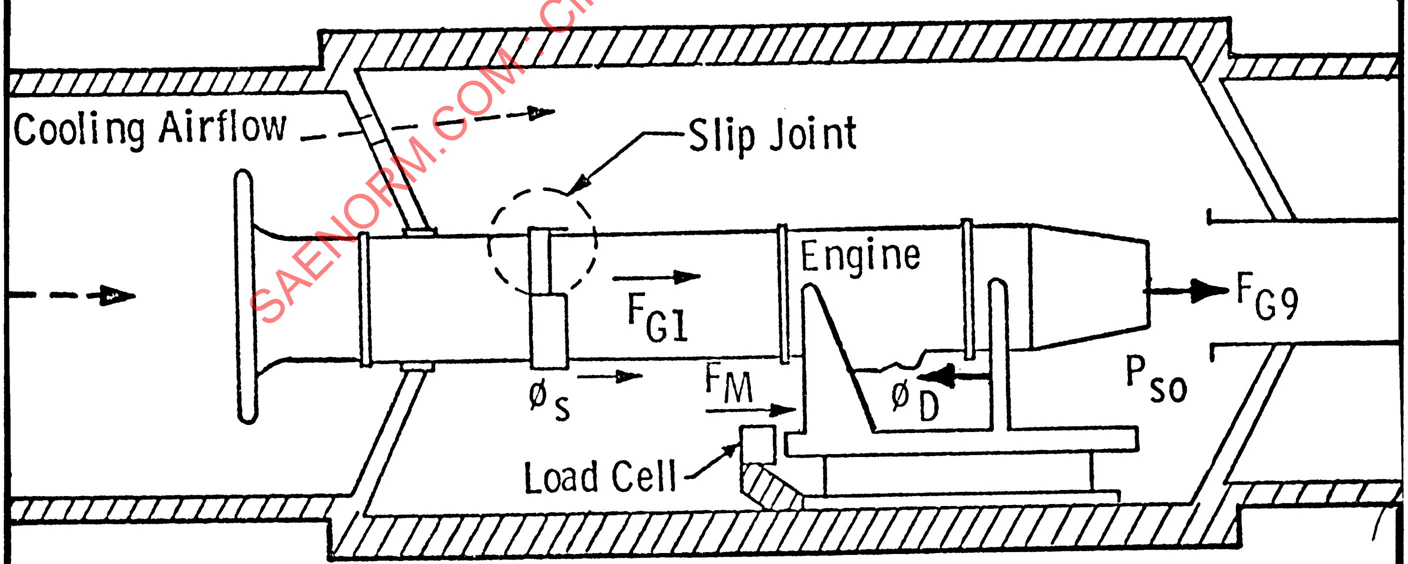


Figure 5.4 - Altitude Test Cell with Under-Engine Thrust Sled

5.1.3 <u>Altitude Test Facility</u> (Cont'd.):

calibrated bellmouth with some accounting for the slip joint leakage flow may be used. Additional information on thrust and airflow measuring systems is provided in Paragraph 5.1.5.2

In an ATF, the desired flight speed for a given altitude is simulated by controlling the engine inlet total pressure, P_{t2}, and temperature, T_{t2}, to the flight speed stagnation values. The aircraft inlet duct recovery and the inlet distortion may be considered in establishing the inlet flow condition. Inlet distortion is simulated by various density screens or airjet distortion generators(5.4, 5.5, 5.6) located in front of the engine to reproduce the flight inlet total pressure profile at the engine face at a specified airflow. The desired altitude pressure is set by the test cell ambient pressure, P_{s0}. The inlet and the altitude pressures are controlled by the ATF flow/pressure control valves. Some ATFs are equipped with fast-acting flow control valves, and the desired flight conditions can be maintained during engine transients.

In an ATF, the flight engine calibration may be a relatively limited performance check at the desired flight conditions or a more extensive evaluation, depending on the requirements of the selected in-flight option. Other factors that impact the scope of the calibration include: previous experience with the specific engine model, availability or capability of the ATF, resources and timing.

For an engine with single exhaust nozzle, Figure 5.4, the gross thrust is determined from the following relationship:

$$\begin{aligned} F_{G9} &= F_M + F_{G1} + \phi_D + \phi_S \\ &= F_M + W_1V_1 + A_1\,(P_{s1} - P_{s0}) + \phi_D + \phi_S \end{aligned} \tag{5.2}$$

where, W_1V_1 = inlet duct momentum

$A_1\,(P_{s1} - P_{s0})$ = inlet duct pressure-area force

ϕ_D = scrubbing force on engine skin and supports

ϕ_S = pressure-area and scrubbing forces at the slip joint

Generally, ϕ_D plus ϕ_S is less than 0.5 percent F_{G9}

Calculation of gross thrust for a dual exhaust engine, Figure 5.3, is similar, although additional pressure measurements along the core afterbody may be required. Relation of the gross thrust to the measured force is as follows:

$$F_{G19} + F_{G9} = F_M + W_1V_1 + A_1\,(P_{s1} - P_{s0}) + \phi_S - \phi_{AB} - \phi_{plug} - \phi_{pylon} + \phi_D \tag{5.3}$$

where, ϕ_{AB} = force on core afterbody

5.1.3 Altitude Test Facility (Cont'd.):

Measurement of the core afterbody, the pylon and the plug forces is difficult; therefore, it is more practical to utilize the modified fan and core exhaust gross thrusts (Section 3). The relation of the modified gross thrust to the test cell measured force can be reduced to the following terms:

$$F^*_{G19} + F^*_{G9} = F_M + W_1V_1 + A_1\,(P_{s1} - P_{s0}) + \phi_S + \phi_D \tag{5.4}$$

5.1.4 Other Ram Facilities: Test facilities that enable partial simulation of the flight environment are also available and may be used for calibrating the flight engines. These facilities are generally designed so that the air compressors can be used either as exhausters or blowers. In the former case, compressors are used to control engine exhaust or altitude static pressure while the engine inlet is open via a pressure valve to the prevailing ambient conditions. A significant portion of the flight envelope may be simulated; however, lack of inlet temperature control may be a limiting factor. When compressors are used for ramming, the engine exhaust is open to atmosphere. In this mode, inlet temperature is usually controlled, and simulation of a wide range of flight speeds at sea level is possible.

Figure 5.5 depicts a ground level test cell with inlet ram capability only. Here, the engine is supplied with air at controlled pressure and temperature, while the environment around the engine remains at prevailing ground level pressure. Engine installation, and thrust and airflow measuring systems and techniques are similar to the ATF shown in Figure 5.4. The major advantage of calibrating the flight engine(s) in a ram facility is that the data can be obtained at higher nozzle pressure ratios.

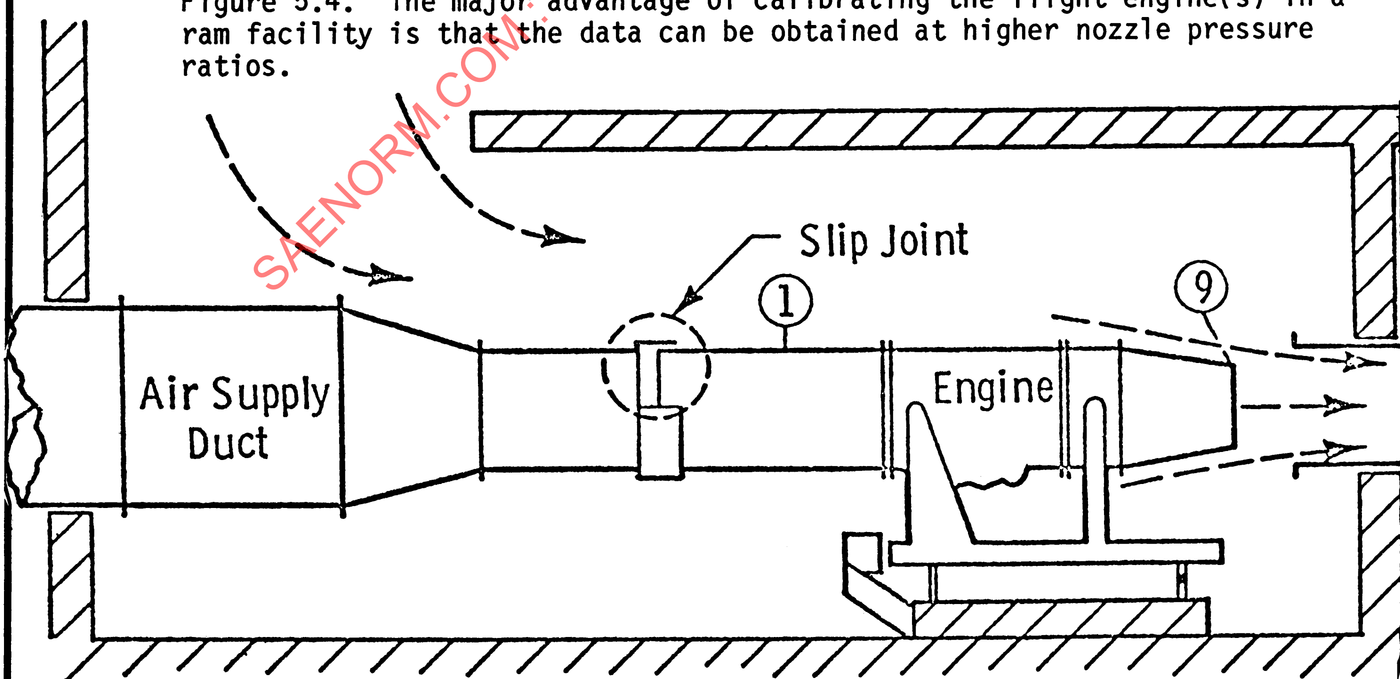


Figure 5.5 - Ground Level Test Cell with Inlet Ram Capability

5.1.5 Special Considerations:

5.1.5.1 Thrust Measurement: The basic elements of the thrust measuring systems are similar for all ground facilities. The test engine and at least a part of the inlet duct is supported by flexures, either from an overhead structure or from a platform. In most GLTB installations the inlet bellmouth is attached to the engine, while in most ATF or ram facilities the bellmouth is isolated from the force-balance system (thrust stand). The flexure arrangement restrains all but axial travel of the thrust stand. Axial force is usually measured by a strain-gage load cell. Hydraulic and pneumatic pressure capsules have been used in the past, but currently, load cells are more common.

Strain-gage load cells may be vented or hermetically sealed. Vented load cells are insensitive to ambient pressure variations. Temperature effects on a load cell can be eliminated by maintaining a constant load-cell temperature. The test stand axial travel needs to be greater than the total deflection of the load cell under full load to provide for thermal expansion and other factors. This is an important consideration for an ATF, where high test cell ambient temperature and pressure variations may occur.

Thrust stand calibrations are performed with the engine completely installed and fuel lines pressurized. To reduce hysteresis, all service and instrumentation lines should have a generous length and be perpendicular to the direction of the measured force. Typically, a hydraulic jack is used to apply a force through a calibration load cell (working standard) to the thrust bed and the test-cell load cell. Ideally, the calibration force should be applied through the engine thrust center. When the calibration force is not applied on the engine center-line, a correction to the calibration may be necessary. This correction can be determined by applying several different moments to the thrust stand during the calibration.

One of the major problems with test cell thrust measurement, especially in the ATF, is the difficulty in reproducing the thrust measuring system environment during calibrations and during the engine tests. Some facilities conduct thrust calibrations with the engine operating, while others may employ a fixed preload which can be applied during the engine test to spot-check the initial calibration. Tare forces must remain unchanged during the system calibration and with the engine operating. Reference 5.3 provides some typical measurement uncertainties.

5.1.5.2 Airflow Measurement: The two most common ways to measure airflow at the inlet of the gas turbine engine involve the use of a flow nozzle (bellmouth) or a critical-flow venturi. Flow nozzles are used extensively in the GLTB, while either the venturi or the flow nozzle is used in the ATF. The airflow is calculated either: (1) from pressure and temperature measurements and a flow coefficient or (2) from pressure profiles and temperature within the flow measuring device.

5.1.5.2 Airflow Measurement (Cont'd.):

Calculation of airflow through the critical-flow venturi requires minimum instrumentation, and the flow coefficient can be calculated with a high degree of confidence.[5.1] Disadvantages include the need for a flow straightening screen or grid to preclude inlet distortion and for accounting of the slip joint airflow. The pressure drop across the flow measuring system may be a limiting factor for a facility that has limited air supply capability.

A calibrated bellmouth requires little instrumentation for calculating airflow, but the system is sensitive to flow disturbances near the bellmouth. Airflow can also be measured downstream of the bellmouth throat provided that sufficient instrumentation is used to measure the duct velocity profiles, including the boundary layer. Locating the instrumented section downstream of the labyrinth seal eliminates the need for determining the leakage flow. The uncertainty band of calculated mass flow becomes larger at lower flow Mach number (less than 0.3).[5.7]

Compressor bleed flow, turbine cooling flow, anti-icing flow, etc., may be measured with a sonic nozzle, venturi flowmeter, or calibrated orifice.

5.1.5.3 Humidity Corrections: Engine performance may require humidity corrections. Gas properties of air are affected by specific humidity.[5.8] For a 90°F day with a relative humidity of 85 percent (specific humidity of 26.5 grams of water per kilogram of dry air), R, C_p and C_v are 1.5-, 2.2- and 2.5-percent higher, respectively, than for dry air. For an engine operating in this environment, the observed thrust would be about 0.4-percent lower and observed fuel flow about 1.3-percent higher than the values that would be attained if the engine were operating at the same rotor speed with dry air. Reference 5.9 provides data for humidity corrections.

5.1.5.4 Inlet Duct Condensation: Condensation of water vapor in the engine inlet duct during static tests may occur if ambient relative humidity is sufficiently high. This problem should be considered in the engine calibrations because condensation has a significant influence on engine performance. The gas properties are affected by the specific humidity, while the condensation problem is related to the relative humidity. Since the engine inlet duct velocity is relatively high (Mach 0.3 to 0.5), the stream static temperature and pressure are significantly lower than ambient (about 3 percent for absolute temperature). Thus, even air at a relative humidity of 58 percent has sufficient water content to saturate the air at the engine inlet. Higher ambient humidity may result in condensate formation. Whether it occurs or not depends on the inlet duct length, the perturbations that produce higher local velocities, and the amount of solid particles in the air stream. Condensation results in an increase in total temperature and a decrease in total pressure of the gas flow entering the engine. These effects are diffi-

5.1.5.4 Inlet Duct Condensation (Cont'd.):

cult to measure. Temperature and pressure changes on engine performance can be derived analytically, but the effects of liquid water ingestion have to be derived experimentally. The problem is further complicated by the fact that the liquid/vapor equilibrium conditions may not exist at the engine inlet, i.e., water vapor may be in the supersaturated state. Currently, the condensation effects on engine performance can only be determined by semi-empirical methods.[5.10, 5.11]

5.1.5.5 Inlet Screen: Ground test facilities have inlet screens to prevent debris from entering the engine and, in some cases, to reduce the adverse velocity gradients at the engine inlet. The inlet screen in the ATF is isolated from the thrust measuring system, while in the GLTB, it may be attached to the bellmouth. If the engine inlet pressure is measured downstream of the screen, the referred or corrected engine parameters are not affected by the screen pressure drop (for a choked exhaust nozzle). A ram pressure ratio correction is required to adjust the data to a desired reference condition.

5.1.6 Engine Calibration Examples: This section presents simplified examples of how ATF engine calibration data are used to derive nozzle coefficients and other correlating parameters for several in-flight options. Examples include:

o The single-exhaust turbofan

o The intermediate-cowl turbofan

o The mixed-flow afterburning turbofan.

These examples were introduced as examples of in-flight thrust methods in Paragraph 4.5.

For clarity, only the parameters that are essential to the specific in-flight option are shown in the illustrations. Many other parameters are generally measured to cross-check measured values and to verify control limits and schedules.

5.1.6.1 Single-Exhaust Turbofan: This example illustrates three alternate methods to determine in-flight thrust for a single-exhaust non-afterburning turbofan. All three methods correlate airflow using fan rotor speed, but use different gross thrust correlations.

The first method correlates nozzle gross thrust with nozzle pressure ratio. Figure 5.6a shows the schematic chart for the ATF calibration generating the in-flight correlating relationship. The fan and core discharge total pressures, P_{t16} and P_{t6}, and test cell ambient pressure, P_{s0}, are used to calculate the area-weighted nozzle pressure

5.1.6.1 Single-Exhaust Turbofan (Cont'd.):

ratio, NPRA. Engine gross thrust is determined from Equation 5.2. For this example, the engine and labyrinth seal scrubbing forces were assumed to be negligible, and for expediency, the pressure-area force at the labyrinth seal was included in the term by adjustment to A_1. The airflow, the inlet duct momentum and the pressure-area forces are calculated from the inlet duct total pressure profile, static pressure and temperature. Station 2 measurements are the reference conditions.

Corrected gross thrust versus NPRA as a function of ram pressure ratio, P_{t2}/P_{s0}, and corrected airflow versus corrected engine rotor speed are the outputs used from the ATF calibration.

For the second method, nozzle gross thrust, referenced to Station 2 conditions, is determined in exactly the same manner described above. It is correlated versus corrected fuel flow rather than nozzle pressure ratio. Figure 5.6b shows the schematic chart for the ATF calibration generating the desired relationships using this method. The exponent used in the fuel flow temperature correction can be estimated analytically using the cycle match simulation and verified experimentally in the ATF and later in the aircraft.

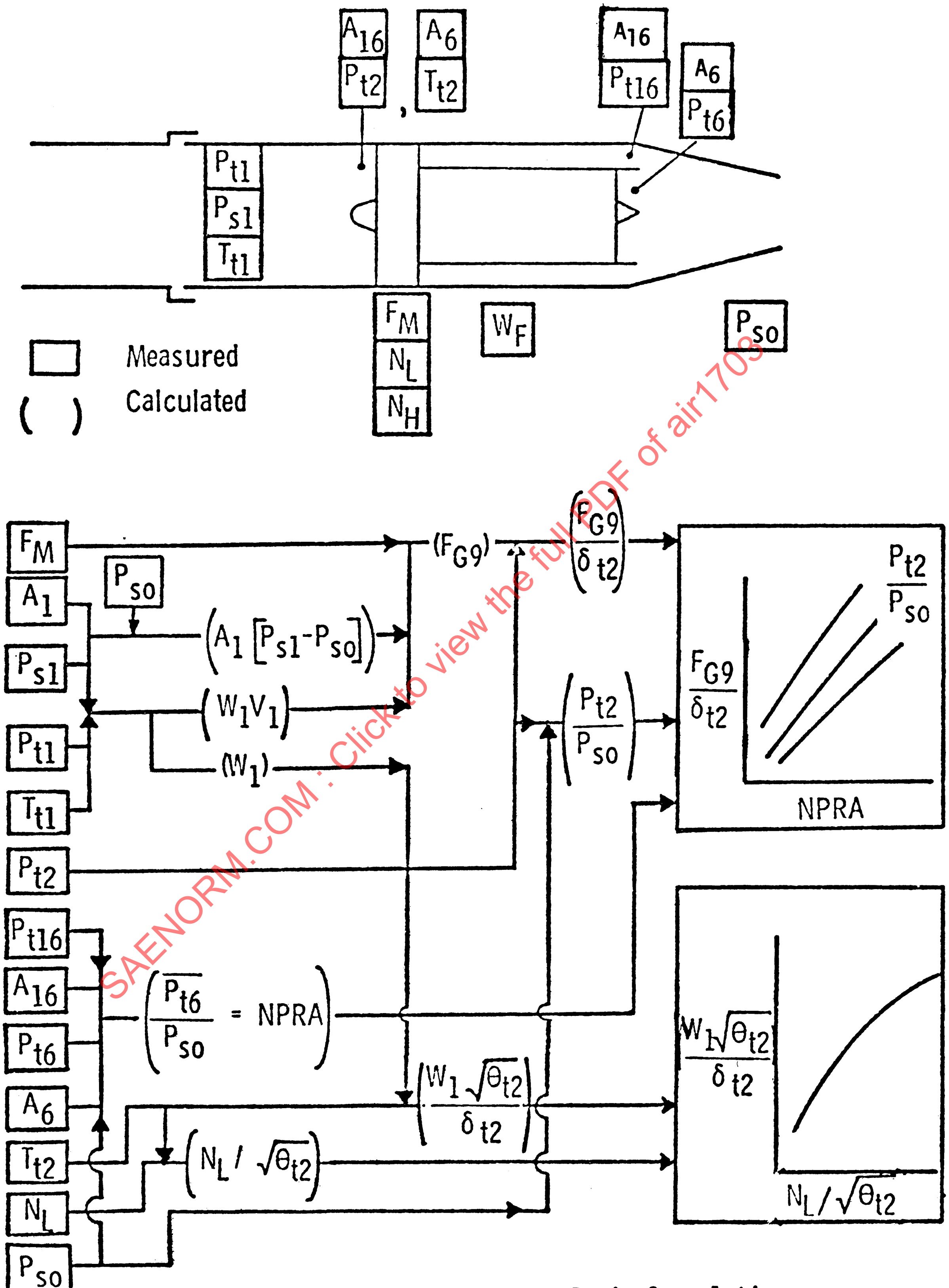
Corrected gross thrust versus corrected fuel flow as a function of engine pressure ratio and corrected airflow versus corrected rotor speed are the outputs used from the ATF calibration.

A third alternative method makes use of a gross thrust parameter correlation versus corrected HP rotor speed. The gross thrust parameter (Equation 5.5) tends to normalize to a single line versus any gas generator parameter for the case of choked thrust nozzles.

$$GTP = \left[\frac{F_{G9}}{A_8 P_{s0}} + 1 \right] \frac{P_{s0}}{P_{t2}} = (F_{G9} + A_8 P_{s0})/P_{t2} A_8 \qquad (5.5)$$

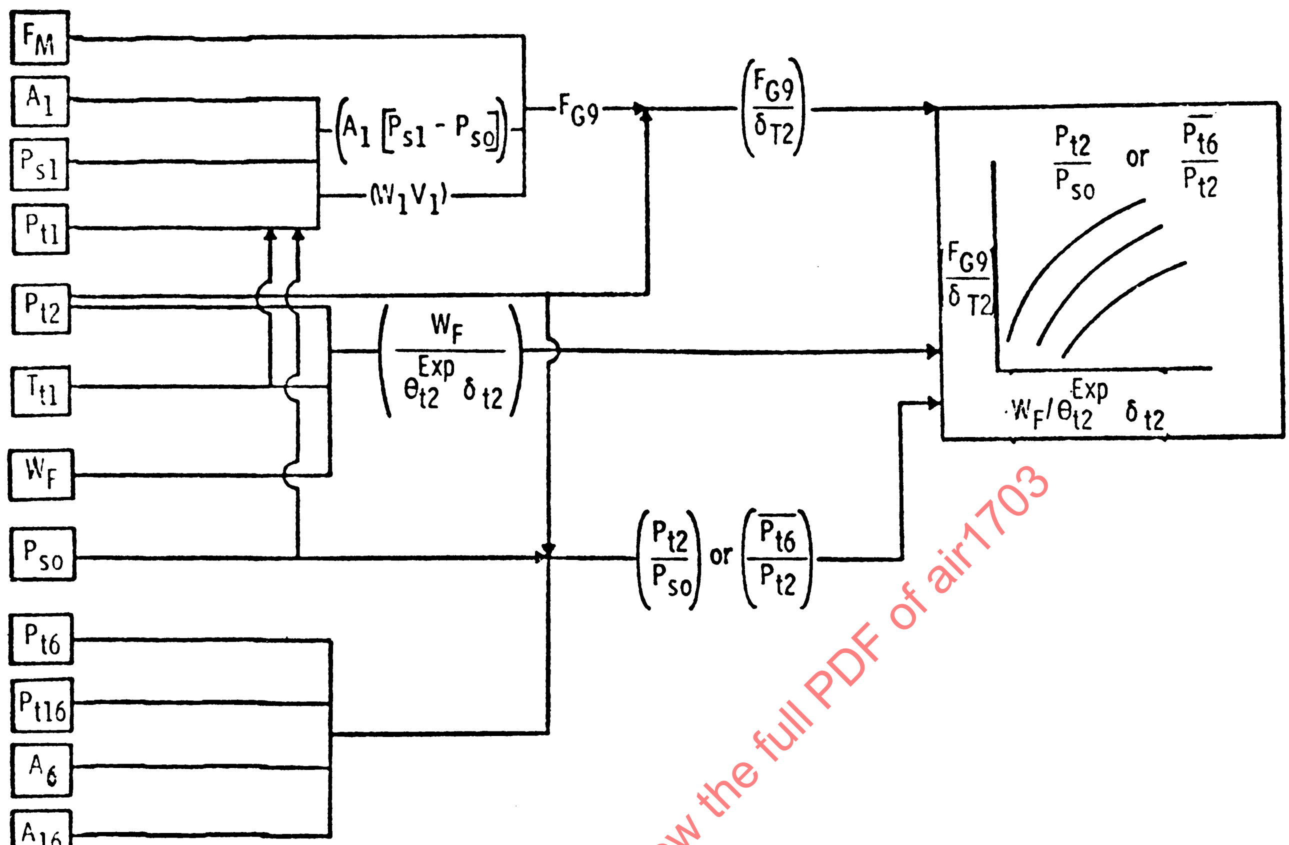
Figure 5-6c shows the schematic chart for the ATF calibrations generating the desired relationships using this method. Gross thrust parameter versus corrected HP rotor speed and corrected airflow versus corrected LP rotor speed are the outputs used from the ATF calibration.

5.1.6.2 Intermediate-Cowl Turbofan: An example of an ATF calibration setup is shown in Figure 5.7. Overall wind-off (quiescent air) thrust is determined from load cell measurements and calculated stream thrust, F_{G1}, at engine entry after correcting for tare forces. The engine is operating at simulated aircraft flight conditions set by P_{t2}, T_{t2}, and P_{s0} for various fixed-throttle power settings.

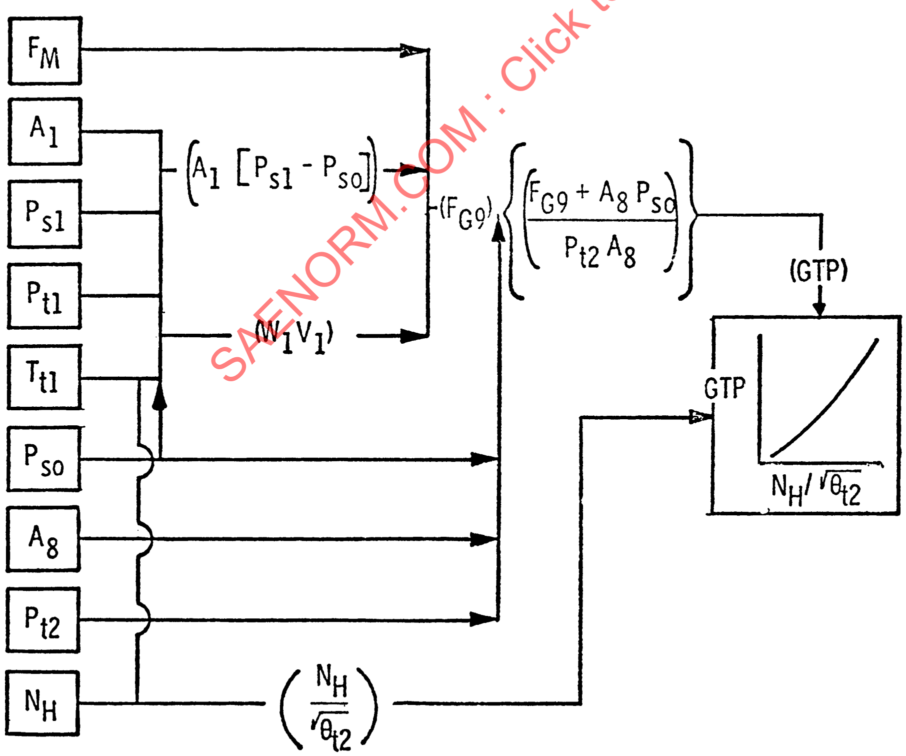


a. Gross Thrust/Nozzle Pressure Ratio Correlation

Figure 5.6 - Single-Exhaust Turbofan ATF Calibration



b. Gross Thrust/Fuel Flow Correlation



c. Gross Thrust Parameter Correlation

Figure 5.6 (Cont'd) - Single-Exhaust Turbofan ATF Calibration

5.1.6.2 Intermediate-Cowl Turbofan (Cont'd.):

Figure 5.7 shows the ATF calibration flow chart for generating the nozzle flow and the thrust coefficients. In this example, the actual mass flow used to define the primary nozzle flow coefficient is determined from the choked turbine nozzle parameter. The scale-model primary nozzle specific-thrust coefficient and the flow coefficient are used to calculate primary-stream thrust. The fan-stream thrust and mass flow are equal to the difference between the total measured values and the core stream values derived above.

Figure 5.7 shows a method option based on the modified thrust definitions. It is assumed that the exhaust NPR plus specific thrust coefficient option will be used with an ideal convergent-divergent $F/W\sqrt{T}$ thrust group reference for both primary and secondary nozzles. The nozzle discharge coefficient is used to account for wind-on/-off (flow suppression) effects, which are derived from scale-model or full-scale tests (see Paragraph 5.2.1.1). The calibration outputs are the primary and secondary nozzle coefficients versus the respective nozzle pressure ratio, as shown in Figure 5.7.

5.1.6.3 Mixed-Flow Afterburning Turbofan: A general arrangement sketch of the afterburning turbofan engine is shown in Figure 5.8. Core engine and bypass flows are partially mixed. Some bypass air is utilized to cool the afterburner liner. The final convergent-divergent nozzle is assumed to be fully variable between dry and maximum afterburner settings.

Figure 5.8 shows the engine parameters that are measured in the ATF calibration of the low-bypass-ratio afterburning turbofan engine. With the exception of airflow, thrust, engine discharge temperature, and nozzle entry pressure, all parameters are also measured in flight. Figure 5.8 is an abbreviated schematic diagram that shows how the in-flight correlation parameters are derived from the ATF engine calibration. Thrust coefficients are generated for use with the two method options: the F/AP method and the $F/W\sqrt{T}$ method.

The fan rotor speed-to-airflow relationship with accounting for the variable vane position, is used in both options to calculate the in-flight ram drag.

Correlation of the nozzle entry pressure to the turbine discharge pressure is used to calibrate the pressure loss calculation for the dry and afterburning pressure losses. The dry mixing loss is a function of engine pressure ratio, and the afterburning pressure drop is primarily a function of afterburner temperature rise.

The nozzle entry temperature is calculated from the engine and afterburner heat balance. Measured turbine discharge temperature is used for engine monitoring and to cross-check the engine heat balance results. Real gas properties are used to calculate the ideal nozzle thrust and flow parameters.

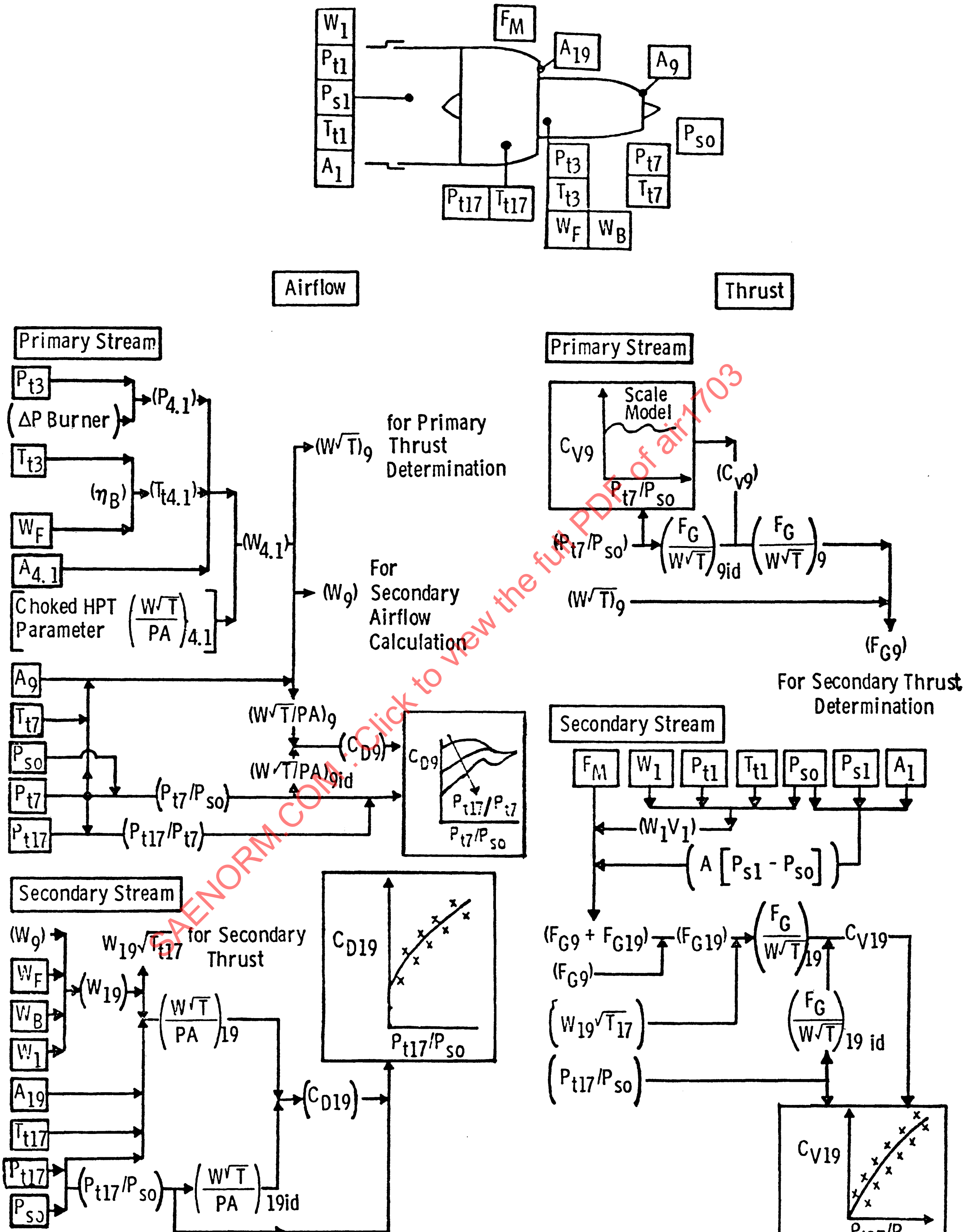


Figure 5.7 - Intermediate - Cowl Turbofan ATF Calibration

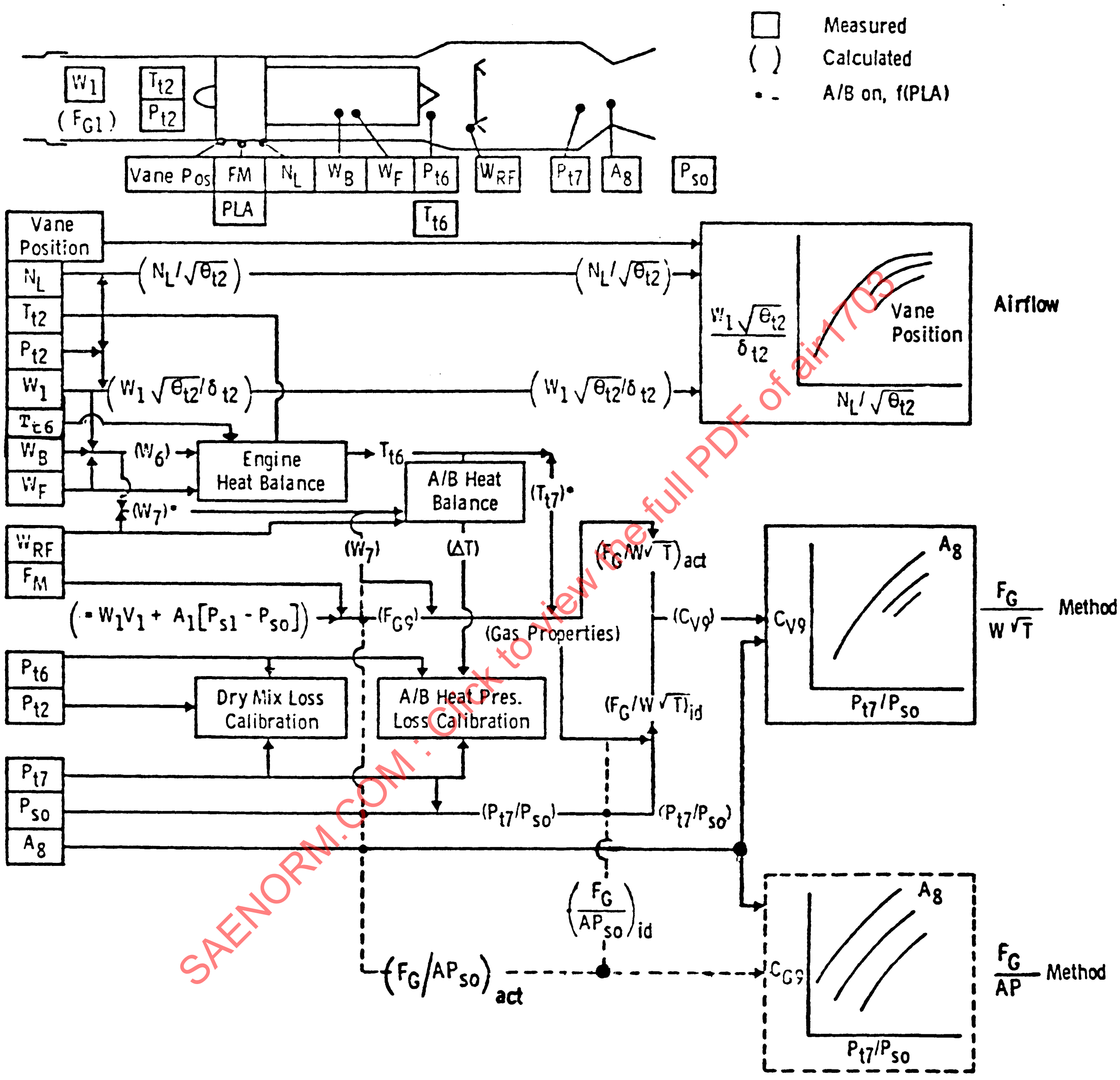


Figure 5.8 - ATF Calibration of Mixed-Flow Afterburning Turbofan

5.1.6.3 Mixed-Flow Afterburning Turbofan (Cont'd.):

The ATF calibration outputs are:

- o Airflow as a function of fan rotor speed and inlet-guide-vane position
- o Nozzle entry pressure as a function of turbine discharge pressure for dry and afterburning modes
- o Nozzle inlet total temperature as a function of fuel flows and airflow
- o Nozzle thrust coefficients as a function of nozzle pressure ratio and measured nozzle area.

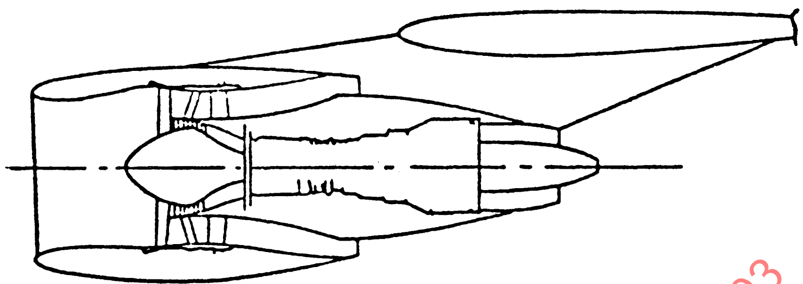
5.2 Scale-Model Nozzle Testing: Scale-model nozzle tests are conducted to develop the nozzle coefficients needed to characterize the engine and exhaust system performance. Model test results, when used in conjunction with GLTB or ATF full-scale engine test results provide the data for:

- o Determination of the magnitude and variation of nozzle performance with nozzle pressure ratio
- o Extrapolation of GLTB nozzle coefficients to higher nozzle pressure ratios which occur in flight
- o Adjustment of GLTB and ATF nozzle coefficients for external flow effects (wind-on/wind-off) which occur with unchoked nozzles.

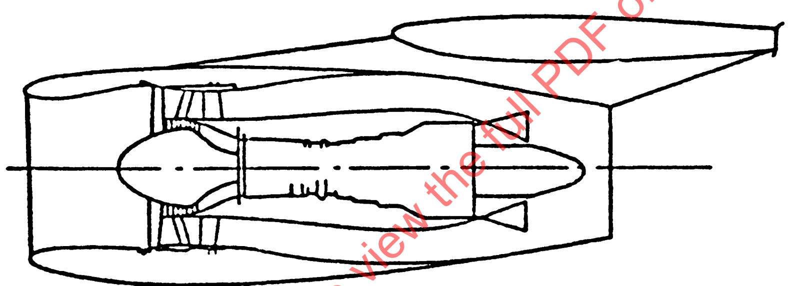
5.2.1 Nozzle Performance Characterization: As discussed in Paragraph 2.5, nozzle coefficients are defined to relate actual to ideal nozzle performance in a manner that is practical for in-flight thrust determination. The coefficients are characterized primarily as functions of nozzle pressure ratio and, for variable area systems, nozzle area. They define internal nozzle performance and are used to calculate nozzle gross thrust. Corrections for external force increments may be required to obtain the installed propulsive force, Equation 2.13.

Separate flow systems require that core and fan nozzle coefficients and the effect of fan flow (core nozzle suppression) be evaluated. Single-nozzle, mixed-flow systems require accounting for both pressure and temperature differences in the two streams. Variable-area convergent/divergent nozzles such as used for supersonic, multi-mission aircraft, require characterization over a broad range of areas and pressure ratios for either turbojets or mixed-flow turbofans.

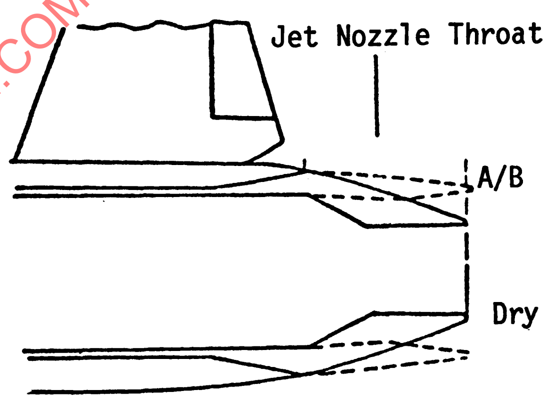
These three exhaust nozzle configurations are shown in Figure 5.9. Examples of these nozzle characterizations and model test requirements are discussed further.



a. Separate Flow



b. Long Duct Mixed Flow



c. Variable Area Convergent/Divergent

Figure 5.9 - Exhaust System Configurations

5.2.1.1 Separate-Flow Systems: Four nozzle coefficient curves, illustrated in Figure 5.10 are required to define the exhaust system performance at static conditions. The core-nozzle specific thrust coefficient curve is established by operating the model with zero fan flow. That test also establishes the upper line on the core-nozzle flow coefficient plot. With both the fan and core streams flowing, the variation of core-nozzle flow coefficient with fan-nozzle pressure ratio is established. That variation is due to the interaction between the nozzle exhaust flows in the vicinity of the core-nozzle exit plane. The interaction between the two flows imposes a modified static pressure on the core nozzle that determines the amount of flow passed through the nozzle. Since the ideal flow remains the same at a fixed core-nozzle pressure ratio, the flow coefficient is apparently changed.

The fan-nozzle flow coefficient is obtained directly from the dual-flow test results. The fan-nozzle specific thrust coefficient is based on a fan stream thrust which is determined by subtracting the calculated value of core-nozzle thrust from the total thrust measured during dual flow operation. Any force contribution due to the interaction between the two flows in the vicinity of the core nozzle is automatically included in the fan-nozzle specific thrust coefficient.

External flow can affect the nozzle flow coefficients and hence the predicted in-flight net thrust. The interaction between the external free stream flow and the engine exhaust flow produces a pressure at the nozzle lip that may be different than the corresponding local static pressure during static testing. In-flight thrust calculation procedures should account for this free steam suppression of the nozzle flow coefficients. The flow coefficient suppression effect may be characterized by accounting the incremental flow coefficient change, relative to the wind-off value, as functions of free-stream Mach number and nozzle pressure ratio, (see Figure 5.11). The degree of suppression is extremely configuration and installation dependent, and may also vary with the wing flap setting and angle of attack.

It is assumed that flow suppression only affects the quantity of flow that is passed by the nozzle and not the nozzle efficiency. Therefore, a change in nozzle specific thrust due to free stream suppression would be bookkept in aircraft drag.

An alternate method used to account for fan suppression of core nozzle coefficients and external-flow suppression of fan nozzle coefficients is based on applied nozzle pressure ratio. Nozzle base static pressures, P_{s9} and P_{s19}, are substituted for free-stream ambient pressure in the nozzle pressure ratio used to characterize the coefficients. This technique may preclude the need for wind-on model testing. Use of this nozzle performance characterization for flight test evaluations would require the measurement of the nozzle base pressures.

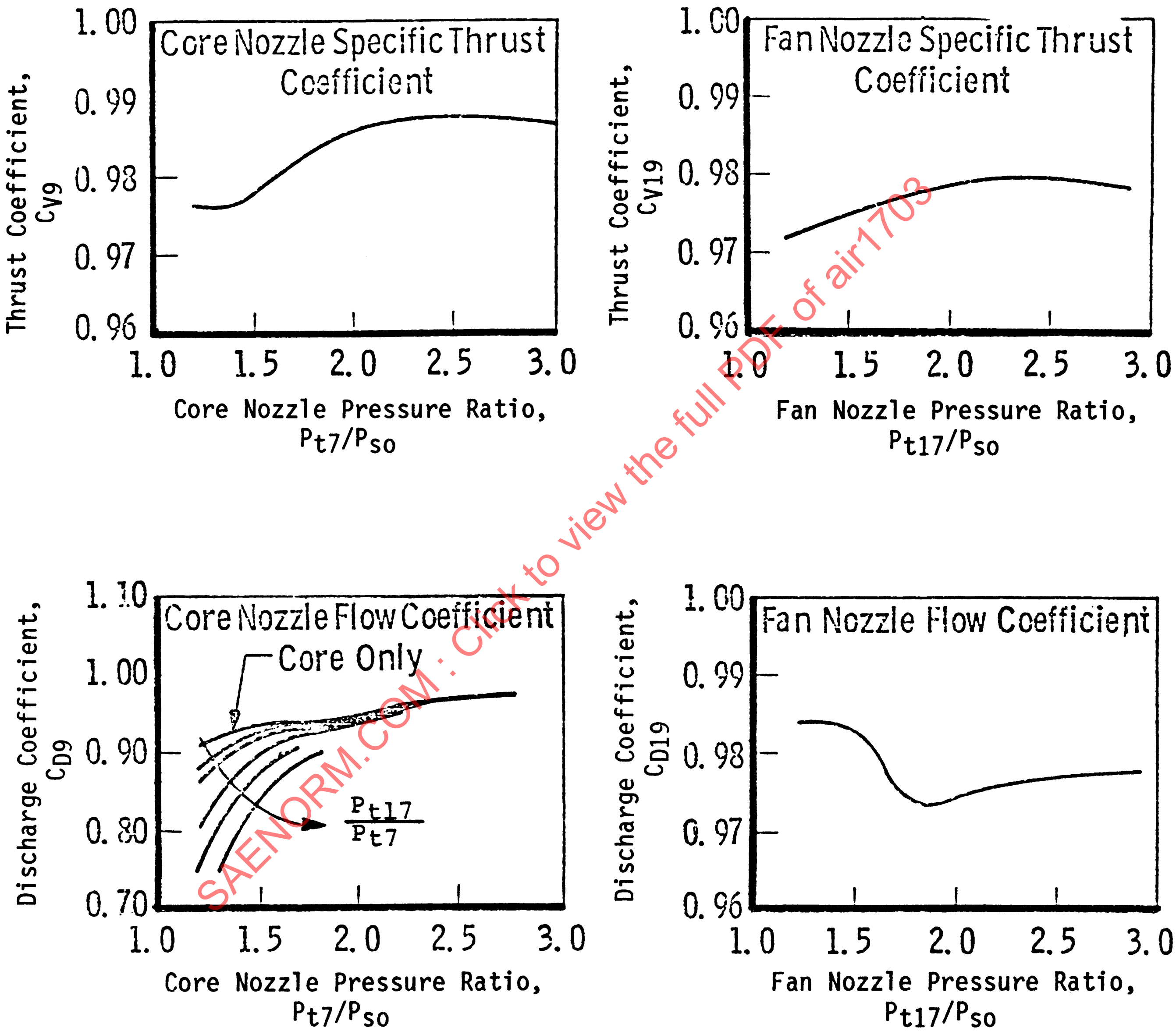


Figure 5.10 - Typical Separate Flow Exhaust System Nozzle Coefficient

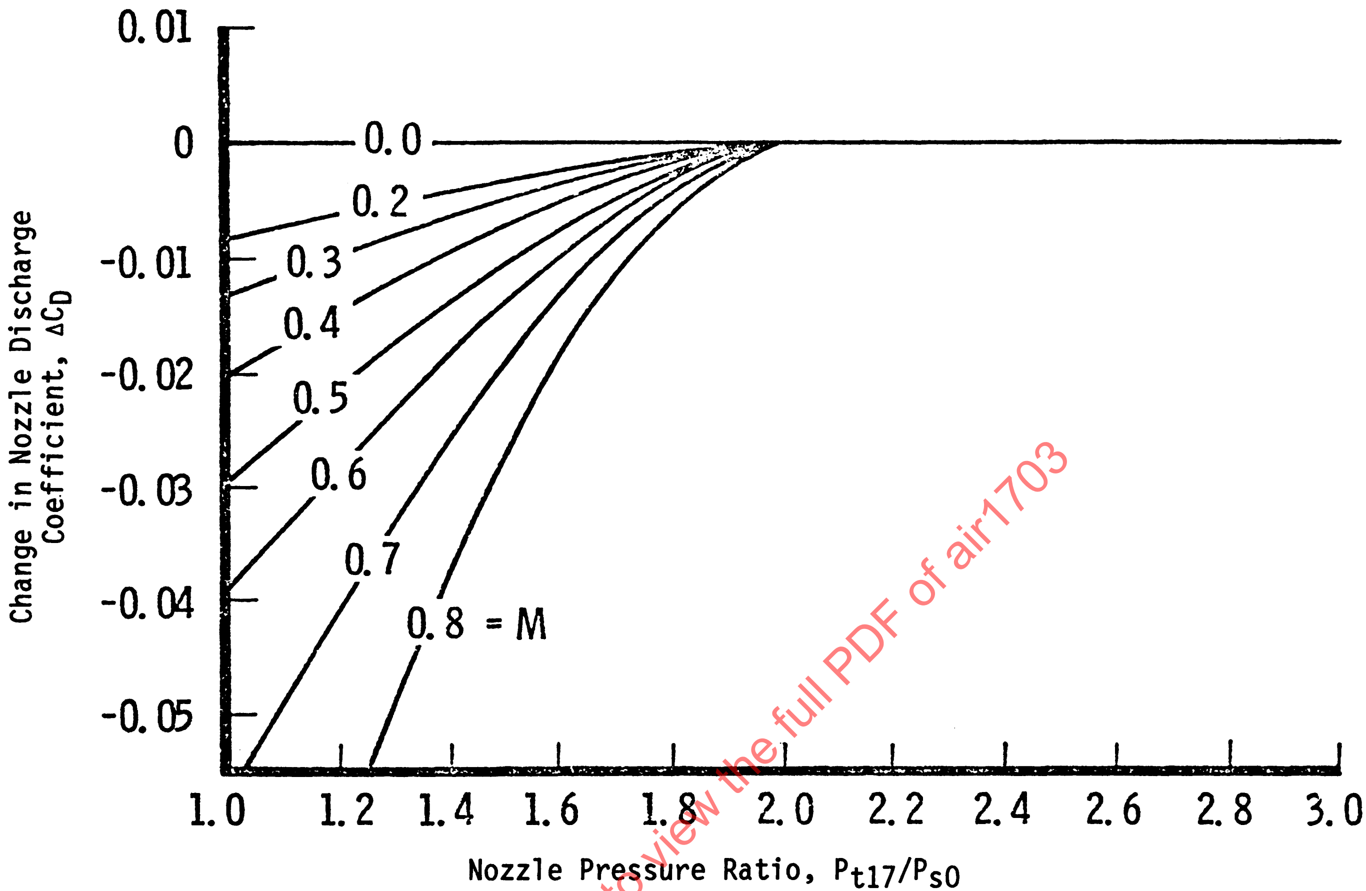


Figure 5.11 - Nozzle Flow Coefficient Suppresion (Isolated Nacelle)

5.2.1.2 Mixed-Flow Systems: For a mixed-flow exhaust system, as shown in Figure 5.9b, single-valued thrust and flow coefficient curves, can be used to characterize the static nozzle performance characteristics. These curves would normally be multi-valued. This form of nozzle performance characterization requires the development of an empirical relationship to define an appropriate average nozzle pressure ratio that will provide a nearly single valued curve. In general the form of that relationship is:

$$\frac{P_{t7_{AVG}}}{P_{s0}} = f\left(\frac{P_{t7}}{P_{s0}}, \frac{P_{t17}}{P_{s0}}, \frac{T_{t7}}{T_{t17}}\right) \tag{5.6}$$

The mass averaged total temperature is used in conjunction with the empirically determined total pressure to calculate the ideal flow and velocities that are needed to determine the nozzle coefficients.

5.2.1.2 Mixed-Flow Systems (Cont'd):

An improved version of this approach is to analytically model the duct losses and mixing process to arrive at a mixed nozzle-inlet total temperature and pressure that can be used with conventional nozzle coefficient curves. An adiabatic constant-area mixing process is assumed, using real gas properties and an empirically determined mixing efficiency.

Another approach to nozzle performance characterization for a mixed-flow exhaust system is based on the three nozzle performance curves illustrated in Figure 5.12 for the wind-off operating conditions. The relationships are parametric in core pressure ratio and bypass pressure ratio. The core and bypass total temperatures are used separately to calculate the flow functions, and the ideal velocities are based on expansion to ambient pressure. The single specific thrust coefficient is applied separately to the core and bypass ideal thrusts, with the sum being the total thrust. In some cases, it may be desirable to use the ratio of bypass to core pressure as a parameter, rather than the bypass pressure ratio.

External flow and installation effects on the nozzle flow coefficient may also be important for a mixed flow system. The approaches for handling these effects are identical to those discussed for separate-flow systems.

5.2.1.3 Variable-Area Convergent-Divergent Nozzle: For the variable-area convergent-divergent nozzle shown in Figure 5.9(c), the nozzle specific thrust coefficient and discharge coefficient curves, as shown in Figure 5.13 can be used to define the nozzle static performance. Performance curves will be needed to cover the range of variable geometry required for installed engine operation. Appropriate design studies should investigate the effect of nozzle throat area and area ratio on the installed performance in order to define the area ratio schedule. After selection of the area ratio schedule, it may be convenient to reduce the nozzle performance characterization to single-valued curves for that specific area ratio schedule.

External flow and installation effects on performance of internal-expansion nozzles are negligible due to the high nozzle pressure ratios of transonic/supersonic aircraft. For external-expansion ramp or plug nozzles, the effects of external flow can be significant.

5.2.2 Model Test Program Planning: This section provides guidelines for scale-model nozzle test program planning including comments on test requirements, facility considerations, and model design and instrumentation.

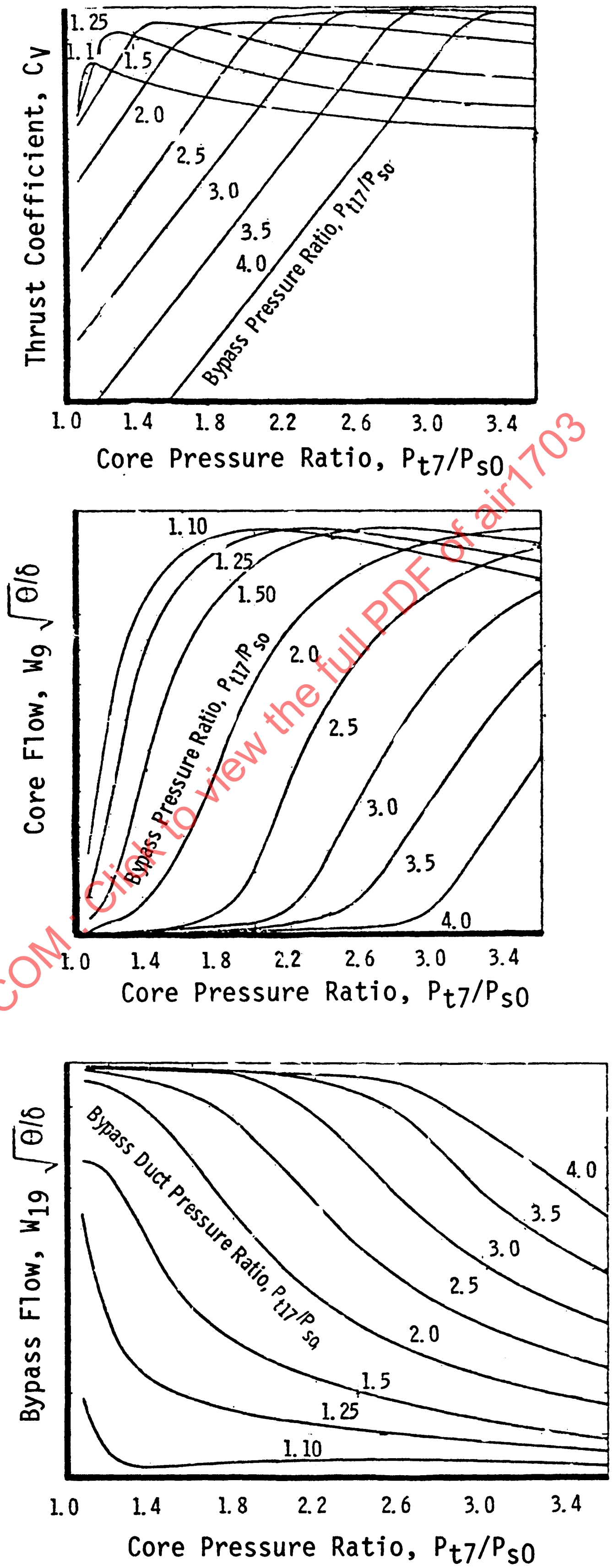


Figure 5.12 - Alternate Mixed Flow Exhaust System Performance Parameters

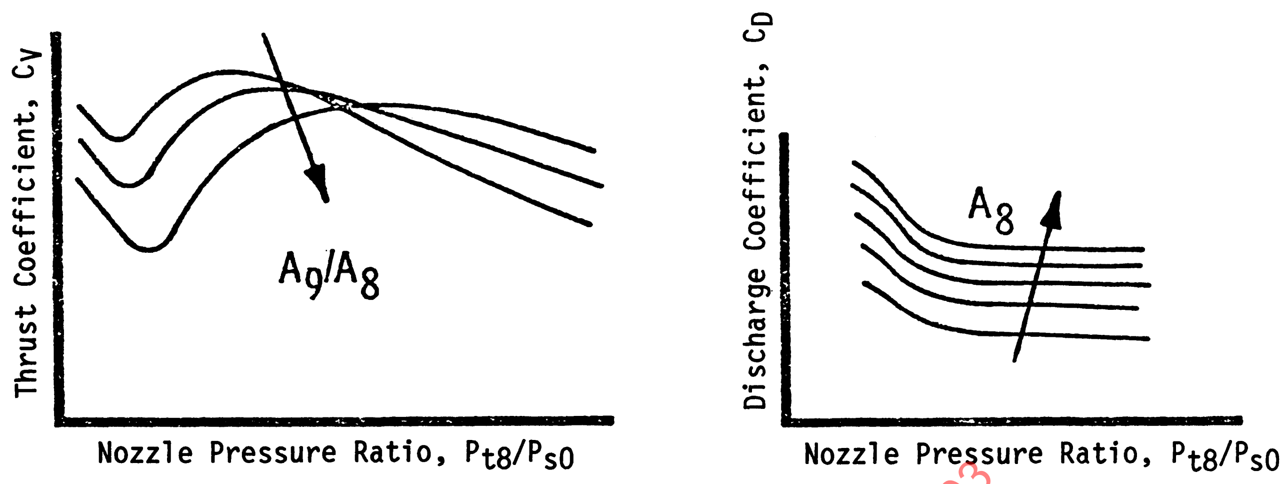


Figure 5.13 - C-D Nozzle Performance Curves

5.2.2.1 Test Requirements: Scale-model test requirements are based on an evaluation of both aircraft missions and engine operation within the flight envelope. This evaluation defines the types of testing and the data-point test matrix required to provide the data necessary to develop the exhaust system nozzle performance characteristics. The scale-model test program must be structured to generate data which completely bounds the pressure ratio ranges produced in flight.

For separate-flow systems, the core nozzle should be tested with zero fan flow over a range of core nozzle pressure ratios. During dual flow testing, the fan nozzle should be tested over the corresponding nozzle pressure ratio range, and for a range of fan-to-core stream total pressure ratios. A sufficient number of static test points should be run to allow cross plotting of the test results to obtain the nozzle performance curves depicted in Figure 5.10.

For mixed-flow exhaust systems, static tests are conducted with cold flow in both streams and with hot core flow at the full-scale engine core-to-fan stream temperature ratio. The cold flow data is used along with the hot flow data to determine a mixing effectiveness based on thrust and is used as a baseline for wind-on flow coefficient suppression tests. Additionally, a cold-flow confluent configuration is tested to determine mixer pressure loss by comparing test results with the cold-flow mixer configuration. An exit temperature and pressure survey may be used for verification of mixing effectiveness and pressure losses.

Tests to determine suppression effects should be included to better characterize the thrust performance of the installed engine. Figure 5.14 provides an illustration of the magnitude of the flow coefficient suppression at the key mission operating conditions. At high engine power settings, no flow coefficient suppression is observed. Significant suppression occurs at the lower power settings for the example considered.

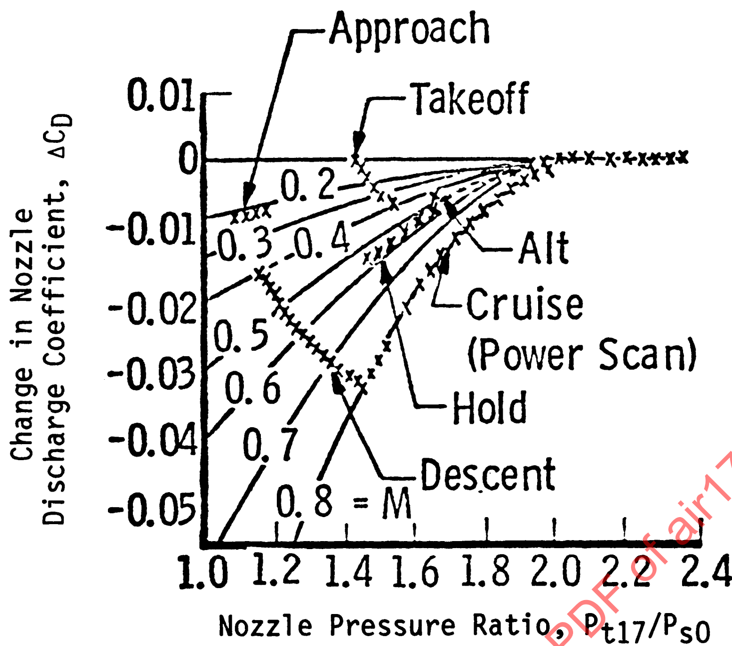


Figure 5.14 - Nozzle Flow Coefficient Suppression

5.2.2.1 Test Requirements (Cont'd.):

A similar evaluation may be made of the effects of the installation (wing/flap position, angle of attack, etc.) on nozzle flow coefficient suppression.

The scope of wind-on testing to determine installation effects for the calculation of in-flight thrust is dependent on the detail required in the installed performance characterization and upon the thrust/drag accounting system employed.

The assumption is made that the installation interference affects only the nozzle flow coefficient, with other effects included in the vehicle drag polar, including the throttle-dependent exhaust system drag. This permits wind-tunnel suppression testing to be accomplished without force measurement.

For the variable-area convergent-divergent nozzle, specific requirements for test matrices will be unique to each nozzle system and related design points, area ratio schedules, operating conditions, etc. The range of test variables (both geometry and pressure ratio) should cover the range of operating conditions over the important performance rating points. There may be some cases where important design point pressure ratios are too high to be attained by the facility. For these conditions, it is accepted practice to define thrust coefficients using a constant stream thrust parameter and flow coefficient. The stream

5.2.2.1 Test Requirements (Cont'd.):

thrust parameter, f_s, is the absolute stream force divided by an appropriate pressure-area term and, for a single nozzle, can be related to the specific thrust and flow coefficients as follows:

$$f_s = (F_{G9} + P_{s0}A_9)/P_{t7}A_8$$

$$= C_D C_V \left[\frac{F_G}{W\sqrt{RT}}\right]_{id} \left[\frac{W\sqrt{RT}}{A_8\ P_t}\right]_{id} + \frac{P_{s0}}{P_{t7}} \cdot \frac{A_9}{A_8} \tag{5.7}$$

Since both the stream thrust parameter and the flow coefficient remain constant once the nozzle operation becomes supercritical, Equation 5.7 can be rearranged to calculate the specific thrust coefficient.

$$C_V = f_s \left\{ - \left(\frac{P_{s0}}{P_{t7}} \cdot \frac{A_9}{A_8} \right) \Big/ C_D \left[\frac{F_G}{W\sqrt{RT}}\right]_{id} \left[\frac{W\sqrt{RT}}{A_8 P_t}\right]_{id} \right\} \tag{5.8}$$

Figure 5.15 is an example of test data which shows that C_D and f_s become constant and permit the extrapolation of C_V. The data must be recorded for at least four different pressure ratios where the stream thrust parameter is constant in order to obtain a good average value. This usually occurs at or above pressure ratios near the peak thrust coefficient. Additionally, data should be recorded at conditions where considerable engine test data analysis is expected, and nozzle performance assessment will be required even though the operating conditions may not be at key rating points.

5.2.2.2 Facility Considerations: Successful scale-model nozzle performance testing depends on the accuracy characteristics and flow capability of the facility being utilized. Facility accuracy and repeatability requirements should be established and imposed to insure a high confidence level in the test results. Facility accuracy and repeatability characteristics are based on performance data acquired from tests of accepted calibration nozzles, usually ASME nozzles.

Both static and wind-on nozzle tests may be required to determine the exhaust system performance characteristics of a particular configuration. Data can be acquired in either continuous flow or blowdown (limited flow) test facilities. The facility selection process involves consideration of facility availability, confidence in the facility (experience and availability of historical data), and test program cost.

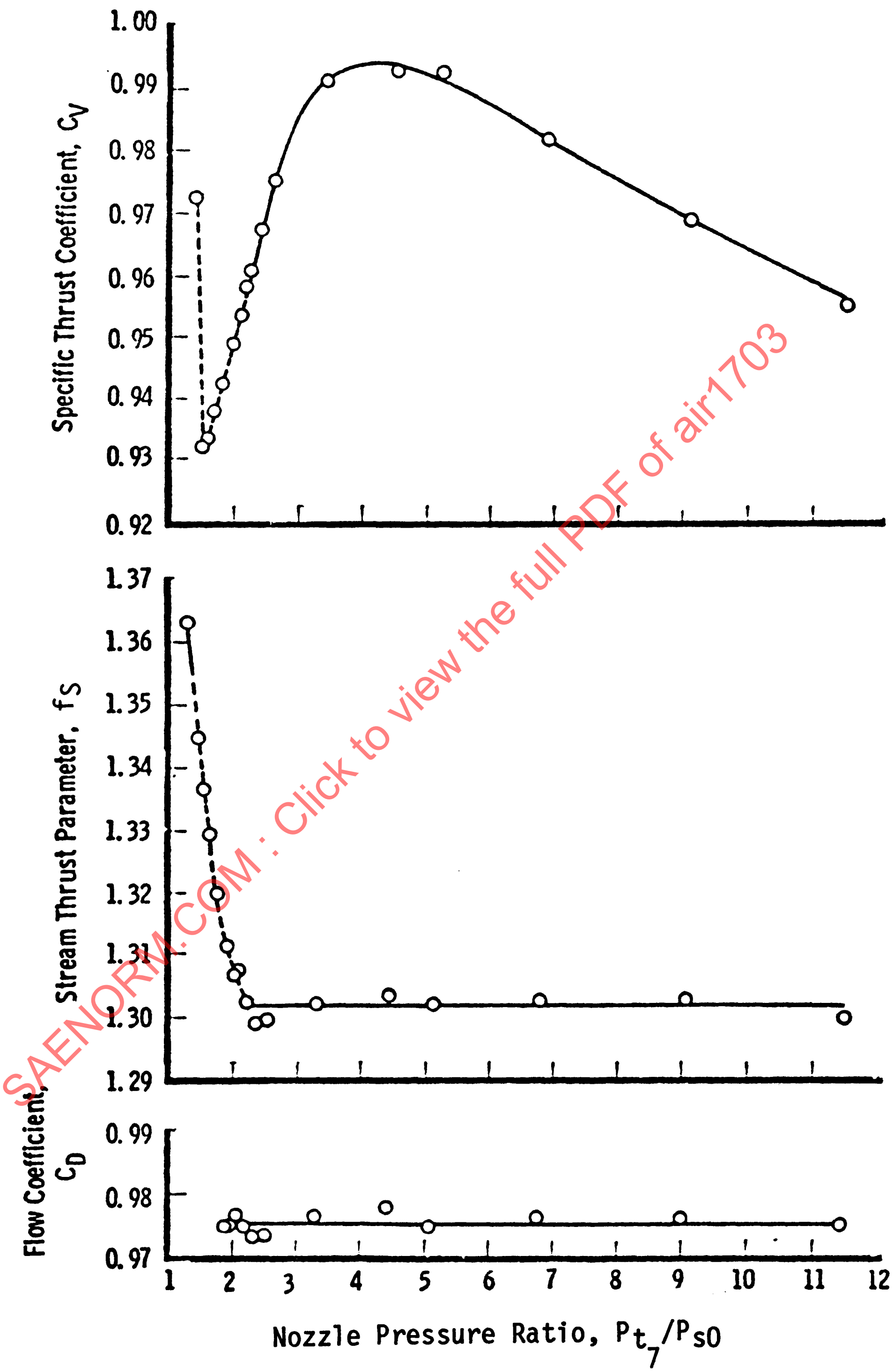


Figure 5.15 - Extrapolation of Nozzle Test Data

5.2.2.3 Model Design and Instrumentation: The design and fabrication of the model hardware depends upon the type of test (separate flow, cold, or mixed hot/cold flow) to be conducted. For example, hot flow tests require special materials, for the facility adaptors and the model, that are not required for cold flow tests. In both cases, the materials used must be both thermally and dimensionally stable. In all cases, the adaptors should be designed to provide a smooth channeled flow path into the model from the facility. Due to the necessity of having to periodically replace malfunctioning instrumentation, an additional design constraint is the ease of model build-up and teardown when test duration is limited. A description of the design and fabrication process for the two types of tests will be covered in the following paragraphs.

Separate flow exhaust system models are generally tested in a cold flow environment in one or more of three modes; static, isolated wind-on, or installed wind-on. The model size (scale) is normally determined by the airflow capacity of the facility, specifically the ability to reach the desired maximum pressure ratio for each mode of testing. If the same model hardware is to be used in all three modes, the model scale is chosen so as to be compatible with the facility that has the lowest flow capability.

To obtain the appropriate model definition for cold flow testing, full-scale nominal coordinates (or offsets) are sized to the chosen scale. The full-scale offsets should be the hot loft lines. The major geometric features of the exhaust system should be simulated in the model; i.e., the bifurcations, pylon, fan case, fan case struts, turbine case, and turbine case struts. However, the model representation of the full-scale hardware is limited, due to physical limitations in providing a simulation of the acoustic treatment, thrust reverser links, guide vanes, bleed ports, and associated steps and gaps.

It may be desirable to simulate in the model the radial and circumferential total pressure profiles and swirl that exist in the engine at the entrance to the exhaust system. Total pressure profile simulation may be accomplished by placing screens in the model/facility adaptor section. The last screen should be located as far as possible upstream of the model total pressure measurement station.

The model instrumentation incorporated to measure the exhaust stream total pressure and temperature for nozzle coefficient determination should be located as close as possible to the nozzle charging station. As a goal, the model should contain at least the same quantity of pressure instrumentation (number of rakes and probes) as the full-scale hardware. Static pressure taps located adjacent to each rake on the inner wall and the outer wall are desirable to facilitate total pressure averaging techniques for the scale-model data.

5.2.2.3 Model Design and Instrumentation (Cont'd.):

Hot-flow models are more difficult to design and fabricate because of the test environment. The model scale is normally determined by the airflow capacity of the hot flow facility. The facility must provide the proper airflow at the desired total pressure and temperature. From a design aspect, thermal expansion of the model requires special attention. The model must be designed so that the hot flow lines are geometrically similar to the corresponding full-scale hot flow coordinates. Provisions are needed to minimize warpage due to the thermal growth.

The external contour of a convergent-divergent nozzle is not critical in scale-model static tests provided steps are taken to insure that ambient pressure exists on this area. When tests are conducted in a chamber to achieve desired operating conditions and Reynolds numbers, a shield may be required at the nozzle exit plane, as shown in Figure 5.16, to prevent jet flow recirculation within the chamber. Static pressure taps should be installed on the nozzle external surfaces to assure constant, ambient pressure on the model. The model internal contour must be identical to full-scale geometry for proper definition of friction and angularity losses. Figure 5.16 describes some of the important major parameters which are necessary to simulate in the scale model.

The quantity of instrumentation used for convergent-divergent nozzle tests is normally not consistent with full-scale engine instrumentation. Scale-model instrumentation should be sufficient to define nozzle inlet total pressure and temperature for the model test. Static pressure instrumentation should be installed on the primary and secondary nozzle to determine gas loads for full-scale hardware mechanical design and diagnostic analyses.

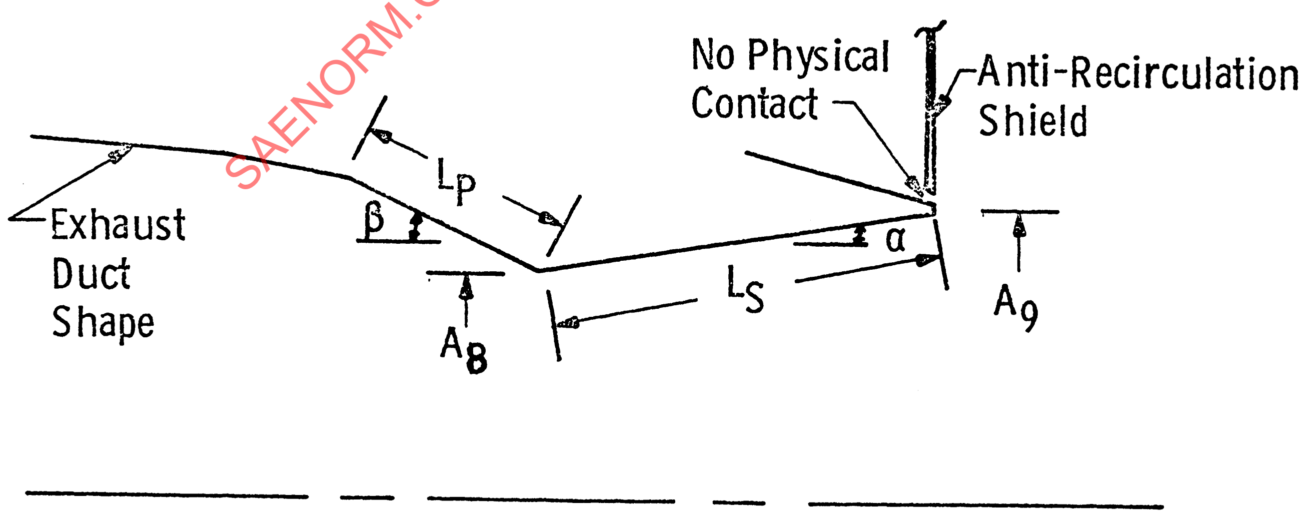


Figure 5.16 - Model Geometric Similarities

5.2.3 Testing and Data Reduction:

5.2.3.1 Test Procedure: Reference nozzles, such as those of ASME, should be run at the start of the static test program to check the facility operation. Data from each system of a dual flow facility should be obtained using an appropriately sized nozzle and compared to previous historical and analytical results. This comparison can be used to demonstrate the repeatability of the facility and to validate the model performance results. The same set of reference nozzles should be run at the end of the test to insure data closure and to provide facility repeatability information.

Two nozzle sizes are normally used for calibration in the test program for a variable-area convergent-divergent nozzle: one corresponding to the minimum dry throat area and the other for the maximum afterburning area.

All pressure instrumentation in the model should be leak checked prior to the start of testing. A continuous check of pressure data, e.g., total pressure profiles, pressure distortion levels, static pressure profiles, and any related facility parameters, should be made during the test program to provide early and timely detection of any instrumentation problems that may arise in the model or the facility.

Accurate flow coefficient determination requires an accurate determination of the nozzle throat area. The techniques used to determine fan and core nozzle throat areas must be analytically sound and physically achievable. The procedures used should be consistent with the area measurement used on the full-scale hardware.

At the start of static testing, the core nozzle is operated over the required range of pressure ratio, taking data in ascending pressure ratio in order to avoid hysteresis. A curve fairing through the core nozzle specific thrust coefficient data is established for use in the data reduction of the dual-flow runs.

Upon completion of the core only runs, dual-flow nozzle performance characteristics are obtained. The model is tested for a series of nozzle pressure ratios, with runs being made at each fan-to-core stream total pressure split. By using the previously measured core specific thrust coefficient, the coefficient level for the fan nozzle may be defined.

If a flow suppression test is to be conducted, the same model is then installed in a wind tunnel. The same set up and performance run procedures are used. For this test, the data from the static test are used as the guidelines for determining the nozzle flow coefficient curve levels and shapes. The model is then subject to forward speed (wind-on) conditions. The resulting data give free stream effects on the exhaust configuration. The boundary layers over the model, as installed in the

5.2.3.1 Test Procedure (Cont'd.):

wind tunnel, should be properly simulated. If the model was disassembled and reassembled, to install the wing, static performance should be repeated and compared to the previous data.

5.2.3.2 Extrapolation to Full Scale: The model test results require adjustment to accurately represent full-scale nozzle performance characteristics. Two types of adjustments are usually appropriate. The first corrects the wind-on flow coefficient data for the effects of test Reynolds number on friction and external boundary layer representation. A good approach is to acquire data at two Reynolds numbers for purposes of extrapolation to full scale. This procedure is required only for nozzle suppression testing for subsonic, low-pressure-ratio (unchoked nozzle) engines.

The second type of model data adjustment accounts for non-modeled losses such as leakage, cooling flow losses, thermal expansion, thrust reverser links, steps and gaps, acoustic treatment, oil coolers and the like. These types of corrections may be analytically derived or based on sub-component tests.

For variable area convergent-divergent nozzles, the Reynolds number corrections may not be necessary. The models are generally tested at Reynolds numbers near full scale, and the adjustment for frictional changes is nearly insignificant. The model data are adjusted for leakage and cooling losses as appropriate for the specific nozzle design. Additionally, variable-area convergent-divergent nozzles require a correction for differences in specific heat ratio between scale model and full scale due to gas temperature and composition. These corrections have the effect of changing the shape and shifting the peak of the gross thrust coefficient curves.

5.3 Scale-Model Afterbody Testing: Wind-tunnel scale-model afterbody testing is used to define the external drag characteristics related to variable geometry and jet effects (nozzle pressure ratio). Two groups of data are required: corrections to the afterbody aerodynamic force data base which adjust the afterbody drag for differences in test configuration and test conditions between the scale model and aircraft, and incremental force data which comprise the throttle-dependent afterbody drag (see Paragraph 2.2.2).

These results, properly scaled to the flight test environment, are required in the in-flight thrust determination procedure. Afterbody force characterization, pre-test preparation and design of the test model, and testing and data reduction are important considerations.

5.3.1 Afterbody Performance Characterization: Afterbody external force is characterized as a function of flight Mach number, nozzle pressure ratio, and nozzle exit area or an equivalent parameter defining nozzle variable geometry position. Figure 2.9 shows the variation in afterbody or exhaust system force, F_{EXH}, at constant Mach number. This is a convenient representation for a single flight test condition. Another representation is shown in Figure 5.17 with nozzle position held as the constant.

The exhaust system force increment, ΔF_{EXH} is the difference between F_{EXH} at full-scale actual and reference nozzle area and pressure ratio conditions.

5.3.2 Model Test Program Planning: Factors which need to be considered during the pre-test phase of a scale-model afterbody test program include the capabilities of the test facility and the degree to which afterbody, wing, or other airframe components in close proximity to the exhaust, should be modeled during the test. The testing techniques should be tailored to obtain results consistent with the established thrust/drag accounting system.

5.3.2.1 Facility Considerations: The choice of the test facility for scale-model afterbody testing should consider the same facility accuracy, repeatability, and types of operation (continuous or blow down), discussed in Paragraph 5.2.1 for scale-model nozzle testing. Facility flow capability and test section size are also important considerations.

Facility flow capability is the primary technical consideration. The Mach number range of the wind tunnel must meet the envelope requirements of the vehicle. Reynolds number capability--achieved by high operating pressures and low total temperatures--should permit model scaling with a minimum of boundary layer corrections and allow reasonable extrapolation to full scale. Depending on the type of engine exhaust system being tested, it may be desirable to attain high mass flows through the nozzle. In low to moderate flow facilities, this requirement may dictate an impractically small scale for the model.

In addition to the facility flow capability, the size of the test section is also an important consideration. The model must be small enough to prevent excessive "blockage" of the test section flow area that could bias the external performance measurements.

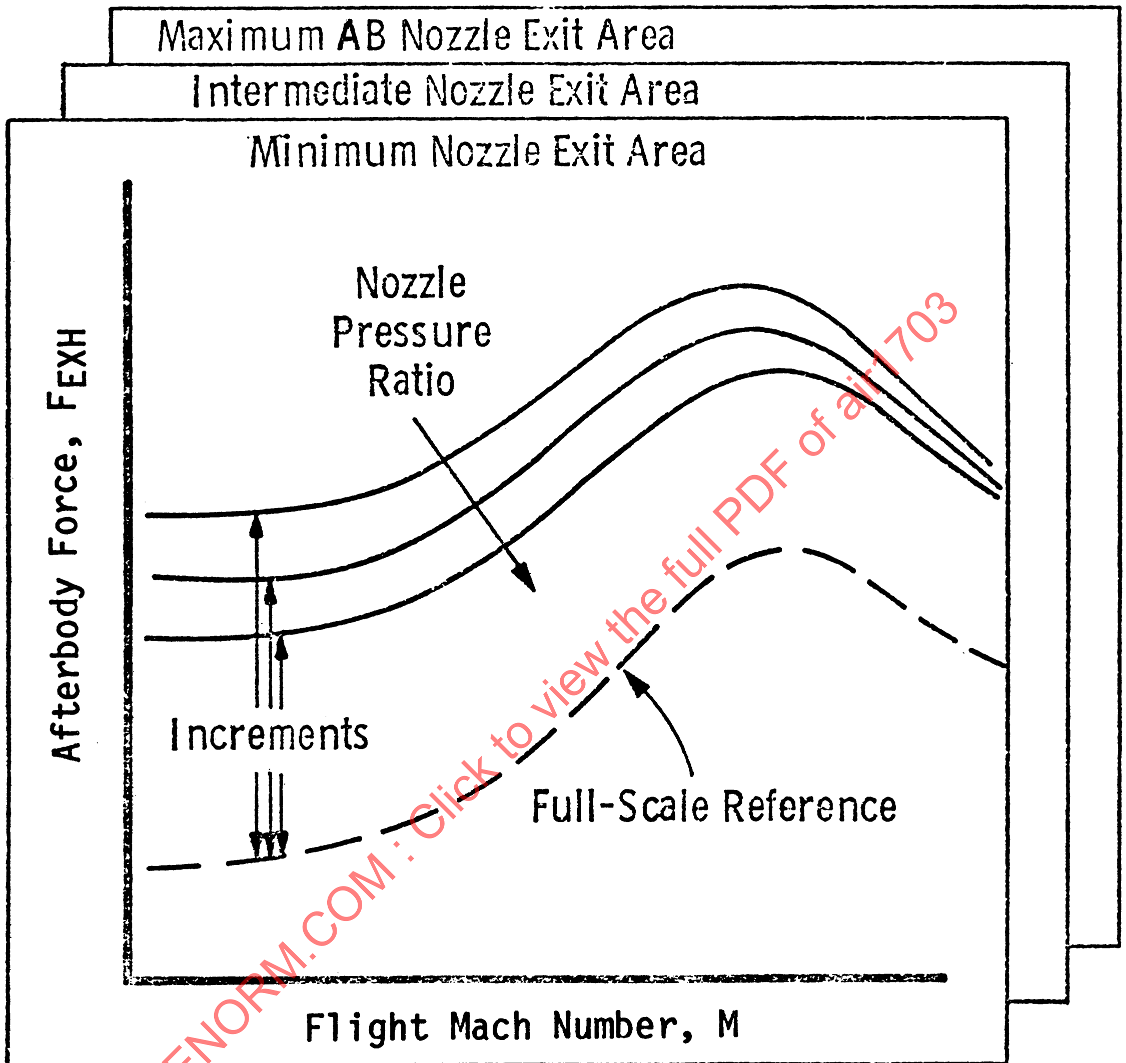


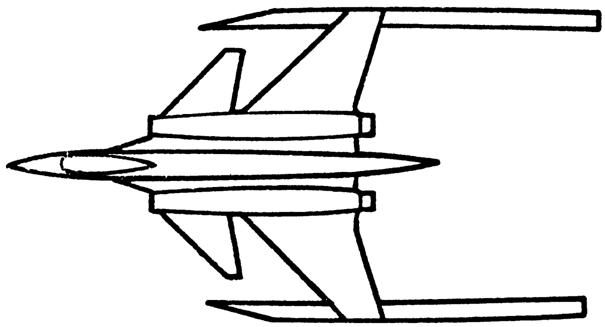
Figure 5.17 - Throttle-Dependent Afterbody Force Increments

5.3.2.2 Model/Support Design: The afterbody model in many cases must include the entire aircraft external surface if systematic errors in afterbody force test results are to be precluded. This is a consequence of two considerations: the model must include all external surfaces whose aerodynamic forces are significantly affected by nozzle operating conditions and must also include other surfaces which interfere with these areas. In most cases the inlets will be faired-over due to the difficulties in simulating simultaneously both inlet and exhaust flows.

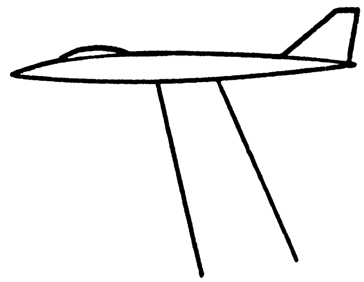
One objective of afterbody force testing is to provide data to correct from the aero-reference-model nozzle conditions to actual nozzle geometry and operating conditions by duplicating the aero-reference in the afterbody tests. The aero reference model may have no nozzle at all, e.g., it may be supported with a rear entry sting(s) up the exhaust nozzle location(s). The afterbody model tests must then include tests with a dummy sting to provide the reference force data.

The model support must provide sufficient cross-sectional area for the passage of the nozzle airflow and should not affect the aerodynamics of the afterbody region. Figure 5.18 shows several different support systems. For engine installation on or near the aircraft centerline, e.g., single and dual engine fighter and trainer aircraft, twin wing-tip supports best isolate the afterbody from support interference effects but may force a compromise of outboard wing geometry to provide sufficient airflow area. Fuselage-mounted struts provide sufficient airflow area but give rise to unquantified afterbody interference effects, particularly for aircraft with fuselage engine installations. The rear entry sting support is a suitable choice for aircraft with wing-mounted engines. It has been used for aircraft with twin fuselage mounted engines but usually with some compromise in afterbody contours and, therefore, with an associated unquantified systematic error in the thrust/drag accounting procedures. A promising newer technique is the rear entry sting with co-annular nozzle flow. This technique is applicable to both single and twin-engine configurations. Some supplementary testing using an alternate support system, e.g., wing tip supports or vertical tail tip support, may be required to establish the annular flow conditions required to properly simulate the complete nozzle flow.

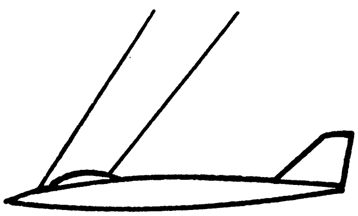
5.3.2.3 Testing Techniques: The afterbody force increments can be measured with force balances, surface pressure integration techniques, or a combination of these approaches. A smaller number of pressure measurements and flow visualization techniques, e.g. oil flow and schlieren techniques, can be used during afterbody design evaluation for configuration optimization and for isolating the origin of unanticipated results, e.g. boundary layer separation.



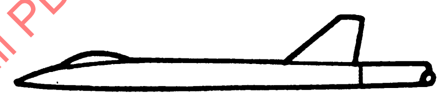
a. Wing-Tip Support



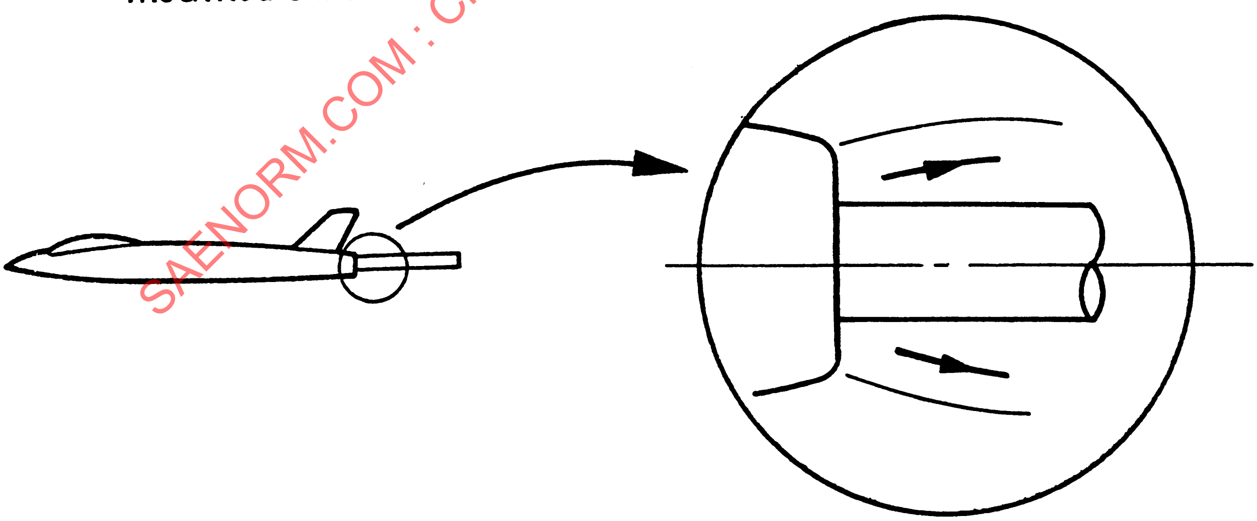
b. Lower Fuselage
Mounted Strut



c. Upper Fuselage
Mounted Strut



d. Rear Entry Sting



e. Coannular Rear Entry Sting

Figure 5.18 - Afterbody Model Support Systems

5.3.2.3 Testing Techniques (Cont'd.):

Where surface pressure measurements are used in conjunction with force balances, testing should be done with and without the pressure tubes to determine if the tubes act as significant redundant load paths. Any redundant load paths should be eliminated or force balance data usage should be limited to tests with the tubes removed. The use of plastic jumper tubes to cross the balance will usually render the redundant forces negligible.

Four methods to quantify the afterbody drag increments are discussed below.

- A single drag balance measuring only external drag forces with non-metric internal thrust nozzles: This method results in a non-representative trailing edge annular gap which is required for the drag balance arrangement. The significance of the gap is accentuated by model scale, and accurate extrapolation of the data to full scale is difficult.

- A thrust-minus-drag balance measuring a combination of external drag forces and nozzle thrust: It is difficult to obtain accurate results with this approach because it involves finding the small force difference between two large measured forces. Recently developed testing techniques have improved the results obtained with this approach.

- A single afterbody drag balance with pressure instrumented non-metric nozzle: The annular metric break is located in a smooth flow region. The nozzle is grounded to remove thrust from the model force equations, and the drag of the nozzle exterior surface is determined from the integration of pressure area terms developed from highly concentrated pressure instrumentation. This approach is described in Reference 5.27. The method provides an excellent compromise between the two approaches described above.

- Independent thrust and drag balances: This approach permits the simultaneous determination of thrust and drag increments. An undesirable feature is the resultant trailing edge gap cited with first method listed above.

In general, it is not possible to test a scale model configuration over the Reynolds Number range experienced by the full-scale vehicle. A separate assessment of Reynolds Number effects is therefore desirable. Typically, the effect of increasing Reynolds number is an increase in

5.3.2.3 Testing Techniques (Cont'd.):

the magnitude of local flow expansions and recompressions. It is recommended that some testing to determine Reynolds number sensitivity be performed.

The data base available with which to assess real gas effects by comparison of hot- and cold-flow test results is somewhat limited. Most afterbody tests have used cold air for nozzle simulation because of technical, cost, and safety considerations involved with hot flow testing. However, sufficient data does exist to make corrections to the cold flow data or to adjust the nozzle pressure ratio in the cold flow test to yield an afterbody drag force which is more representative of the hot gas conditions of the aircraft.

5.3.3 Testing and Data Reduction:

5.3.3.1 Instrumentation: Afterbody force model test instrumentation consists primarily of a force balance (or balances), pressure taps, and transducers for the measurement of nozzle pressure ratio and variable-geometry nozzle position. Frequently variable nozzle geometry is simulated in the wind tunnel with discrete test nozzles corresponding to key operating conditions of the full-scale variable nozzle. Freestream Mach number and model attitude is usually provided by permanent wind tunnel instrumentation and pre-existing calibration data.

The force balances and pressure transducers must be calibrated against known standards. Typically, a balance with six-component capability, three orthogonal forces and three orthogonal moments, will be used. The calibration must include loading of all components except possibly that corresponding to aircraft rolling moment. Further, the calibration data reduction must accommodate the possibility that any component may have a non-negligible interaction with any other components. Balance seals, which may provide a significant redundant load path, should be installed and thereby included as an integral part of the balance basic calibration. Further, following the basic calibration, a calibration for the balance forces induced by pressure differences across the balance seal must be performed. Finally, if a thrust-minus-drag balance is used, it must be calibrated with the nozzle flow momentum at the metric/non-metric interface.

5.3.3.2 Test Approach and Run Schedule: The test data must be sufficient to permit parametric determination of aircraft performance; yet the cost of wind tunnel testing precludes testing all combinations of all variables over their complete expected operational ranges. Consequently, the proper test approach involves the formation of a test matrix including all key mission points and limiting the number of test points devoted to multi-variable parametrics. For example, large excursions of angle of attack are required primarily for maneuver points. By contrast rather

5.3.3.2 Test Approach and Run Schedule (Cont'd.):

complete excursions of flight Mach number and nozzle pressure ratio will usually be done.

The test matrix should be structured in a run schedule considering the relative difficulty of changing test variables in the wind tunnel. If changing tunnel Mach number requires tunnel wall movement, all testing at a given Mach number should be grouped together.

5.3.3.3 Data Reduction: The data reduction procedure requires close attention to detail. The algorithms for obtaining the drag components must be carefully designed and checked. Data quality checks on all measurements should be performed frequently throughout the test. Presentation of the data in printed and plotted form should employ precise nomenclature and provide maximum visibility of the resultant data. Afterbody data should use the same area reference as the aerodynamic force model for the drag polar correction data.

5.4 Scale-Model Inlet Testing: Scale-model inlet testing is used to define both the internal (total and static pressure distortion, and recovery) and the external (lift, drag, etc.) performance characteristics of a particular design. These two test objectives are usually accomplished in separate model tests. The fact that inlet system external forces cannot be directly measured in-flight elevates the importance of obtaining accurate scale-model results where direct measurement is possible.

The inlet external force test results can be divided into two groups of data: corrections to the vehicle aerodynamic force data base which adjust the vehicle drag for differences in test configuration and conditions, and incremental force data which comprise the throttle-dependent inlet spillage drag. The first group of data addresses the external and interference force increments associated with the representation, by the inlet model, of full-scale inlet configuration and the operating reference mass flow ratio, which usually differ from the reference configuration and mass flow of the aerodynamic model. These data are acquired across the Mach number and, frequently, angle-of-attack spectrum. The incremental force data are required for the entire Mach number and operating mass flow ranges.

These results, properly scaled to the flight test environment, can increase the accuracy of the final in-flight thrust determination procedure. Inlet performance characterization, pre-test preparation and design of the test model, and testing and data reduction are important considerations.

5.4.1 Inlet Performance Characterization: Inlet system performance is characterized as a function of one key parameter, the inlet capture ratio, A_O/A_C, (or mass flow ratio), as defined in Figure 5.19. For a given freestream Mach number and inlet geometry, there is a maximum inlet capture ratio that corresponds to choked conditions at the inlet throat. This is called the critical capture ratio.

The breakdown of the inlet captured airflow into its constituent parts is an important performance characteristic. The bleed and bypass system flows must be known to compute the drag associated with each system. A typical set of airflow curves is given in Figure 5.20a. The duct airflow shown in this figure typically includes the engine airflow, any accessory airflow (e.g., for use by the Environmental Control System), and an airflow allowance to account for duct leakage.

Inlet internal performance can be characterized as a function of Mach number, inlet geometry, inlet capture ratio, and angle of attack, α. Typical curves of internal inlet performance are shown in Figure 5.20b.

Inlet external performance (i.e., aerodynamic forces) is also characterized by Mach number, inlet geometry, inlet capture ratio, and angle of attack. This is true for both an isolated inlet and an integrated inlet/airframe installation. Typical curves of inlet lift, drag, and pitching moment are shown in Figure 5.20c. In the propulsion thrust/drag accounting, the external inlet force, F_{INL}, is divided between the aircraft drag and the installed propulsive force by assigning the spillage drag at the full-scale reference conditions to the aircraft drag polar (see Section 2.2.1). The spillage drag increment, ΔF_{INL}, due to a change in mass flow ratio from the full-scale reference, is included in the installed propulsive force. Inlet lift and moments effects are included in the aerodynamic polar.

$$\left(\frac{Ao}{Ac}\right) = \frac{\text{Captured Streamtube Freestream Area}}{\text{Inlet Projected Capture Area}}$$

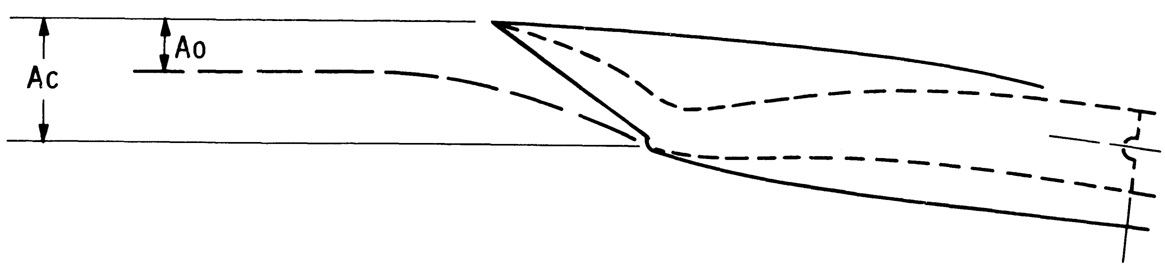


Figure 5.19 - Capture Ratio Definition

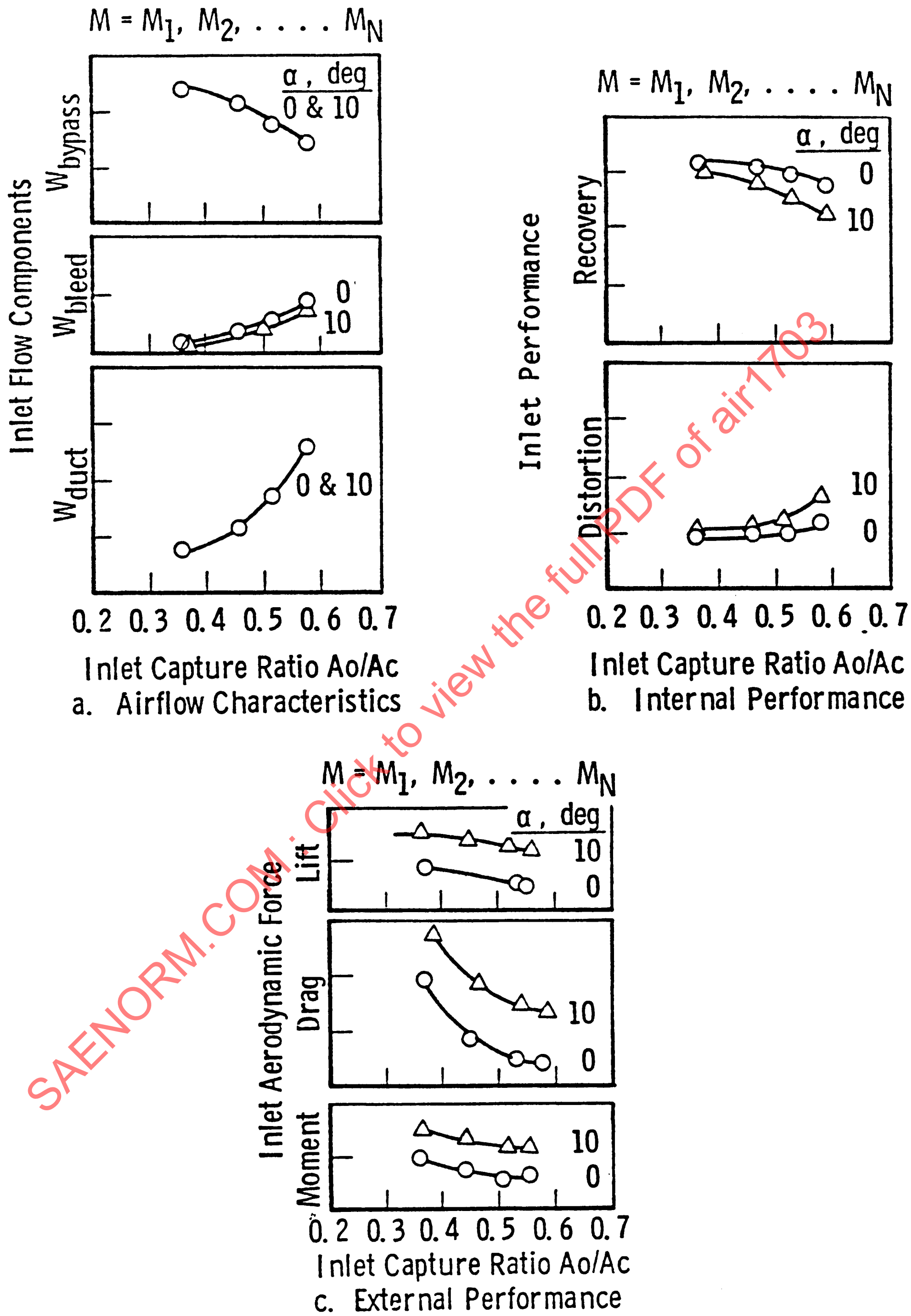


Figure 5.20 - Typical Inlet Performance Characteristics

5.4.2 Model Test Program Planning: Factors which need to be considered during the pre-test phase of a scale-model inlet test program include the capabilities of the test facility and the degree to which forebodies, wings, or other airframe components in close proximity to the inlet, should be modeled during the test. Testing techniques should be established to obtain results which are consistent with the selected thrust/drag accounting system.

5.4.2.1 Facility Considerations: The choice of the test facility for scale-model inlet testing should consider the same facility accuracy, repeatability, and types of operation (continuous or blow down), discussed in Paragraph 5.2.1 for scale model nozzle testing. Facility flow capability and test section size are also important considerations.

Facility flow capability is the primary technical consideration. The Mach number range of the wind tunnel must meet the envelope requirements of the vehicle. Reynolds number capability--achieved by high operating pressures, low total temperatures--should permit model scaling with a minimum of boundary layer corrections and to allow reasonable extrapolation to full scale. Depending on the type of inlet system being tested, it may be desirable to attain critical mass flows through the inlet. In low to moderate flow facilities, this requirement may dictate an impractically small scale for the model.

In addition to the facility flow capability, the size of the test section is also an important consideration. The model must be small enough to prevent excessive "blockage" of the test section flow area that could bias the external performance measurements. Also, the facility test section must be large enough to accommodate the desired range of model attitudes (angle of attack and sideslip) without creating undue model/test section wall interactions.

5.4.2.2 Model Design: Simulation of the inlet flow field environment will require the inclusion of proximate vehicle surfaces such as wing and pylon for a podded installation and nose, forebody (including any significant protuberances), diverters, strakes and possibly the wing root for a buried installation. It is important to duplicate vortices, flow angularities and local Mach number. In addition, local surface pressures at bleed or bypass discharge ports must duplicate the full scale vehicle.

Podded installations can safely be modeled (at least upstream of the inlet) as isolated inlets, if they are located a sufficient distance from the airframe forebody/fuselage. The degree to which airframe features located downstream of the inlet should be simulated on the test model depends on the speed regime in which the model will be tested. Full or at least partial modeling of wings, pylons, horizontal/vertical tails, etc. is generally required for testing at subsonic Mach numbers, 5.4.2.2 Model Design (Cont'd.):

because disturbances caused by downstream features can propagate upstream and affect the inlet flowfield. Full downstream modeling for supersonic testing is generally not required.

The decision to fully or partially model the inlet/airframe is also affected by the development status of the configuration under test. Early in the development of an inlet system, it may be advantageous to limit the degree of overall vehicle modeling. Usually, this will permit a larger scale model of the inlet to be tested with a larger number of surface pressure taps.

Maximum model size is dictated by the degree of full aircraft simulation included as well as test facility limitations and fabrication costs. A 15- to 20-percent scale would generally represent a practical upper limit on model size; this is influenced, of course, by full scale vehicle size. Minimum model size is influenced by Reynolds number requirements, geometric complexity and volume for instrumentation. A 5- to 7-percent scale would be a lower limit and should be used only if the full vehicle is simulated. Experience has proven that the model scale should be as large as possible within the aforementioned constraints. Model size directly improves the test quality due to the Reynolds number similarity and the larger forces measured.

The inlet force model must represent the inlet configuration of the full-scale vehicle. Any variable geometry features such as moveable compression surfaces, translating/rotating cowls, flapper doors, or blow-in doors should be represented. Compression surface bleed, inlet throat and/or engine bypass airflow, and any auxiliary flow which contributes to total inlet flow should be duplicated in the model. This is true regardless of the degree of vehicle modeling or the scale of the model. In some instances the requirement to fully model all bleed and bypass systems may be the determining factor in establishing the scale of the model.

Variable geometry on the model should be remotely positionable from outside the tunnel, if model scale permits. This results in significant improvements in tunnel utilization efficiency.

5.4.2.3 Testing Techniques: Accurate measurement of inlet total pressure recovery is essential during scale-model testing. This measurement has two important uses. First, it serves as a check on the accuracy of the in-flight measured inlet recovery. Secondly, it is used to compute the scale-model inlet drag that is ultimately used in the determination of in-flight engine thrust.

The presence of high levels of distortion can significantly degrade the accuracy of the inlet total pressure recovery measurement. Therefore, testing to document aircraft/engine/inlet performance should be done at conditions where inlet distortion is as low as possible, for example, at low to moderate angles of attack and with little or no sideslip.

5.4.2.3 Testing Techniques (Cont'd.):

External inlet performance is characterized by the resultant aerodynamic forces exerted on an inlet by the flow of air through and around it. Accurate determination of these forces during the scale-model testing is critical, since they cannot be directly measured in-flight. The model results must be properly scaled for use in the calculation of installed propulsive thrust, as discussed in Paragraph 2.2.

There are two primary techniques used in the determination of inlet scale-model aerodynamic forces: surface pressure integration and force and moment balance. Both have their relative strengths and weaknesses.

The surface pressure integration technique allows the contribution of the individual inlet elements (cowl, sideplates, ramps, etc.) to the overall inlet forces to be determined. This is an important advantage in the early development stages of an inlet system. It helps to identify those elements which can be adjusted to improve the overall inlet external performance. The disadvantage of this technique is that it generally requires a large scale model to accommodate the number of surface pressure taps needed to provide acceptable accuracy. Also, the large scale required for the model usually will limit the degree to which other airframe features (forebodies, pylons, etc.) can be simulated on the model. For some configurations, this limitation can severely reduce the usefulness of the data obtained.

The force and moment balance technique allows the use of a small, compact model. Important airframe features or even the entire airframe/inlet configuration can be modeled. Where the configuration has been frozen, this ability to model as much of the total vehicle as possible has been shown to provide accurate overall results for use in determining in-flight thrust. The disadvantages of this technique are two; extensive calibration of the balance must be performed, and a thermal control system to maintain the balance at a uniform temperature must be included in the model design. Despite these complications, the use of a fully metric total vehicle model has been demonstrated as a highly successful technique for measuring the effects of inlet airflow spillage on vehicle external performance.

The effect of inlet system airflow spillage on overall vehicle aerodynamic forces can be significant. Reference 5.12 concludes that the best technique for characterizing the effect of inlet spillage on overall vehicle aerodynamic forces is to employ a fully metric vehicle model which completely simulates the inlet system(s). The force and moment balance technique is recommended over the pressure integration technique because it is more accurate at low subsonic conditions. Reference 5.12 also re-emphasizes that complete simulation of the inlet system(s) is necessary to minimize bias errors in total vehicle aerodynamic force measurements caused by interaction effects resulting from bleed and bypass system operation, and inlet geometry variations.

5.4.3 Testing and Data Reduction:

5.4.3.1 Instrumentation: The manner in which testing is conducted contributes much to the degree of success (or lack thereof) in accomplishing the test objectives within the allotted schedule and cost. A detailed test run plan, designed to maximize test quality and efficiency, is a basic requirement. Similarly, a data reduction plan and thoroughly checked data reduction software must be available in advance of the test. The following paragraphs present examples of instrumentation requirements and the practices and procedures for test conduct and data reduction which serve to promote a successful test.

The effort to maximize model geometry and flow field similarity must be supported by the employment of quality instrumentation in sufficient quantity to provide the data being sought. The two primary measurements in the inlet force model test are force and mass flow. Other measurements required are static and total pressures, temperatures, position (displacement), and flow angles.

The force balance requirements--the number of axes instrumented for force and/or moment--are set by the objectives of the test. The majority of inlet drag tests require the measurement of axial and normal forces. The balance is mounted inside the model and supports the entire model (or metric portion thereof) on the sting. Thermal control blankets are employed to maintain the balance at a constant temperature during calibration and testing. Properly calibrated, such a balance will achieve a measurement uncertainty of 0.5 percent of rated load.

Inlet mass flow is a primary variable and must be controlled and measured with great precision. A goal of 0.5-percent uncertainty of the airflow measurement is justified. A 1-percent measurement uncertainty should be considered an upper limit for the inlet mass flow measurement. A 1- to 2-percent uncertainty in the measurement of bleed and bypass flows, which represent up to 20 percent of the total inlet flow, is adequate.

Inlet mass flow measurement on a sting-supported, flow through model is best accomplished using a calibrated flow plug installed in the sting. The plug is actuated to translate for flow modulation.

Some small-scale complete models with flow-through ducts and the wing-mounted podded nacelle require the use pressure rake data for airflow measurement. The mass-flow calibration, in terms of total and static pressure and temperatures, is capable of yielding accuracies equivalent to the flow plug system, but may require calibration. Mass flow modulation may be accomplished by inserting screens of varying porosity or plugs in the duct system. If plugs are used, care must be taken to isolate any effect on the force balance measurements.

A multiple-probe engine face rake is used to measure inlet total pressure recovery. The number and type of probes used on this rake depend on the particular application and on the desired/required accuracy. The design of a typical engine face rake is discussed in

5.4.3.1 Instrumentation (Cont'd.):

Reference 5.13. It is critical that the rake used for the scale model inlet testing allow integration with the flight test results.

The rake should have low response (<1/2 Hz) instrumentation to measure steady-state inlet total pressure recovery and distortion, and high response (up to 6000 Hz) instrumentation to measure time-unsteady inlet total pressure distortion, if deemed necessary.

A typical inlet scale model will contain many surface pressure taps. This is true even if the pressure integration technique is not used to determine the aerodynamic forces. Internal (ramp and duct) and external (cowl) pressures should be obtained at locations consistent with planned flight vehicle measurements to provide correlation/conformation of full-scale vehicle spillage drag characteristics. Other uses for surface pressure taps are: to locate regions of possible boundary layer separation, to locate the position of normal shock waves, and to determine the pressure potential available on any compression surface to drive flow through a bleed system.

Bleed and bypass systems should be adequately instrumented. As a minimum, the entrance and exit total and static pressures and the respective plenum pressures should be measured. These pressures are critical in determining the amount of flow through these systems.

Finally, all variable geometry elements should be instrumented. Position measurements of variable geometry are required to assure that they are correctly positioned before a given run and that they did not shift during the course of a run.

5.4.3.2 Test Approach: The overall test approach maximizes the usefulness of the data obtained while minimizing the tunnel time required to record it. This is accomplished through careful review of the aircraft mission/usage requirements. The test matrix is then tailored to provide as much data at critical mission/usage points as possible. All pertinent parameters are varied throughout their expected operational ranges to provide a broad data base for interpolation to conditions not specifically tested.

A run program is established that will minimize tunnel down time for test article configuration changes. For example, it is usually desirable to establish the correct tunnel flow conditions for a given run and then vary the inlet operating point (capture ratio) from its maximum to minimum value. This provides some continuity in the data being recorded and facilitates "on-line" detection of bad data. However, for flow-through type models, the inlet operating point is varied by installing different throat area nozzle "chokes". In this case, it is more efficient to record all of the required data for a particular nozzle "choke", and its associated inlet operating point.

5.4.3.3 Test Matrix: As discussed in Paragraph 5.2.3, the scale-model test matrix is established after careful review of the aircraft mission/usage requirements. While extensive data point replication would be desirable, it is simply not practical in the typical wind tunnel test environment.[5.12]

Test Mach number and angle-of-attack coverage should be thoroughly consistent with vehicle performance capability. Transonic testing should include a finer Mach number increment in order to reveal any local shock effects on cowl lip suction.

All possible combinations of the pertinent parameters cannot be tested. The tested combinations are usually limited to those containing all nominal values and those that combine the limit values of one parameter with the nominal values of all the others. In this manner, good baseline documentation data are obtained and the limits of the effects of off-nominal parameters are defined. To investigate any possible cross-coupling, a limited amount of data is also recorded with more than one parameter set at off-nominal values.

Selected test points should be repeated at a second Reynolds number to provide data for extrapolating the test results to full scale. A pattern of testing should be used which considers force balance accuracy, such as testing in the direction of increasing loads. This would, for instance, consist of progressively increasing angle-of-attack and/or decreasing mass flow ratio at a given Mach number. Balance output should be sampled three times at each stabilized condition. This can contribute increased accuracy through averaging.

5.4.3.4 Data Reduction: The data reduction procedure requires close attention to detail. The algorithms for obtaining corrected balance forces, mass flow, and the various drag components must be carefully designed and checked. Data quality checks on all measurements should be performed frequently throughout the test. Presentation of the data in printed and plotted form should employ precise nomenclature and provide maximum visibility of the resultant data. Force and mass flow diagrams are required showing the various components included in the data reduction scheme. Inlet model data should use the same area reference as the aerodynamic force model for the drag polar correction data.

5.4.3.5 Extrapolation to Full Scale: Test data acquired at two different Reynolds numbers will provide the primary means to adjust the vehicle drag and spillage drag to full scale. The primary difficulty in properly scaling the drag of integrated systems lies in the boundary layer characteristics and resultant inlet flow field effects. The reduced Reynolds number of the model can frequently cause improper scaling of the forebody boundary layer height. This effect may cause more boundary layer to be ingested into the model inlet than would occur on the full-scale vehicle and may sometimes be corrected by adjusting

5.4.3.5 Extrapolation to Full Scale (Cont'd.):

the model geometry (such as moving the inner sideplate outboard) to properly scale to boundary layer height. Another approach would be analytical correction using a 3-dimensional forebody flowfield computation. Pressure data acquired from the model will also provide an indication of boundary layer effects.

5.5 Turbopowered Simulators:

5.5.1 Introduction: As discussed in Paragraph 2.2, the aircraft drag polar is generally obtained by scale-model wind tunnel testing. Example models and tests required to define the baseline reference aircraft drag polar and installed propulsive force are also covered in Paragraph 2.2 and illustrated in Figures 2.6 and 2.7. An alternate test approach is the use of propulsion simulators. A propulsion simulator is essentially a miniature jet engine which enables the correct or near correct simulation of inlet and exhaust system flows, simultaneously. The simultaneous simulation of inlet and exhaust system flow is especially important when large flowfield interactions are likely between the inlet and exhaust systems and/or the propulsion system and the aerodynamic surface flow field; e.g., in-flight vectoring/reversing, STOL and VSTOL applications, thrust reversing, and close-coupled inlet and nozzle. For the above examples, testing one component without simulating the other (inlet but not nozzle) is liable to introduce significant errors in the evaluation of the component throttle-dependent drag forces and/or adjustments to the aircraft polar for power-on effects. Thus, the propulsion simulator was introduced as a test technique for scale-model wind tunnel testing.

It should be recognized that, although turbopowered simulators provide a better simulation of the inlet and exhaust flowfields, their use adds a degree of complexity to the test program. The simulator units require a significant amount of instrumentation which must be carefully monitored to provide high quality data, and the units must also be calibrated in a static test stand in order to accurately define the inlet airflow and nozzle thrust during the wind tunnel tests.

5.5.2 Types of Simulators: There are two types of turbine-powered propulsion simulators available for this type of testing: a turbine powered simulator (TPS) for subsonic applications and a compact multimission aircraft propulsion simulator (C-MAPS) primarily for applications with mixed subsonic/ supersonic mission requirements. The TPS is an off-the-shelf item and has been used quite extensively throughout industry.[5.14 through 5.18] Figures 5.21 and 5.22 show a schematic and photograph of a TPS model. On the other hand, the C-MAPS is still in the development stage. Figure 5.23 shows a cross-section of the C-MAPS and illustrates the C-MAPS airflow system. Literature regarding the development and operation of the C-MAPS may be found in References 5.19 through 5.26. Another approach employs ejector-powered simulators, although these are not illustrated in this document.

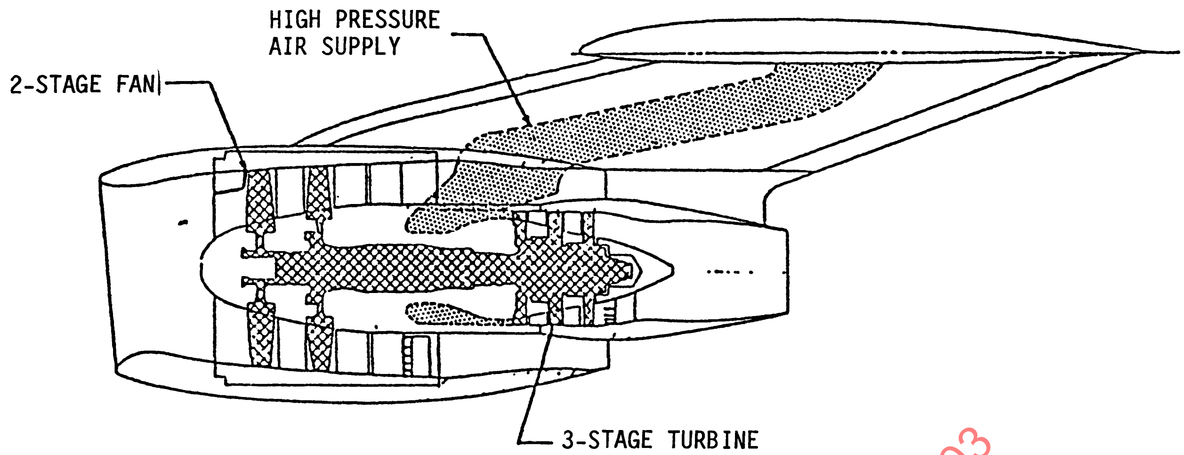


Figure 5.21 - Subsonic Turbo-Powered Simulator Cross Section

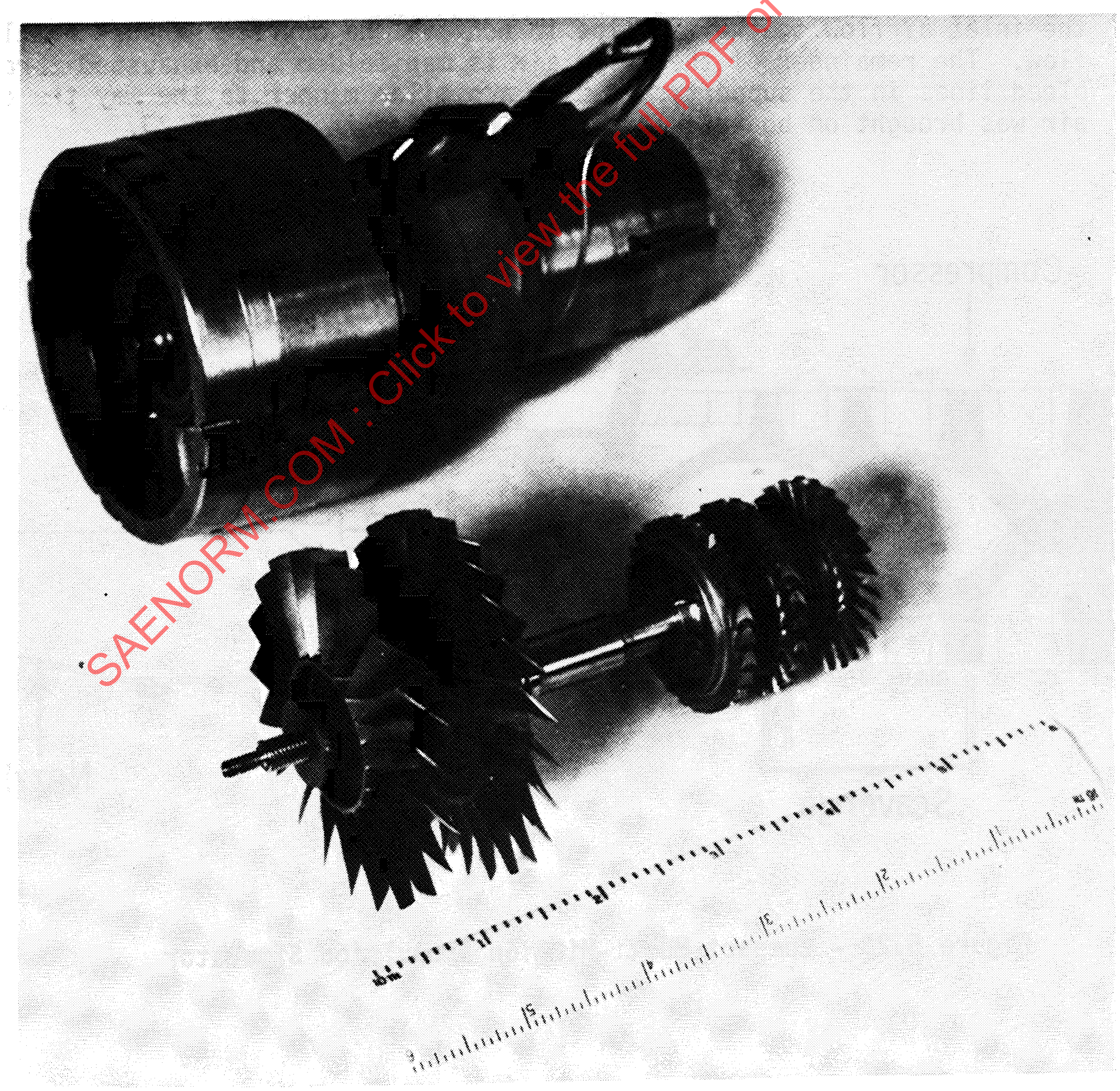


Figure 5.22 - Subsonic Turbo-Powered Simulator

5.5.3 Operational Characteristics: Both of the turbopowered simulators are driven by high pressure air brought on board the model through the support system. In the case of the TPS, which is primarily used for high-bypass-ratio turbofan engine simulation, all of the drive air is exhausted in a manner to simulate the engine core airflow. Therefore, the TPS only pumps the amount of air required for the fan nozzle or bypass, and the scaled model inlet is smaller than the scaled real inlet which would handle core and bypass airflow. This test technique is acceptable for directly comparing the installed performance of different nacelle aft ends with the same inlet. However, if power-on-adjustments to the aircraft drag polar are required, a more rigorous approach must be used that includes flow-thru nacelle model testing of both the real inlet and the TPS inlet in order to make the proper adjustments.

For the C-MAPS the inlet airflow does scale directly; however, because heat is not added to the flow, part of the spent drive air is mixed with the inlet airflow in the tailpipe to achieve the correct exhaust nozzle flow. The remainder of the drive air is manifolded and exhausted through bleed lines in the support system in a similar manner to the way the drive air was brought on board. This is illustrated in Figure 5.23.

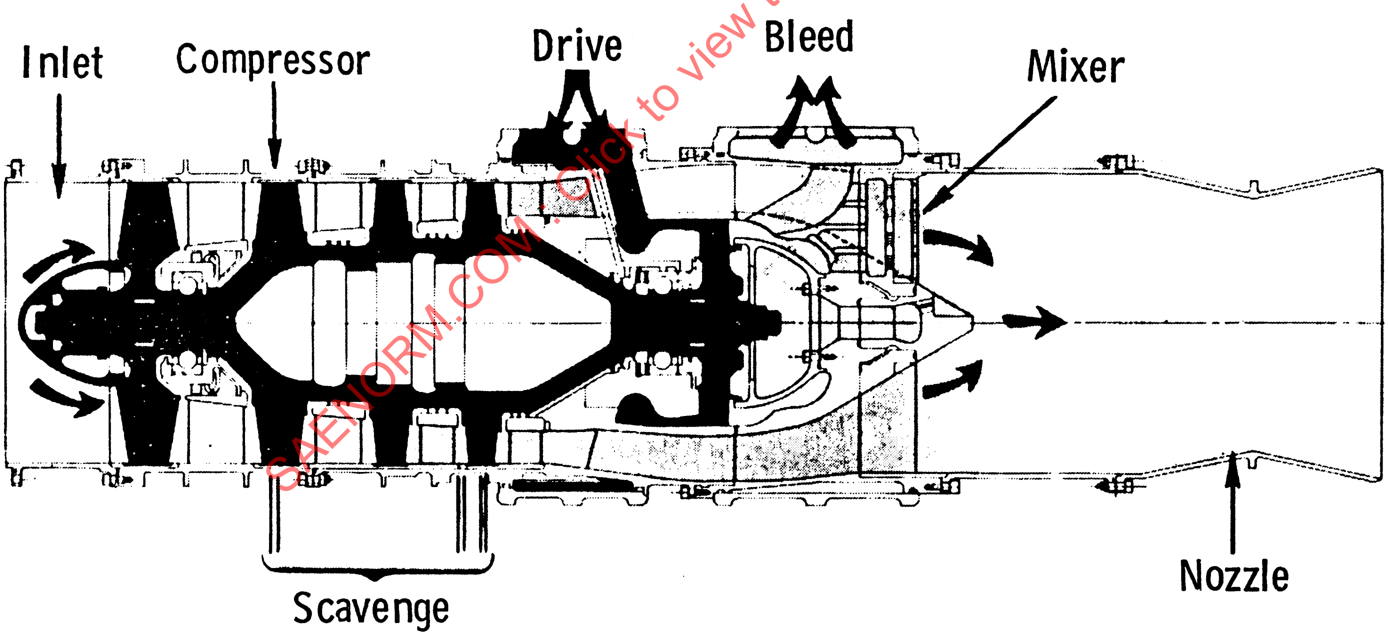


Figure 5.23 - Compact Multi-Mission Propulsion Simulator

6. DATA ACQUISITION:

The data acquisition guidelines center on sensor and data system requirements, both in flight and in ground facilities, for the described thrust options. Included in this discussion are sensor needs, signal conditioning, calibration techniques, recording, processing and checking.

6.1 Sensors: The sensors to be used for a test program as well as their number and location should be selected to match the thrust options chosen, the desired accuracy as determined through an uncertainty analysis, and the data analysis techniques to be used. The class (special purchase or off-the-shelf), range, number and location of these instruments should be based on an analysis of the planned flight conditions, the expected value and error for each of the measurements, and the anticipated environment. The type of flight testing will determine the frequency response desired from each sensor.

Good measurement resolution and freedom from interference are primary sensor needs. Resolution is the smallest change in an instrumentation reading that can be detected by the whole measurement system including the data reduction facility. Data system resolution capabilities must be compatible with overall system requirements as determined from an error analysis, e.g., sufficient bit size is a requirement which may in some instances necessitate a double channel signal. It is suggested that the resolution of a measurement be 2 to 5 times smaller than the allowable error.[1.1]

Interference can be encountered when an AC component is added to a DC signal or spurious pulses are added to a pulse counting system. These must be eliminated or minimized by improving the signal-to-noise ratio through appropriate filtering and grounding.

6.1.1 Types of Sensors: The types of sensors used for in-flight thrust determination are those pertaining to the measurement of pressure, temperature, rotational speed, fuel flow, area, force, acceleration, engine inlet mass flow, angle-of-attack, and pitch angle.

A pressure measurement at a specific point in a gas stream is usually sensed by a probe and then conveyed to a transducer. Many types of pressure transducers have been developed in recent years for use in engine testing.[6.1] Extreme care should be taken in the selection of pressure transducers for the air data system. The static pressure measurement is difficult, and the selection and placement of the tap contributes greatly to the accuracy of the measurement.

Hysteresis, non-linearity, zero drift, creep and temperature effects are factors that affect measurement error and must be considered in the selection of a pressure transducer. It is necessary to minimize these factors or account for their effects in the data analysis program.

6.1.1 Types of Sensors (Cont'd.):

Normally, the pressure measurement is made with a single transducer; however, for widely varying ranges of pressure in a flight program this may be inadequate. As alternatives, delta-pressure transducers or transducers of different range, which are controlled by a pneumatic switch, might be used.

Temperatures are measured using either thermocouples or resistance thermometers. The most general type of thermocouple used throughout an engine is that consisting of chromel wire and alumel wire, but the type of material chosen at a particular engine station should fit the temperature range of application.

Rotational speeds are sensed by either magnetic transducers or tachometer generators used in conjunction with eddy current type indicators.[6.2]

Two types of meter which are used to measure fuel flow rate are the turbine-type which measures volumetric flow and the mass-angular-momentum meter which measures mass flow.[6.3] Both types have some sensitivity to in-flow conditions so the calibration setup should include a representative portion of piping. Calibration of fuel flow meters is discussed further in Paragraph 6.6.

In addition to engine performance and vehicle specific range analyses, fuel consumption, as determined from the engine fuel flow meter, may be utilized to calculate aircraft weight during the flight mission. Aircraft weight is calculated as the empty vehicle weight plus remaining consumables (fuel, nitrogen, water, etc.) and the crew. Preflight and postflight weighing should be utilized to verify the calculation procedure.

It is necessary to quantify airflows other than the engine propulsive flow, such as engine compressor bleed, installation ventilation and inlet bypass. This may be accomplished through in-flight measurements or by analytical estimates. Where measurement is appropriate, the preferred method is a pre-calibrated flow-section using pressures and temperatures measured in flight. Other less direct methods have been used, including hot-wire anemometers.

Variable areas are most often measured indirectly by sensing the displacement of actuating gear with either or both a rotary or a rectilinear potentiometer. Because of the indirect nature of this method, the potential exists for large errors in this measurement. Alternative means of obtaining nozzle area are the use of a pucker string length or optical and digital encoders.[2.5]

Fixed areas can be accurately measured in the cold condition, but in order for this area to be used in flight, the material cold-to-hot relationship must be known as a function of temperature.

6.1.1 Types of Sensors (Cont'd.):

Measurement of scale force, as described in Section 5, is the most commonly used method of thrust determination for propulsion engines in a GLTB or ATF.[6.4] The sensors used to measure this force consist of (1) an element to which the force is applied and which reacts in a measurable manner as a result of this force, and (2) a measuring element which detects the change on the loaded element and supplies an indication of this change.

Common mass-spring devices for sensing acceleration are the strain gage and piezoelectric type transducers. This type of instrument has no fixed reference or structure; rather, the measurement is made with respect to some element which tends to remain fixed in inertial space. Factors which should be considered in the choice of these transducers are repeatability, compensation for temperature variation, cross axis sensitivity, stability under vibration, and frequency response.[6.5]

Accelerometers have been used to determine excess thrust in flight through direct measurement of acceleration. These methods employ highly accurate accelerometers to obtain an instantaneous measurement of inertial acceleration along the flight path. In order to obtain a direct measurement of excess thrust, the accelerometer must be aligned to the flight path; either mechanically, as in the Flight Path Accelerometer (FPA), or mathematically, as in the Body Axes Accelerometer (BAA). The FPA package is mounted internal to the boom and mechanically connected to external angle-of-attack vanes which are free to align themselves and the accelerometer with the flight path of the aircraft. The BAA is hard mounted to aircraft structure, and the flight-path acceleration must be mathematically calculated during data reduction. The derivation of each of these techniques can be found in Appendix C, Paragraph C.3.3.

Measurements of angle of attack and sideslip angle for determining the aircraft's attitude in flight may be obtained by vanes that align themselves with the flow field. A potentiometer, connected to each vane, relates the flow field angle with the attitude of the vehicle. It should be noted that corrections for upwash, position error, flow components resulting from angular velocities, surrounding flow field effects, and bending effects need to be made to the raw output.[6.6] Another method is the use of an array of static pressure taps along with a data reduction scheme that interprets the attitude of the aircraft from the relationship of the measured pressures, thus eliminating the need for a pitot-static noseboom.[6.7]

Pitch rate measurement is used in correcting for angle of attack and, also, as a correction to accelerometers which are not located at the vehicle's center of gravity and to dynamic data used in developing the performance characteristics of the vehicle. The sensor used may be a rate

6.1.1 Types of Sensors (Cont'd.):

gyro or part of the inertial guidance system, depending on the accuracy and sensitivity needed for the task. Sensor selection should consider the method of data acquisition, either dynamic or steady-state, and the follow-on use of the measured output as a correction factor to other data.[6.8]

6.1.2 Sensor Location: Sensors must be located to take into account radial or circumferential flow profiles which occur because of basic engine design features or installation effects. Locations are often selected to provide either area or mass-flow weightings. Sufficient total pressure and temperature probes and static pressure instrumentation must be located so as to determine mean values or consistent reference values of the flow parameters of interest at each measurement plane.[1.1]

6.1.3 Sensor Redundancy: Sensors, in addition to those needed for the basic thrust options, should be considered if failure of an important sensor would interrupt the test, if a significant improvement in measurement uncertainty can be obtained, and if changes in airframe or engine configuration during a test program may change the flow distribution at a critical measurement plane resulting in an erroneous average measurement at this plane. Should it not be possible to have sufficient sensors to fully characterize the altered distribution, there should at least be enough instrumentation to indicate whether or not changes have occurred. Redundant sensors might also permit the use of alternate thrust method options should key instrumentation for a primary option fail. Additional sensors can be used to improve accuracy or to cross check the accuracy of a sensed parameter. The uncertainty of a measurement may be lessened through association with other measurements (e.g., a voting system to eliminate bad measurements).

6.1.4 Sampling Rate and Dynamic Response: The frequency with which a measurement must be taken and the dynamic response required of the sensing transducer is determined by the type of testing. As an example, for steady-state conditions, pressures can be measured with a rotary-valve system, and the data sampled at a slow rate, e.g., 1 to 6 ports/second stepping rate, depending on the port-to-port change in pressure level. Transient conditions, on the other hand, will necessitate the use of high response pressure transducers which are monitored continuously, if engine operating pressures are to be known with a high degree of fidelity.

6.1.5 Range of the Instrument: The normal measuring range of an instrument is the range of values of the measurand which the instrument can measure with the accuracy, sensitivity and linearity specified for it.[6.9] Since the errors of many types of transducers are proportional to their range, generally the best accuracy will be obtained if the range of the instrument is equal to or slightly larger than the range of the measurand. The maximum measurand levels are typically encountered at the simultaneous occurence of lowest altitude, highest flight speed and highest engine

6.1.5 Range of the Instrument (Cont'd.):

power setting, and the minimum levels at highest altitude, lowest flight speed and lowest engine power setting. Obviously the instrumentation used must be capable of durably withstanding this minimum/maximum range.

6.2 Signal Conditioning and Conversion: The output signal from a transducer is usually modified several times before it reaches the final form in which it can be used for computations. Examples of these operations are signal conditioning activities that include amplification or attenuation, filtering, zero shifting, compensation, and signal conversion which is a class of operation where the signal is changed by such methods as modulation, demodulation or analog-to-digital conversion.[6.9]

6.3 Data Recording: Data acquisition in flight can be achieved in a number of different ways, and the method used should be determined from an analysis of relative costs, accuracies and program requirements. Several different approaches are discussed. It should be noted that data recording contributes to measurement uncertainty.

6.3.1 Cockpit Displays: The least expensive approach to data acquisition in flight is the use of cockpit gages. Contrary to popular opinion, cockpit gages are not necessarily less accurate than other data systems. However, the usual method of recording is by manual observation and can be subject to error and not easily verified. In addition, time correlation of the data is difficult. A further shortcoming is space limitation; the test aircraft may not be able to accommodate the required number of gages. This drawback may be improved by locating all the measurements on a separate, remotely located panel. Recording can be done by camera, thus eliminating the disadvantages of manual observation. Data reduction, however, becomes an unpleasant and time-consuming task of film reading and recording before analysis can begin. Film recording of cockpit data has been employed to provide permanent records and time-consistent data; but light parallax and interference problems render this a poor choice, at best.

Another type of cockpit display is the Head-Up Display (HUD), which permits the presentation of primary flight information, either situation or directory type, on a reflector or combing glass through which the pilot views the projected display and the outside world simultaneously. This system would normally be used to monitor critical test parameters rather than as the primary data recording system.

6.3.2 Paper Trace: On-board systems can also include paper traces, such as the pen and ink recorder and the recording oscillograph. The paper traces are helpful for the type of test where interpretation can be made directly from the recorded time histories, but these recorders have limited channel capacity, limited accuracy, and automated data processing is not possible.

6.3.3 Special Instrumentation Package: Most developmental test aircraft commit to greater data acquisition capability than is available with cockpit data. As previously stated, the cockpit data are usually sufficient to obtain thrust information and might be employed in a quick-look special effort, but its use is limited for an overall development program. For this, a special instrumentation package is usually employed and will consist of the various stimuli transmitters, signal conditioning equipment, and recording hardware. For some time, the use of this system implied the installation of tape recorder(s) in the aircraft, with the data recorded on these tapes at the command of the flight crew. Tape capacity usually limits the quantity of data that can be recorded. This method, if properly designed, provides data of a high level of accuracy, time correlated; and, when used in conjunction with appropriate data reduction programs and system hardware, is fairly easily reduced and presented for analysis. The major disadvantages, other than cost, are the size of the recorder, the required handling of the tapes, the possible limited number of test points imposed by the tape capacity, and the possibility of not activating the tape for the test point.

At a further increase in initial cost, some of these disadvantages can be eliminated by telemetering the data to a ground station and reducing it in real, or near-real time. The transmitted data are recorded on the ground, and because the ground station is not limited to one tape per flight, the entire flight can be recorded. Moreover, should something interesting occur at other than a planned test condition, it too can be recorded and analyzed. Processing the data in real time eliminates the tape handling and software setup required for reducing aircraft tape data. The disadvantage, other than cost, is transmission interference, which can inhibit data acquisition or discredit its validity. A possible second disadvantage might be transmission capacity limitations. Obviously, the most sophisticated (and expensive) course of action is to transmit, record and reduce data in real time, but also record the data on aircraft tape, which can be used should the telemetered information be deficient.

6.3.4 Ground Test Facility: GLTB and ATF's have sophisticated data systems not inhibited by the space limitations of a flight test vehicle. A larger number of transducers, signal conditioners, recorders and indicators are available in contrast to the aircraft system, which may be limited to just those systems needed for a specific test objective. Specialized calibration systems and techniques are also available in the ground facility. In short, all the data recording capabilities used for the flight program are available in the ground facility, but the system can be larger and more sophisticated.

6.4 Data Processing: Data processing includes the conversion of analog tape signals to digital form; the removal of redundant data samples; the merging of analog and digital data; the scaling of data to engineering units; the application of correction factors to measurements; and the arranging of data on plotted and tabular presentations.[6.10] Uncertainty analyses must include data processing.

6.4 Data Processing (Cont'd.):

Data processing equipment in the ground facility is available for processing, computing and reducing instrumentation system calibration data and data acquired during engine test operations. A broad range of computer programs is available for processing both digital and analog data to satisfy test data requirements. The ATF and GLTB data processing operations include on-line and off-line modes to provide processed data during and after the test operation.

6.4.1 Quick-Look: Data processing in the quick-look or on-line mode is concerned with monitoring safety-of-flight measurements and those measurements critical to the success of the test flight, with evaluating airborne instrumentation, and with time editing of data. Critical parameters can be monitored on the ground and in flight using paper traces, and on the ground as tabulated data on CRT displays or in flight on a HUD. This allows a selection of only those time segments of data for post-flight processing that demonstrate test objectives, allows for repetition of test points, and allows for identification of instrument malfunctions which could affect the processing of post-flight data.

6.4.2 Sample Rate: The desired sampling rate is a function of the type of test being performed, steady-state or transient, and the importance of the parameter being measured. In short, the requirements for the individual channels and the requirements for the total data collection system must be weighed before selecting the best sampling rate.

6.4.3 Synchronization: Aircraft flight tests usually involve the acquisition of large amounts of measured data. Raw data may be stored for subsequent processing or may be telemetered to a ground station. For most tests, some form of absolute or elapsed time reference must be recorded simultaneously with the data. The recording of time enables data recorded on different devices to be correlated accurately in time, even when processing is not performed simultaneously. It also allows an accurate time history of each measured parameter to be obtained.

The number of possible time-measuring systems is large, one of the simplest (for elapsed time only) being a periodic voltage waveform of a constant and known frequency. A more complex, but also more generally useful, system may provide absolute time-of-year (days, hours, minutes and seconds) in parallel digital form.(6.11)

6.5 Datum Checks: As previously mentioned, more than one thrust option should be used for a flight test program, and if a particularly sensitive measurement or thrust calculation is needed, a number of measurement options should be included. In particular, those measurements in inaccessible areas should be provided with back-up measurements in the event of a failure. In an adequately redundant system, an erratic or failed probe need not cause an aborted test nor a repeated test, even if the failure is not found until after the flight.

6.5 Datum Checks (Cont'd.):

Test procedures should include pre- and post-flight checks as well as instrumentation monitoring during a flight. Pre-flight checks should include an inspection of the mechanical condition of transducers, connectors, cabling, and tubing, recorders, etc. Each channel should be sampled and compared with known levels, e.g., differential transducers at zero and absolute transducers at barometric pressure, and where appropriate, leak checked. Stability and noise levels of all outputs, particularly with regard to any adverse interaction with aircraft systems, should also be checked.

A few basic parameters or measurements should be continually monitored on-board the aircraft to insure success of the flight. Already mentioned in this regard were gages and paper traces. On the ground, paper-trace data should be supplemented with basic parameters displayed on CRT's. In-flight calibrations and calibrations within seconds or minutes of a test reading are preferred. A comparison of these values with known references will verify whether a calibration has shifted. If so, corrective action should be completed prior to proceeding with the test. A shunt across the ports of a differential transducer during flight will also check for zero shift. Otherwise, a constant check should be made of the data against known profiles, distortion levels, or predicted values. Post-flight checks are necessary to insure that the entire data system is still functioning properly.

6.6 Calibration: Calibration is the process of comparing the measuring characteristics of an individual instrument with more accurate instruments or standards as references; such as international standards, national standards, and individual laboratory standards. The objective is to reduce the measurement uncertainty. Pre- and post-flight checks can also be regarded as a part of this calibration procedure.

The calibration should not necessarily be directed at achieving minimum error at full-scale (maximum) measurand value. It may be more important to consider the operating range at the test conditions of primary interest. For example, if the objective is to gather information primarily during altitude cruise conditions, the engine internal pressure levels will typically be in the region of one-third of full-scale value, and the instrumentation should be calibrated for minimum error in this range.

The static calibration of pressure transducers can be made with a dead-weight piston gage in which precision weights equivalent to a known pressure are applied to the test apparatus and thus to the transducer. Thermocouples most commonly undergo immersion tests in a fluid whose temperature after a sufficient stabilization time is measured with a precision resistor thermometer or thermocouple whose calibration is traceable back to a primary standard. Tachometer-generators do not require calibration due to their principle of operation, while the eddy-current indicators are usually calibrated using a tachometer generator driven by a motor whose speed is

6.6 Calibration (Cont'd.):

known and adjustable. Magnetic sensors do not require calibration, but their associated electronic devices, e.g., frequency-to-voltage converters, are calibrated using a sine-wave voltage generator.

Flow meters should be calibrated with piping representative of the actual aircraft installation since, as was previously mentioned, the flowmeters are sensitive to inlet flow profile, distortion, swirl, etc. A flowmeter calibration method having a high degree of accuracy collects the flow through the meter in a tank for a given period of time and weighs the contents of the tank. This method is time consuming and can be complicated. A second method of calibration involves comparing the flow through the meter with the flow through a simple, basic meter which has already been calibrated against a primary standard.

Force-sensor calibrations are accomplished by applying known forces to the instrument being calibrated and recording the output. Several known forces covering the range of the instrument are applied using a proving ring or weights, which are traceable to a primary standard, first up scale and then down scale, to determine hysteresis errors.

Steady-state and dynamic calibration techniques are used to document the characteristics of accelerometers.[6.5]

Position-error corrections, particularly for the pitot-static-system, angles-of-attack and sideslip-angle measurements, should also be included as part of the calibration factors. These corrections account for the effects of aircraft attitude on the sensed flow field surrounding the probes. They may be determined during the calibration of the probe system in a wind tunnel environment, where a series of attitudes for each Mach number are tested.

The accuracy desired for any calibration should be determined by a measurement uncertainty analysis. Each calibration should be compared with previous ones, and a record maintained through periodic checks. This provides for a near-term check and also for a long-term check of repeatability characteristics. This calibration should also be checked, to the extent possible, after the complete system has been installed. Finally, every effort should be made to have the same set of instrumentation for GLTB, ATF and flight tests to remove the uncertainties associated with different types of instruments, different calibrations, and different techniques for calculating thrust.

7. TEST ANALYSIS FOR THRUST VALIDATION:

Previous sections have dealt with the individual elements of propulsion system thrust definition and measurement. This section discusses the preflight and postflight consistency checks and validation of the data that are done to enhance confidence in air vehicle thrust measurements.

Procedures for investigating and quantifying the uncertainty of thrust measurements are presented in AIR 1678. This process must be tailored to the complexity and resources of each test program. The improvement in accuracy/uncertainty is accomplished through improvement in procedures and equipment, and correction, if possible, of errors which may be identified through analysis of all available information. The potential error sources that may be isolated by test-data analysis include those which are:

- Caused by unanticipated instrumentation problems (examples: instrumentation bias shifts subsequent to calibration, temperature sensitivity effects on the accuracy of pressure transducer measurements, or flight test maneuver/instrumentation response incompatibilities).

- Uncertainty analysis error propagation items that are either erroneously excluded, improperly modeled or inaccurately bounded (example: incorporation in the error propagation analysis error bands for instrumentation equipment without a sufficient historical data base).

- Attributable to ground test facility inadequacies, limitations or test techniques (examples: ATF basis for assessment of transient engine operation in flight, wind tunnel methodology for determining airframe/engine interactions, and transonic wall interference effects on subscale model test results used to define airframe/engine interactions).

- Attributable to changes in engine operating characteristics (examples: "rematching" of multi-spool engines caused by differences in installation effects between the altitude test facility (ATF) and flight environment, and engine control system operational differences between flight and that defined for the engine performance data base).

- Caused by what are classified as "mistakes" (examples: data reduction programming errors and thrust measurement errors associated with the use of different techniques to calibrate nozzle areas between ATF and flight test engines).

Potential errors in the error prediction analysis and the test results must be closely scrutinized. All available information should be used to improve data quality. Confidence is enhanced when two or more methods with relatively weak coupling between input and output are employed and their results compare favorably. Failure of the predicted uncertainty bands to overlap may indicate an error source and should be investigated and resolved. The following paragraphs discuss this validation process for both ground and flight test data using examples that are representative of actual test programs.

7.1 Preflight Engine Consistency Checks: The primary purpose of the preflight performance testing is to develop and calibrate the in-flight thrust procedure. The engine performance definition is developed in a multi-phased process. Some of the steps have been discussed in previous sections and include:

- o Engine component testing to define compressor, combustor and turbine efficiencies; fan/compressor airflow relationships; tailpipe/afterburner pressure drop; and combustor/afterburner temperature rise.

- o Scale model testing to define nozzle flow and thrust coefficients, engine installation effects, and airframe/engine interactions.

- o Ground level test bed (GLTB) and ATF tests which provide: uninstalled engine operating characteristics, i.e., uniform inlet airflow, no horsepower extraction and no compressor bleed airflow extraction; and installation sensitivities, i.e., horsepower extraction, compressor bleed airflow, and distorted inlet airflow which simulate the airframe installation.

All performance data should be included in the preflight engine consistency checks. Individual parameters, coefficients and overall performance are compared between model and full-scale tests and with simulation models. Inconsistencies should be investigated prior to the flight test portion of a program. When satisfactorily completed, these consistency checks improve the overall confidence in the test program.

7.1.1 Comparison of Mass Flow: Mass flow measurements for the in-flight determination of inlet ram drag and exit gross thrust may be obtained from one of the several methods discussed in Paragraph 4.2.1. Bookkeeping of leakage, engine bleed and fuel flows is necessary. Ground tests should incorporate more than one mass flow method, in addition to the primary facility airflow measurement. Airflow calculated using each of the methods should be consistent. Comparison of test data should be used to confirm the choice of the in-flight measurement method and to provide the calibration.

The primary airflow measurement systems in engine ground-test facilities are sonic venturis and calibrated engine inlet bellmouths. Both systems are discussed in Paragraph 5.1.5.2. The airflow consistency checks should be preplanned activities to validate both the primary airflow measurement and the preferred in-flight measurement method. Some typical airflow check methods are described in more detail.

7.1.1.1 Duct Checks: Duct checks consist of comparing the magnitude of engine airflow from the primary measurement system with the calculated airflow value at the engine inlet station or engine-duct slip-joint location (see Figure 5.5). A duct airflow calculation requires measurements of the total pressure and temperature, of the duct flow area, and of the wall static pressure which may or may not be used in conjunction with a

7.1.1.1 Duct Checks (Cont'd.):

duct flow coefficient or with a prescribed static pressure profile. Typical duct-check comparisons for a nonafterburning turbojet test are shown in Figure 7.1. The duct check data are within one percent of the facility primary airflow from the critical flow venturi. Since the facility airflow data is within the estimated measurement uncertainty band for the duct-check airflow, the consistency check is considered to be satisfactory. If the airflow data were outside of the estimated uncertainty band, additional analysis would be required to identify the error source.

Airflow duct checks are most effective for duct Mach numbers greater than 0.3. Measurement uncertainty bands for lower duct Mach numbers are too large to effectively serve as a data check.[5.5]

7.1.1.2 Tailpipe Continuity Checks: Mass flow at the exhaust nozzle can be obtained (for comparison with the facility primary airflow measurement) by measurement of the exhaust gas properties and their numerical integration. This is similar to the in-flight nozzle-exit-traverse method discussed in Paragraph 4.3, but it is accomplished by a more complete survey, just upstream of the nozzle throat, using facility equipment[5.5, 7.1].

The essential components of this method are shown in Figure 7.2 for a single exhaust nozzle. If the engine is equipped with multiple nozzles, the entire system must be repeated for each exhaust nozzle.

The computer software required to support this method must include a flow-field model which will define some flow-field properties not measured directly; the two most important such properties are the static pressure profiles and the flow-field angularity. The nozzle geometry must be defined as a function of engine operating condition.

7.1.1.3 Facility-To-Facility Checks: During most engine performance programs, an engine will be tested in more than one test facility. It is standard practice to compare airflow measurements for these different test installations. Care must be taken to assess any test geometry or condition that will affect engine performance. Some such conditions are differences in:

- Engine inlet-flow pressure and temperature profiles, turbulence, angularity, and humidity
- Large secondary flow effects in the vicinity of the nozzle afterbody
- Instrument sampling rate
- Engine hardware

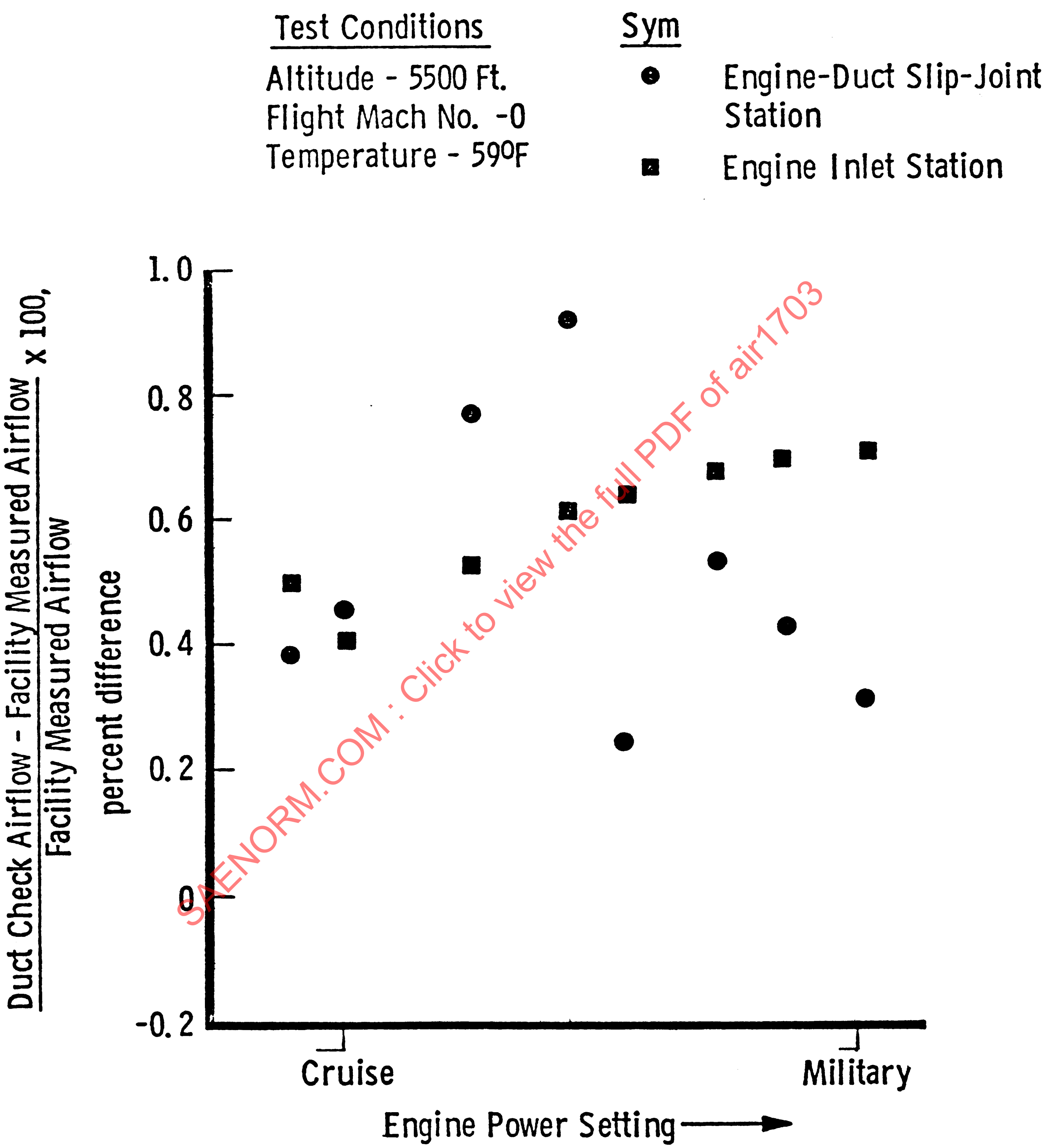


Figure 7.1 - Airflow Duct Check (Nonafterburning Turbojet Engine)

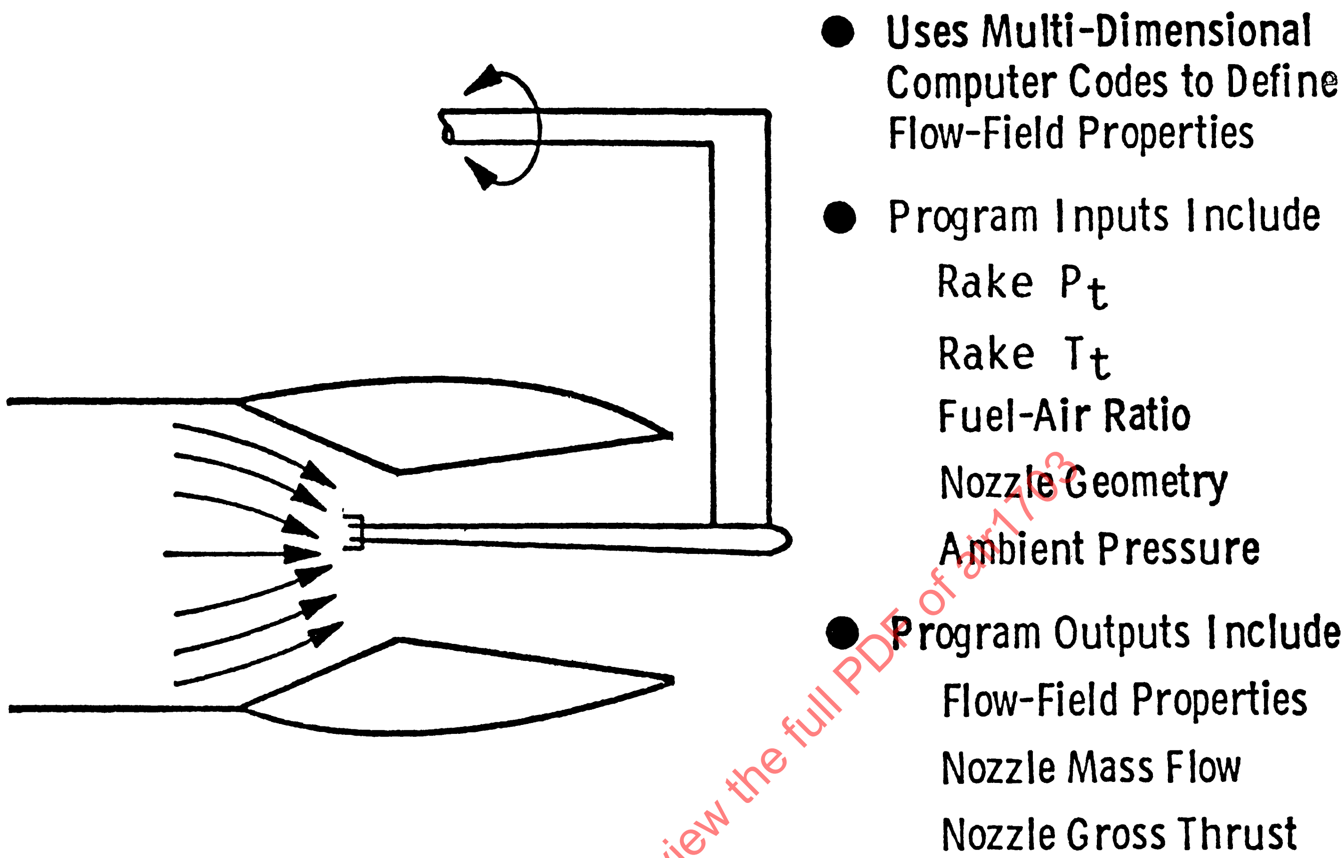


Figure 7.2 - Tailpipe Continuity Check Method (Rake Located at Nozzle Throat)

7.1.1.3 Facility-To-Facility Checks (Cont'd.):

- o Capability to simulate ambient pressures
- o ATF cell heating and recirculation effects.

Any differences must be quantified in terms of engine operational behavior before comparing data. After all engine assessment checks have been completed, the facility-to-facility airflow check should be consistent with the estimated measurement uncertainty bands.

7.1.1.4 Turbine Nozzle Checks: The HP turbine-nozzle flow function, $W_4\sqrt{T_{t4}}/P_{t4}$, or $W_4\sqrt{\theta_{t4}}/\delta_{t4}$, are often used for flow checks, because they will be essentially constant for a choked turbine, independent of the particular test condition. The total temperature and pressure at the HP turbine nozzle is usually calculated from measured values at the compressor discharge, measured fuel flow, and demonstrated combustor performance. Only an engine core-flow check is achieved by this method.

7.1.1.5 Scale-Model Checks: Airflow data from scale-model tests has been used to aid in validating engine airflow data from a ground facility test. However, this presents significant difficulties because of the inherent differences between the scale-model and full-scale tests. It is much more common to use the scale-model data to validate the trends and curve shapes of the engine test data. This is particularly true when exhaust-nozzle flow coefficients are used in the in-flight thrust procedure. In any case, analysis to verify the consistency of the model- and engine-derived coefficients should be completed.

Differences between the model and full-scale coefficients should be consistent with the estimated measurement uncertainty and account for phenomena not simulated in the scale model. Adjustment of the model derived nozzle coefficients will be needed to account for all significant real-engine effects, including steps and gaps, leakage, Reynolds number, surface roughness, instrumentation, gas properties, flow mixing, and swirl. Adjustment due to differences in struts, reverser drag links, and blocker doors may be needed in some installations. Then, any residual differences between model and full-scale coefficients are considered to be due to sampling errors in the measurement of nozzle entry pressure or temperature in the engine.[7.2]

7.1.1.6 Checks Using the Computer-Model Simulation: A digital-computer performance simulation is normally available for all gas turbine engines. As described in Paragraph 4.1.2, these computer models assemble an extensive amount of component data together with the managerial logic necessary to completely simulate the thermodynamic performance of the engine. Normally, the pretest accuracy of the computer-model simulation will not be good enough to use the model as a validity check of the facility measurement system, unless the model has been "trimmed" to a particular engine using data from a ground test facility. Therefore, an important activity during the engine ground testing is to validate the engine computer-model simulation to provide a calibrated math model (Refer to AIR 1678) for in-flight thrust determination. To validate an engine math model, the engine operational envelope is examined in terms of flight Mach number and Reynolds number index.[7.3, 7.4] An output from this validation process identifies the bias differences between the ground test data and the computer model. It may not be practical to validate the model over the complete engine operating envelope, and these regions should be identified.

The differences for many engine performance parameters, not just thrust and mass flow, should be identified. The facility-to-computer-model check is usually conducted after the engine airflow has been validated by the methods discussed previously. Some modifications to the component performance assumptions may result that improve the overall performance simulation. Agreement within the program requirements should exist between the computer-model predictions and the test data, before the data check is considered to be satisfactorily completed.

7.1.2 Thrust Checks: The primary facility measurement used for determination of engine thrust in ground test facilities is the test stand measured force, F_M. The essential characteristics of the engine test facility are that the engine is installed as a free body and that the other forces acting on the external surfaces of the free body may be quantified and summed to provide a determination of the engine gross or net thrust. The engine is mounted in an unrestrained thrust stand, and a load cell is used to measure the net excess force acting on the engine free body. The gross thrust measurement uncertainty range for a ground facility varies depending on the test conditions, engine cycle, and engine geometry. References 5.3 and 5.5 provide some typical magnitudes of measurement uncertainty.

Thrust consistency checks should be preplanned activities to validate both the facility thrust measurement and the preferred in-flight thrust determination method. Some typical thrust check methods are described in more detail.

7.1.2.1 Tailpipe Momentum Checks: The tailpipe momentum check method utilizes the same hardware and computer codes as the tailpipe continuity check (Paragraph 7.1.1.2). The difference in the continuity and momentum check is that the flow properties at the nozzle charging station are integrated for flow impulse in addition to mass flow.

The measurement uncertainty range for the tailpipe momentum check method in the nonaugmented regime of engine operation is basically the same as the primary facility measurement. In the augmented regime, however, the measurement uncertainty of the tailpipe momentum check increases because of the inability to measure directly the local total temperature of the nozzle gas stream.

7.1.2.2 Facility-To-Facility Checks: It is standard practice to compare engine thrust as determined in different ground test facilities. The same attention to differences in test geometry or conditions must be maintained as was discussed for airflow checks (Paragraph 7.1.1.3). When successfully completed, the gross thrust obtained by the primary facility method should be consistent with the estimated uncertainty band.

7.1.2.3 Scale-Model Checks: As is the case for engine airflow (Paragraph 7.1.1.5), scale-model checks are not typically used to validate the primary facility thrust measurement. However, such checks are necessary to validate in-flight thrust methods and, possibly, to extrapolate ground test data.

Scale-model checks compare scale-model nozzle thrust data to full-scale engine thrust. For example, gross thrust from a nonafterburning turbojet obtained from a ground test facility is compared in Figure 7.3 to nozzle thrust from scale-model data [7.1]. In this particular approach to thrust measurement, the gas-path properties at the exhaust nozzle inlet were measured experimentally during engine operation. The

7.1.2.3 Scale-Model Checks (Cont'd.):

measured gas property values were then used in combination with thrust coefficients from duplicate system scale-model tests to obtain a gross thrust value which was independent of most of the parameters used to obtain the gross thrust value by the facility method. Since the difference in thrust measurement is obviously unacceptable for validation of engine gross thrust measurement, the thrust data from

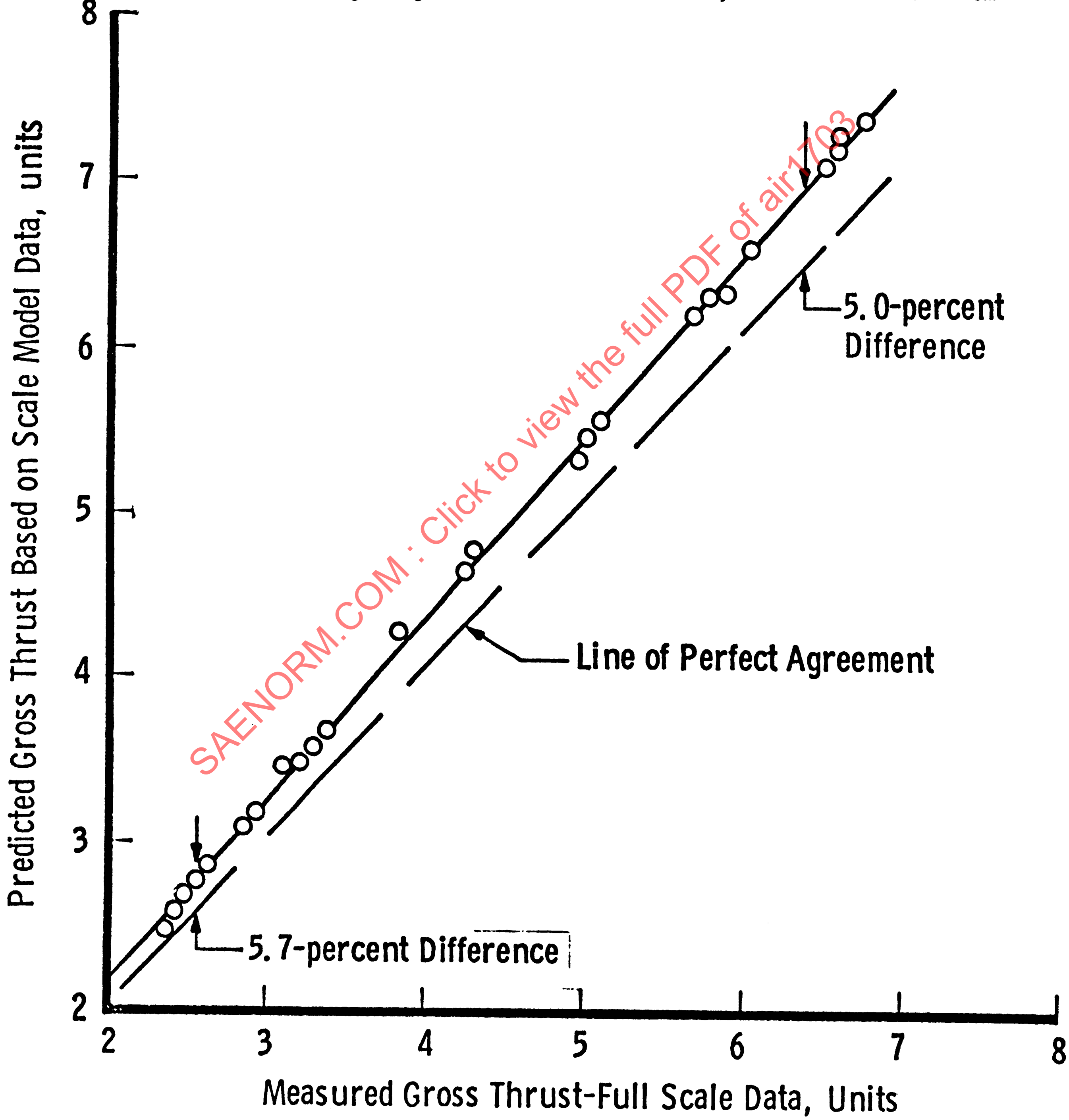


Figure 7.3 - Scale Model Check Thrust Comparison - Initial Results (Nonafterburning Turbojet)

7.1.2.3 Scale-Model Checks (Cont'd.):

these two relatively independent methods were analyzed and compared to quantify the term-by-term differences and determine the cause. The results of this effort are shown in Figure 7.4. When all factors are properly placed into the scale-model check, the results from the two methods agree within one percent. This type of comparison requires a concentrated effort but is necessary to achieve thrust validation.

7.1.2.4 Checks Using the Computer-Model Simulation: All engine thrust checks should include a comparison of facility determined thrust with the output generated by the digital-computer performance simulation. As discussed for the airflow check in Paragraph 7.1.1.6, agreement should exist between the computer-model predictions and the facility test data, before the data check can be considered to be satisfactorily completed.

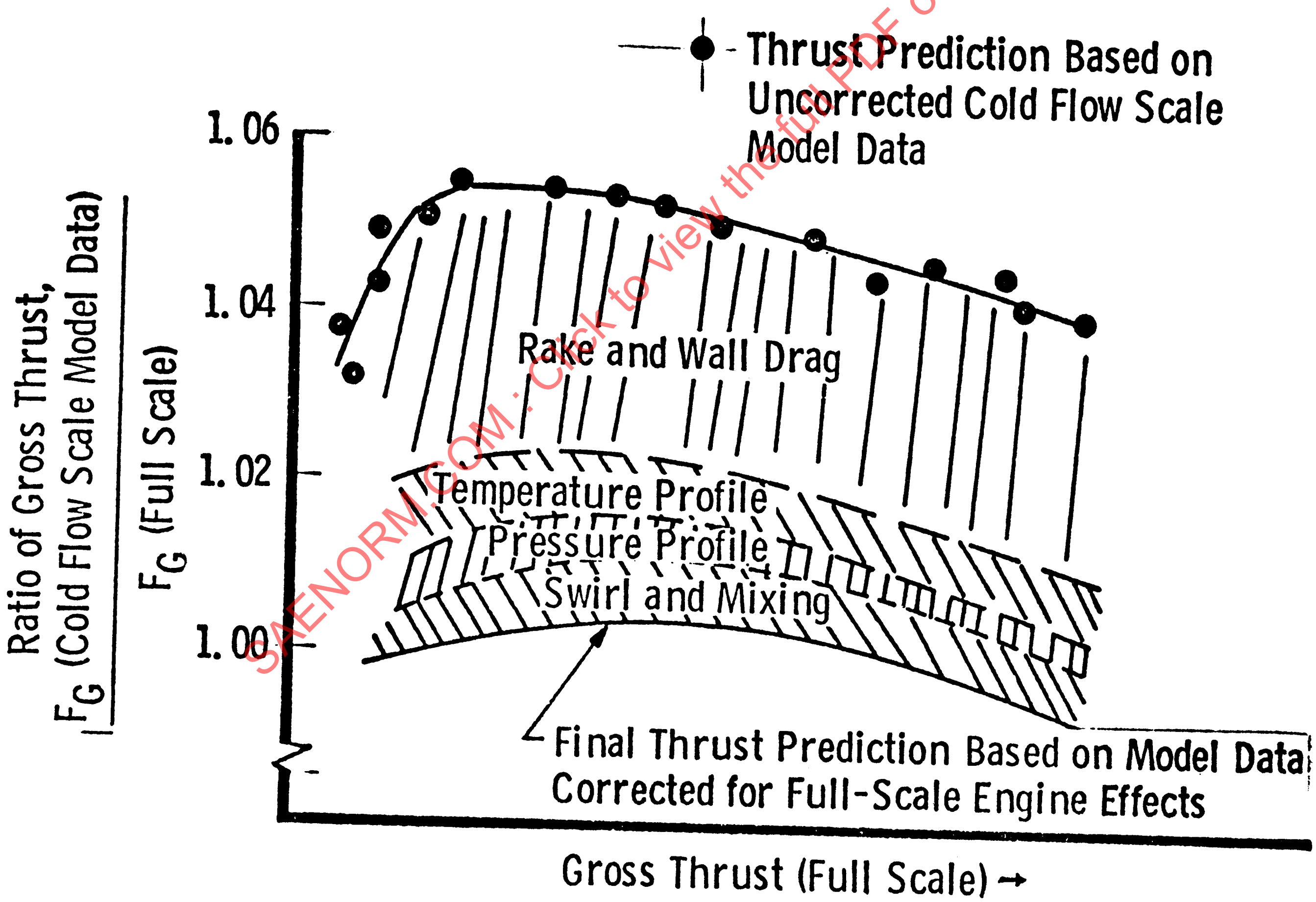


Figure 7.4 - Resolution of Scale Model and Full Scale Engine Thrust (Nonafterburning Turbojet)

7.2 In-Flight Engine Performance: The final definition of installed engine operation is determined in the air vehicle by obtaining engine operating data at both sea level static and flight operating conditions. Prior to the flight test program, the aircraft may be installed on a thrust stand, in the flight configuration, and thrust stand force measurements are obtained over the sea-level-static nozzle pressure ratio range in conjunction with the vehicle instrumentation measurements. Engine consistency checks are performed in a manner similar to the preflight performance checks discussed in Paragraph 7.1.

Specific flight tests are performed to determine and validate installed engine operating characteristics, i.e., thrust, fuel flow, and measured gas generator parameters as functions of engine power setting and flight operating conditions. The type of flight tests and data correlation techniques that are discussed in this section are those required to verify and validate engine calibration information. This information includes Mach number effects, Reynolds number and installation effects, and the effects of varying inlet and engine control schedules.

7.2.1 In-flight Engine Tests: Air vehicle performance data are obtained during a series of flight maneuvers termed steady-state, quasi-steady-state and dynamic maneuvers, as described in Appendix C. Engine characteristics data may be obtained either separately or in conjunction with air vehicle performance tests. The maneuvers currently used are limited to steady-state and quasi-steady-state maneuvers. However, acquisition of engine characteristics data during dynamic aircraft maneuvers, through employment of the techniques described in Appendix B, may become a future consideration.

7.2.1.1 Engine Thrust/Fuel Flow Characteristics - Steady-State: The thrust and fuel flow characteristics for part-power engine operation for single-engine installations and for multi-engine installations, with the restriction of using nearly equal thrust for each engine, are defined by judicious selection of steady-state maneuvers covering the air-vehicle operating envelope (altitude and Mach number), as illustrated in Figure 7.5. The test results can be used for comparison with the predicted model and ATF tests. Ideally, the use of corrected, or referred, parameters tend to collapse the engine characteristics to one line for all altitudes, provided the nozzle is choked. However, Reynolds number effects, which can degrade component efficiency, cause a decrease in overall cycle efficiency with altitude (lower pressure). This manifests itself in requiring greater fuel flow for an equivalent value of thrust as altitude increases. The use of total pressure and total temperature ratios, in lieu of static values for correcting engine parameters, will aid in minimizing altitude effects.

For multi-engine aircraft where symmetric power is not a requirement, it is possible to stabilize the aircraft at a specific steady-state operating condition for a finite period while varying engine thrust from idle through a high power setting. This is accomplished by varying the

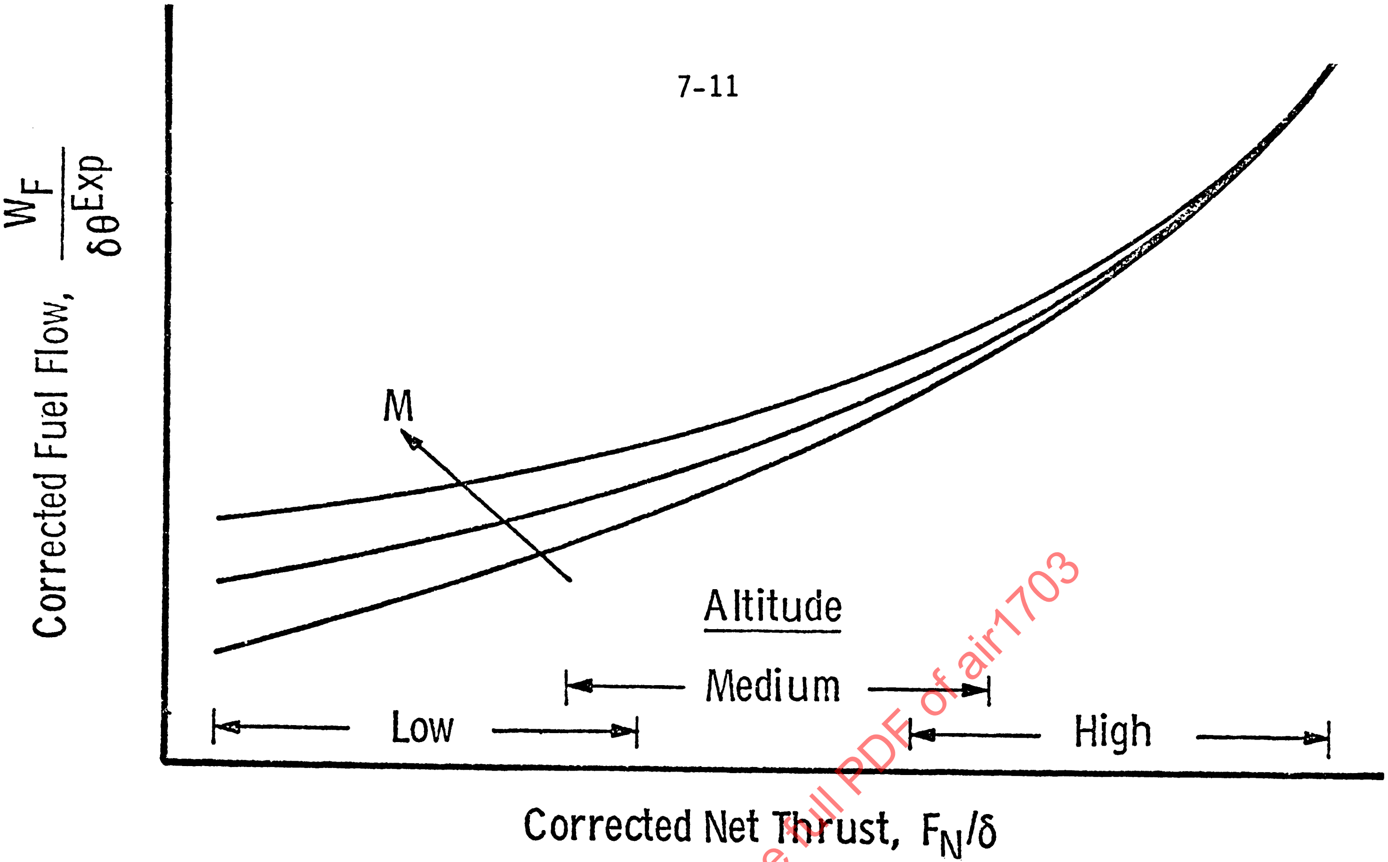


Figure 7.5 - Engine Part-Power Corrected Thrust Versus Corrected Fuel Flow

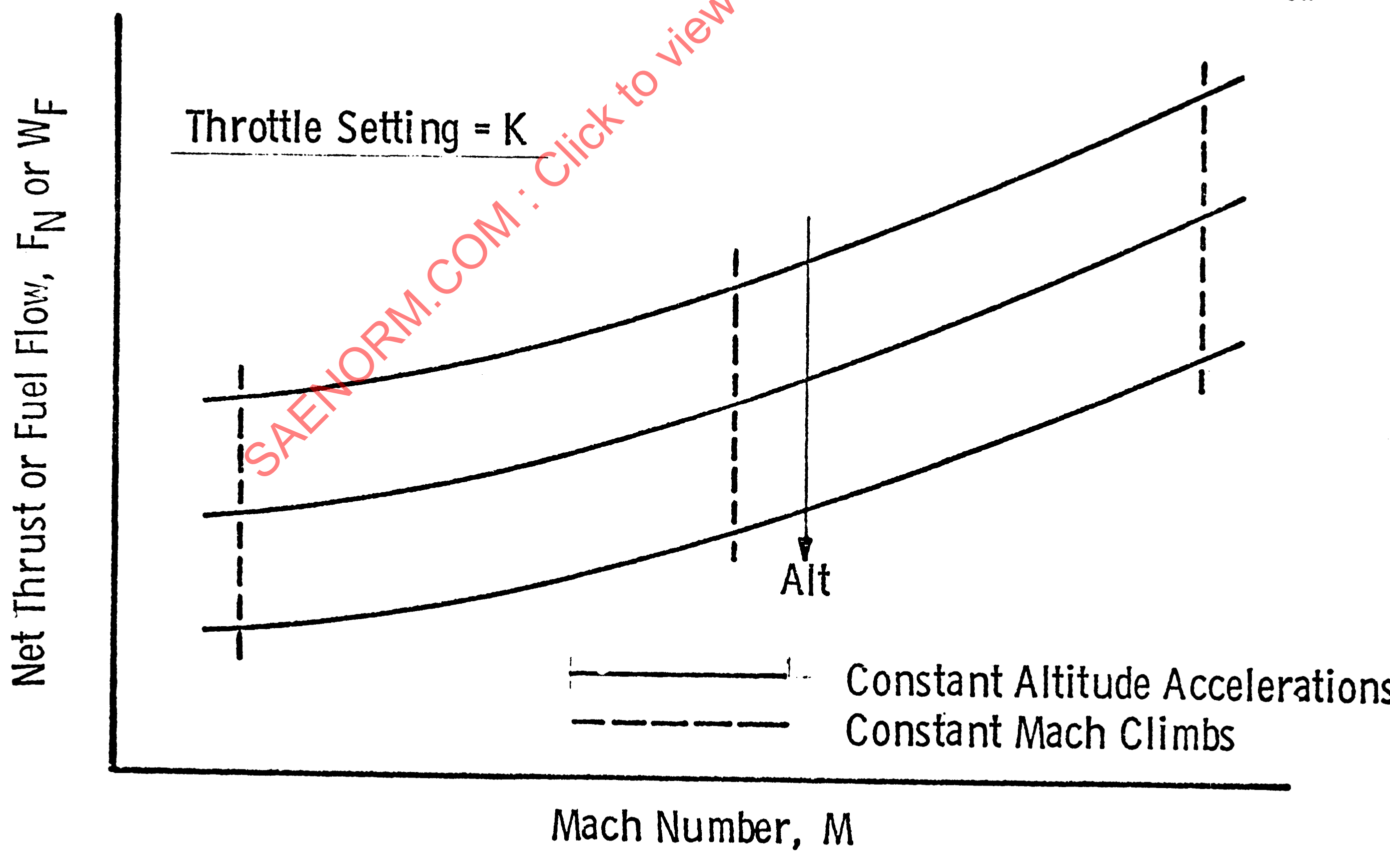


Figure 7.6 - Thrust and Fuel-Flow Definition at a Constant Thrust Setting Using Quasi Steady-State Maneuvers

7.2.1.1 Engine Thrust/Fuel Characteristics - Steady-State (Cont'd.):

thrust on the remaining engine(s) to maintain the desired equilibrium conditions. Use of this approach is restricted to engine installations where external drag effects due to power setting are minimal or are accounted for separately (see Appendix C). The flight test results from this technique at a given altitude define a constant Mach number line in Figure 7.5. The flight test results can be compared with the predicted model or ATF data.

7.2.1.2 Engine Thrust/Fuel Flow Characteristics - Quasi-Steady-State: Constant altitude accelerations and constant Mach numbers climbs (quasi-steady-state maneuvers) provide engine data over a wide range of flight conditions. These maneuvers are performed at constant throttle settings (usually intermediate or maximum afterburning thrust). These engine data expand the range of flight conditions with respect to inlet-temperature and pressure influence obtained over the steady-state engine test data. Figure 7.6 illustrates the thrust and fuel flow information obtainable from these types of quasi steady-state maneuvers. Care must be taken to assure stabilized aircraft and engine operation prior to commencement of data acquisition for quasi-steady-state maneuvers.

7.2.1.3 Engine Operating Characteristics: As many engine operating characteristics as possible should be compared with predictions and/or ATF results to determine whether changes exist that might affect the validity of the calculated in-flight engine thrust values. Flight-to-flight and engine-to-engine data comparisons are also advisable to develop the largest possible data base. The selection of engine variables and the type of comparisons evolve from the engine cycle, installation, thrust measurement methodology, and engine parameter uncertainty values. For example, engine thrust and fuel flow may be evaluated and compared as functions of other independent measurements, such as engine rotor speed, engine pressure ratio, nozzle pressure, etc. Engine measurements other than thrust and fuel flow should also be evaluated for consistency. In addition, data validity checks for error isolation must be developed and be tailored to the in-flight thrust measurement methodology.

7.2.2 Engine Installation Effects: Flight test results include effects on the engine operation due to their installation in the aircraft. Therefore, comparison of the flight results with ATF and model data must be made. A partial list of engine installation effects to be scrutinized include the following:

- Engine accessory horsepower extraction
- Engine compressor bleed airflow
- Inlet airflow pressure recovery
- Inlet airflow distortion

7.2.2 Engine Installation Effects (Cont'd.):

The sensitivity of engine performance to these installation effects is determined from engine test cell measurements or a calibrated computer-model simulation. The effect of each item on the in-flight engine performance is determined from these sensitivity data and estimates or actual flight measurements, i.e., actual bleed rate, horsepower extraction, etc.

7.2.2.1 Engine Horsepower Extraction: When the amount of air-vehicle horsepower extraction is a small proportion of engine power, it can be accounted for analytically. The engine performance sensitivity to horsepower extraction, which is derived from GLTB and ATF tests or calibrated cycle match simulation programs, is used with the amount of horsepower extraction to define the effect of horsepower extraction on installed engine performance.

7.2.2.2 Engine Compressor Bleed Airflow: The best procedure for determining the amount of engine compressor bleed airflow is by measuring the bleed airflow using temperature and pressure instrumentation. Flight and ATF engine characteristics may then be compared directly at equivalent compressor bleed flow rates, as illustrated in Figure 7.7. In-flight engine measurements should also be performed with engine compressor bleed selected both "on" and "off" to verify the compressor bleed flow sensitivity developed in the ATF. Poor correlation of the effects of flight and ATF predicted compressor bleed flow measurements indicates errors in either the ATF sensitivity tests or the flight compressor bleed flow measurements.

Analytical estimates may be made, if in-flight compressor bleed flow measurements are not available. Verification of the estimated amount of bleed flow extraction is acquired through evaluation of in-flight engine measurements with compressor bleed selected "on" and "off", in conjunction with the ATF developed compressor bleed engine performance sensitivities and isolated bleed system flow simulator tests.

7.2.2.3 Inlet Airflow Pressure Recovery: During scale-model tests at representative flight conditions, inlet pressure recovery is measured and evaluated. The engine performance sensitivity to inlet pressure recovery is verified by measuring the engine performance during GLTB and ATF tests with inlet pressure set to simulate the inlet pressure recoveries. Estimated inlet recovery values instead of actual pressure measurements add a small uncertainty in the test results. However, during supersonic flight, the requirement for in-flight pressure recovery measurement becomes a necessity, if aircraft and engine performance are to be accurately defined. Flight-verified model data are generally adequate for follow-on flight tests.

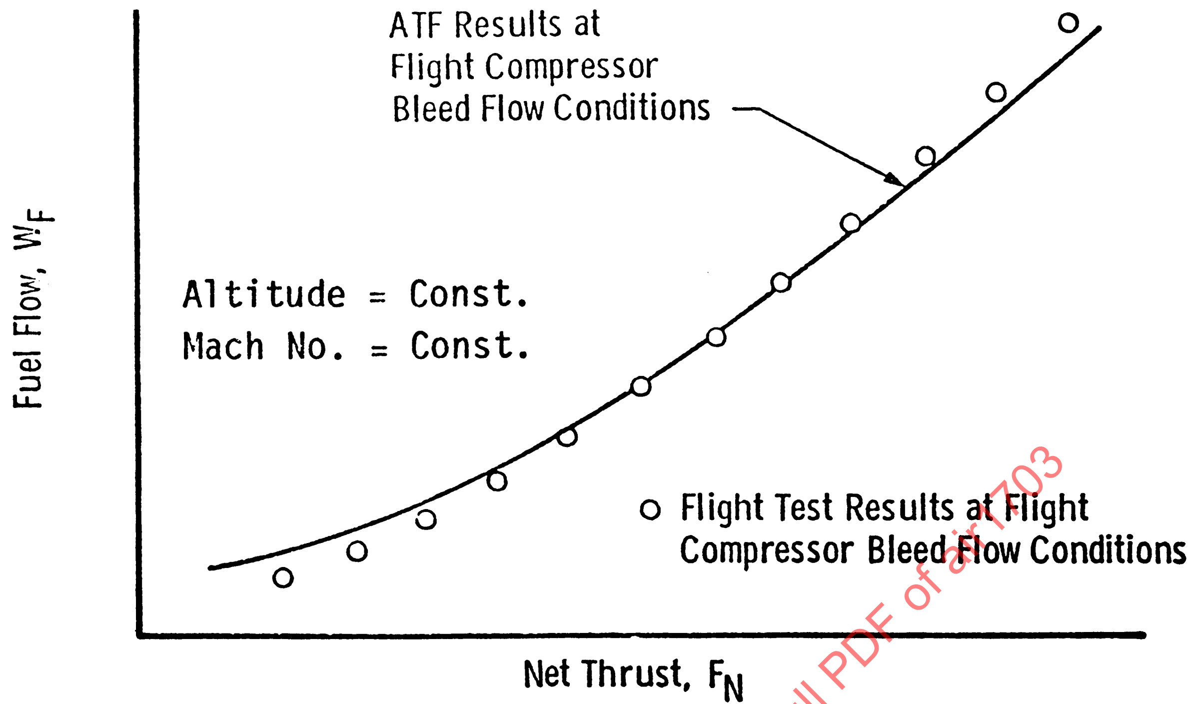


Figure 7.7 - ATF/Flight-Engine Thrust Versus Fuel Flow With Compressor Bleed Airflow

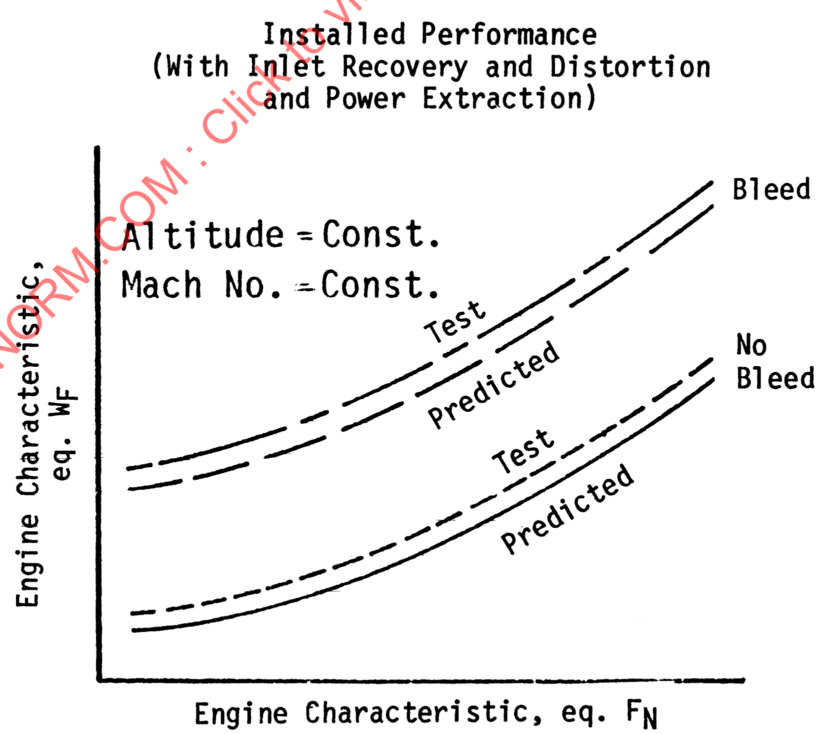


Figure 7.8 - Engine Calibration Characteristics Comparison

7.2.2.4 Inlet Airflow Distortion: In addition to the total pressure loss developed by the inlet, inlet flow may be distorted and affect engine performance. The development of distortion sensitivities is far more complex than for compressor bleed airflow sensitivities and inlet pressure recovery sensitivities (usually one-dimensional parameters), because of the associated multi-dimensional characteristics (circumferential and radial location, magnitude and time dependency). The aerospace industry has devoted considerable effort to developing distortion methodologies. Guidelines to some of these methodologies are provided in ARP 1420 [7.5] and AIR 1419 [7.6].

Static pressure profiles or flow angularity may be as significant as total pressure profiles for some applications.

Sensitivity of engine performance to inlet airflow distortion is measured during GLTB and ATF testing by operating the engine behind distortion screens or other types of distortion simulators. The distortion patterns to be generated by the screens are initially determined analytically and by inlet model testing. These results are often verified by in-flight measurements, using temperature and high-response pressure instrumentation. As in the case of inlet pressure recovery, the necessity for in-flight measurement of distortion is far greater for supersonic operation. Distortion effects on engine performance tend to be minimal for most aircraft in subsonic flight.

7.2.3 Consistency of Installed Net Thrust and Engine Characteristics: A comparison process, such as illustrated in Figure 7.8, is often employed, to compare thrust, fuel flow and engine characteristics for consistency and validation to predictions. The results are analyzed and refined until agreement is reached or differences explained. Areas to be investigated for resolving discrepancies, include:

- In-flight measurement inaccuracies
- Prediction inaccuracies such as errors in model data
- Incorrect engine installation sensitivities
- Incorrect engine installation factors such as magnitude of inlet pressure recovery
- Unaccounted installation factors such as rematching of twin-spool engines between ATF and flight.

Another correlation process is to compare the quasi-steady-state data (climbs and accelerations) to the steady-state engine calibration results. Unaccountable differences may be caused by engine control system lags, such as the failure to keep up with changing inlet temperature and pressure. Analysis of key engine operating parameters will reveal whether or not the apparent differences are due to variations in effective power settings caused by control system lags or are due to other causes. If the

7.2.3 Consistency of Installed Net Thrust and Engine Characteristics (Cont'd.):

gas generator characteristics are not within the expected uncertainty interval at the same flight Mach number, the reason for the difference should be determined and corrected or accounted for in the data analysis.

If the effects of Reynolds number and inlet distortion on gas-generator, ATF, and flight-test results are small, free-stream-suppression can be determined from the gas-generator. The suppression effect on the in-flight nozzle discharge coefficient or overall performance correlation must be included in the in-flight thrust-determination procedure. A comparison of ATF and flight data will determine whether differences are consistent with nozzle-area suppression effects determined from model test.

7.2.4 Summary: In-flight engine performance analysis is enhanced by proper planning of the flight test program and by the correlation and validation of various test results. All tests prior to the actual flight tests (scale model and engine GLTB and ATF tests) must be planned and conducted in a manner to produce data of maximum value to the flight test program. The engine calibration data from these tests may produce families of coefficients or calibration curves, as functions of altitude, that are due to aerodynamic and mechanical effects, such as a loss variation with Reynolds number or a variation in jet-nozzle area with pressure loading. For a high-bypass-ratio engine, families of coefficients, as a function of flight Mach number, can occur due to fan-exhaust pressure and temperature profiles resulting from fan-operating-line shifts which occur as the nozzle pressure ratio changes from an unchoked condition. For this case, the changes in the nozzle coefficients are partly real, due to profile effects on nozzle and scrubbing losses, and partly apparent, due to a difference in the flow-sampling error of the different profiles. Families of coefficients with flight Mach number can also occur if the environmental pressure, to which the nozzle flow expands, changes as conditions in the adjacent nozzle flow stream change. In any event, sufficient analysis should be conducted to explain the correlations.

Engine installation sensitivities from these tests should be defined in a manner such that the effects of differences between actual installation factors and their predicted values can be adequately assessed. The flight test program must be planned so sufficient data are acquired to enable cross-checking and validation of the engine performance and operating parameters.

7.2.4 Summary (Cont'd.):

Overall performance methods can be developed from correlations of measured thrust with measured gas-generator parameters. Part of the recommended process for ensuring accurate in-flight thrust is a consistency-analysis of gas-generator data from all sources effectively producing an overall-performance method for each compared gas-generator parameter. The most sophisticated, overall-performance method is the cycle-match, computer simulation. The simulation combines the gas-path/nozzle and overall-performance methods into a single bookkeeping system which is internally consistent and satisfies the mass-, energy-, and momentum-conservation laws, and the laws of thermodynamics. The process for ensuring accurate, in-flight thrust involves continuous development of the cycle-match computer simulation within an acceptable and accountable tolerance, using data from all sources.

7.3 Aerodynamic Characteristics: Analysis of the flight test aerodynamic data is useful in identifying trends that are symptomatic of errors or inaccuracies within the various elements of the thrust/drag measurement process. These errors may be evidenced by a disagreement in the level of total air vehicle drag between one or more types of maneuvers or flights of the same aerodynamic configuration, or through erratic representation of the air vehicle induced drag and/or compressibility (Mach) characteristics. Evaluation of the aerodynamic characteristics for this purpose requires an understanding of the theoretical aspects of air vehicle aerodynamics, the in-flight drag measurement processes and techniques, and the isolation and validation of the incremental drags used in "normalization" of the air vehicle drag polar. Appendix C contains an overview of these items.

The examples in the following section include the concepts of thrust validation through proper definition of air vehicle aerodynamic characteristics.

7.4 Examples of Test Analysis: The examples of test analysis for thrust validation are a continuation of the examples in Paragraphs 4.5 and 5.1.6 for a single-exhaust turbofan, an intermediate cowl turbofan, and a mixed-flow afterburning turbofan. The examples discuss and illustrate many of the facets of preflight engine consistency checks, flight engine performance analysis, and definition of aerodynamic characteristics required to enhance confidence in air vehicle in-flight thrust measurements. The examples are taken from in-flight thrust measurement programs designed to isolate and document air vehicle aerodynamic characteristics. The examples are taken from both published (7.7 through 7.13) and unpublished sources.

7.4.1 Single-Exhaust Turbofan: A series of in-flight thrust measurement programs have incorporated the single-exhaust turbofan engine installation.

A significant aspect of these programs was that the in-flight thrust measurement methodologies involved the calibration in an ATF, prior to the conduct of the flight tests, of each engine used for air vehicle perfor-

7.4.1 Single-Exhaust Turbofan (Cont'd.):

mance measurements. The ATF calibration conditions (altitude, Mach number and power setting) were designed to encompass the anticipated flight conditions for each individual test vehicle. Therefore, the in-flight engine performance measurements were tailored to and based on the individual engine ATF calibration information. The examples develop and correlate the results of three independent (or alternate) methods for in-flight thrust determination. All three methods deduce engine airflow from correlation of engine rotor speed with ATF facility airflow measurements, but each uses a different gross thrust correlation procedure.

7.4.1.1 Gross Thrust Correlation With Nozzle Pressure Ratio: Initial full-scale wind tunnel and flight propulsion/aerodynamic measurements under this program resulted in development of the nozzle pressure ratio (NPR) method. The application of this method during the full-scale wind tunnel test program provided no particular problems.[7.7] The initial flight test application, however, produced a difference in drag polar definition for two successive flights with the same air vehicle aerodynamic configuration, as shown for the vehicle No. 2 flight data in Figure 7.9. Even though initial evaluation of the vehicle No. 2 engine ATF and flight measurements revealed no obvious discrepancies which would account for the different results between flights, suspicion of error(s) in the in-flight thrust measurement process existed. Isolation and correction of this error required analysis of the flight-test engine fuel flow measurements discussed in the following paragraph.

7.4.1.2 Gross Thrust Correlation With Fuel Flow: The rationale for the change in drag characteristics observed on the first flight of vehicle No. 2 was not resolved until the air-vehicle specific range data from in-flight measurement of fuel flow and true airspeed were reviewed. Almost exact agreement in specific range values existed for test conditions common to the two flights of vehicle No. 2 (circled points in Figure 7.10). The ATF engine calibrations for the two engines showed virtually identical uninstalled thrust/fuel flow characteristics, and no known changes affecting drag were made to the vehicle between the two flights. It was concluded that the drag characteristics on the two flights must be identical. This information prompted a thorough re-examination of all instrumentation parameters involved in the in-flight thrust calculation procedures. The investigation revealed an unexplained calibration shift on the first flight of vehicle No. 2 in either or both of the engine fan or core discharge pressure transducers used to measure the manifolded total pressure at the primary/bypass duct mixing station. In this in-flight thrust measurement program, redundancy of methods was provided by the engine ATF calibrations

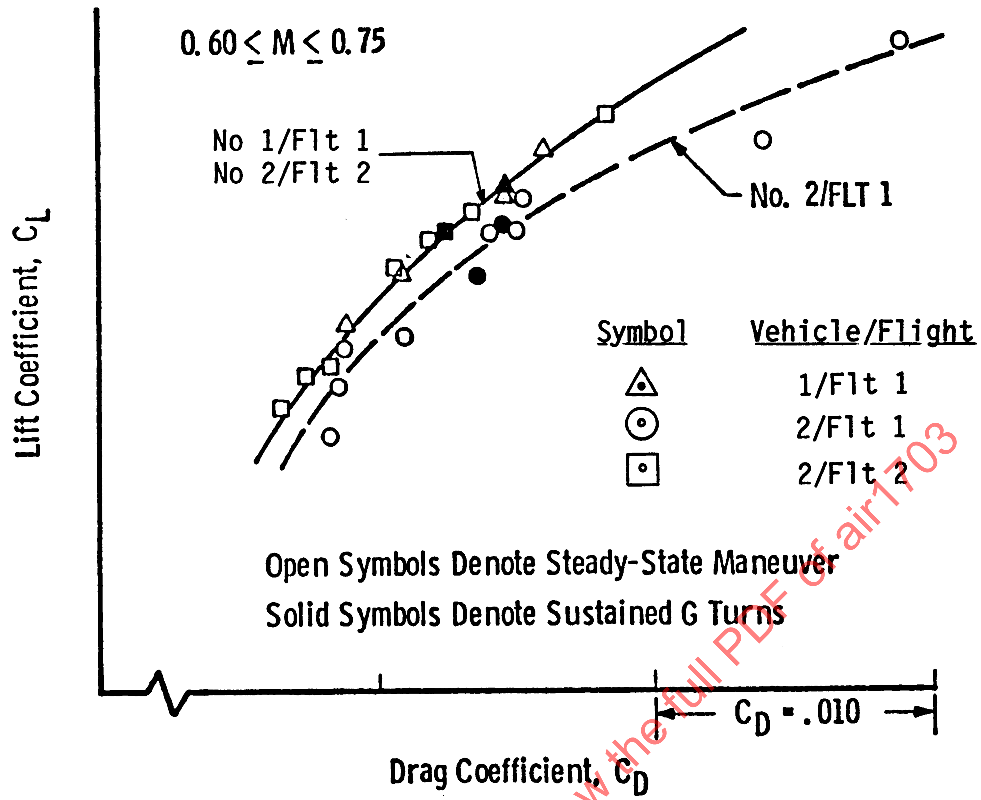


Figure 7.9 - Flight Test Drag Characteristics

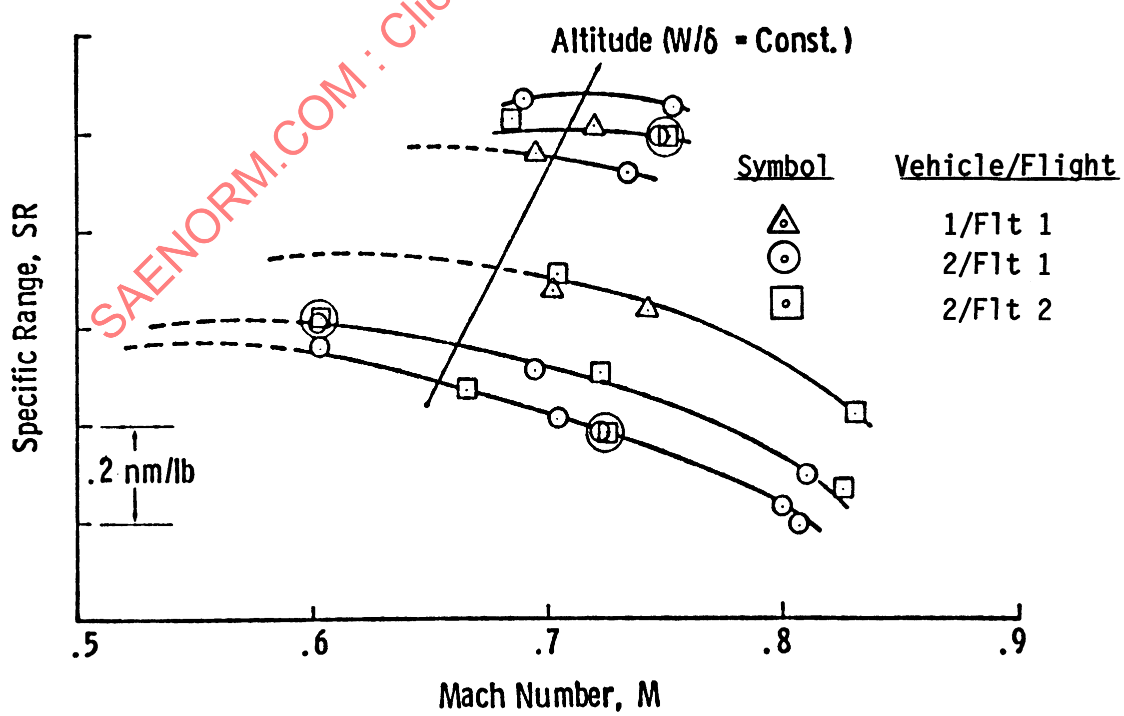


Figure 7.10 - Flight Test Specific Range

7.4.1.2 Gross Thrust Correlation With Fuel Flow (Cont'd.):

defining the uninstalled engine thrust/fuel flow characteristics. Therefore, the in-flight thrust and drag data from the first flight of vehicle No. 2 were salvaged by adjusting the ATF calibration thrust/fuel flow characteristics for the installation effects defined on the second flight of vehicle No. 2 (primarily for the effect of inlet pressure recovery) and by extracting the flight installed engine thrust using the in-flight measured fuel flows. This process brought all aerodynamic drag data into agreement, and enhanced confidence in the accuracy of the in-flight thrust measurements.

7.4.1.3 Gross Thrust Correlation With Gross Thrust Parameter: A later development program, which involved moderate changes to the vehicle aerodynamic configuration, required a redefinition of the vehicle aerodynamic characteristics to validate performance estimates. The development of the gross thrust parameter (GTP) in-flight thrust method during this program and correlation with the NPR and fuel flow methods is discussed herein.

Prior to commencement of the flight test program, all three approaches to gross thrust calculation (NPR, fuel flow and GTP) were subjected to an extensive error analysis to develop an estimate of total uncertainty of flight test derived lift and drag parameters.[7.10] The error analysis revealed that the three in-flight thrust measurement processes were essentially equal in accuracy. A slight advantage was shown for the NPR method, which was selected as the preferred method because it had been successfully employed in the original flight test program.

After review of the initial flight test results, however, excessive data scatter caused by instrumentation problems with the NPR method resulted in selection of GTP as the primary thrust method. GTP was preferred over the fuel flow method because a single primary variable (engine rotor speed) is required for gross thrust determination. The fuel flow method requires determination of three primary variables (fuel density, fuel viscosity and fuel volume flow) for gross thrust determination.

Figure 7.11 is a comparison of installed propulsive force (F_{IPF}) variation with time during a typical steady-state maneuver for the three in-flight thrust measurement procedures. The thrust data scatter observed with the NPR method is 5 to 6 percent and was attributed to electrical difficulties in the acquisition of the engine fan and core discharge pressure transducer signals. While this level of the data scatter is excessive, the average thrust values were found to be in close agreement with the values obtained with the GTP and fuel flow methods.

The gross thrust data for the GTP method resulted in two F_{IPF} levels representing a difference of approximately two percent, as shown in Figure 7.11. The thrust level changes were traced to high-pressure rotor speed shifts attendant to the least significant bit of the data

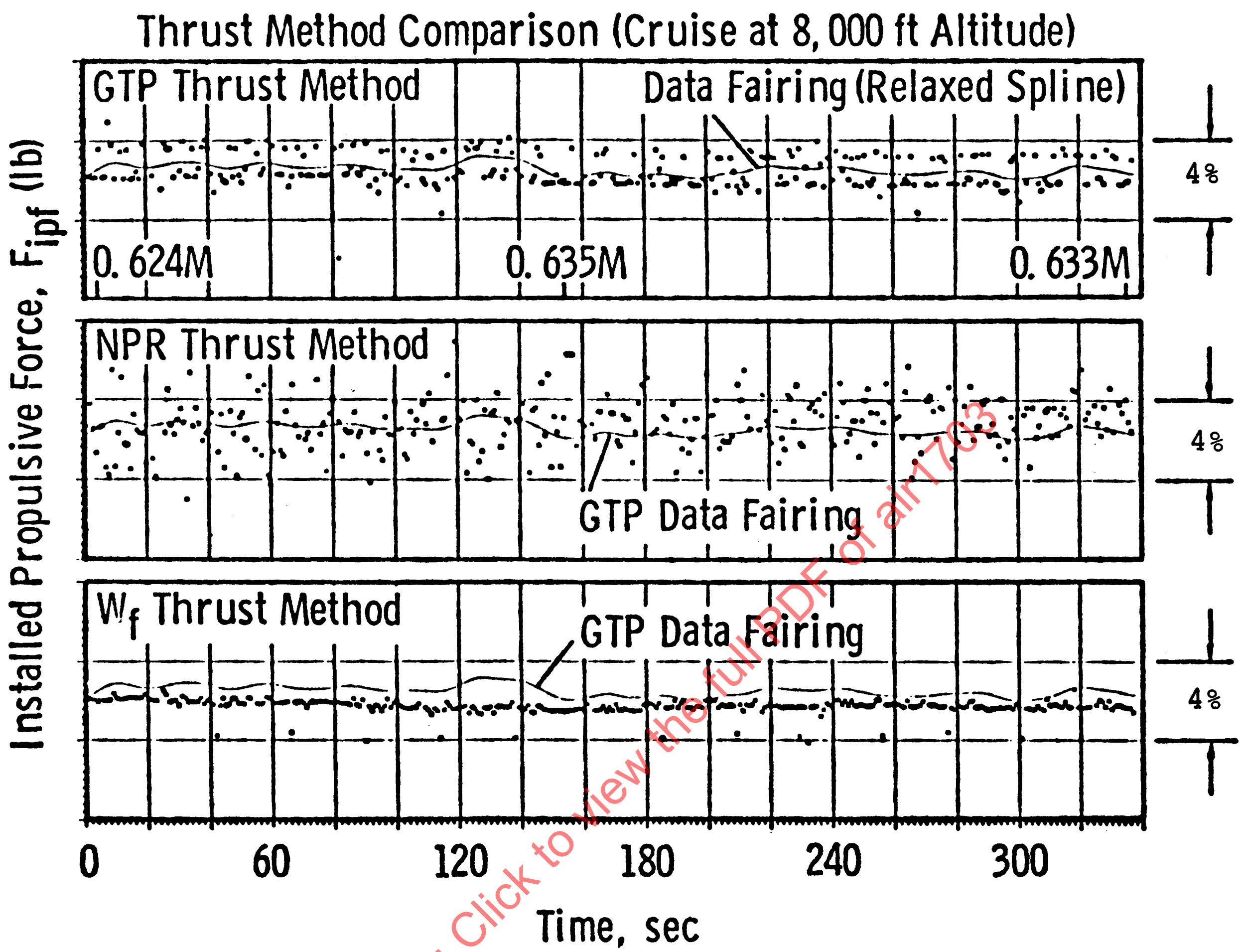


Figure 7.11 - Net Propulsive Force Variation During a Typical Steady-State Flight Maneuver

7.4.1.3 <u>Gross Thrust Correlation With Gross Thrust Parameter</u> (Cont'd.):

transmission link. This problem was alleviated using a relaxed spline-curve-fit procedure to obtain representative F_{IPF} values (weighted average based on data density).

The flight test drag polar derived with the selected GTP thrust method is presented in Figure 7.12 and is compared with the drag polars obtained using the NPR and fuel flow thrust methods. The drag polar is repeated (with the drag coefficient scale staggered) for comparison with the test data points derived with the NPR and fuel flow thrust determination methods. A low level of data scatter is observed in the drag polar characteristics obtained with the GTP thrust method. Ninety percent of the data points for the Mach number range of 0.59 to 0.75 fall within a scatter band of ± 1.5 percent relative to the faired drag polar. Although the data scatter is greater for the other two thrust methods, good agreement exists in the defined drag polar characteristics.

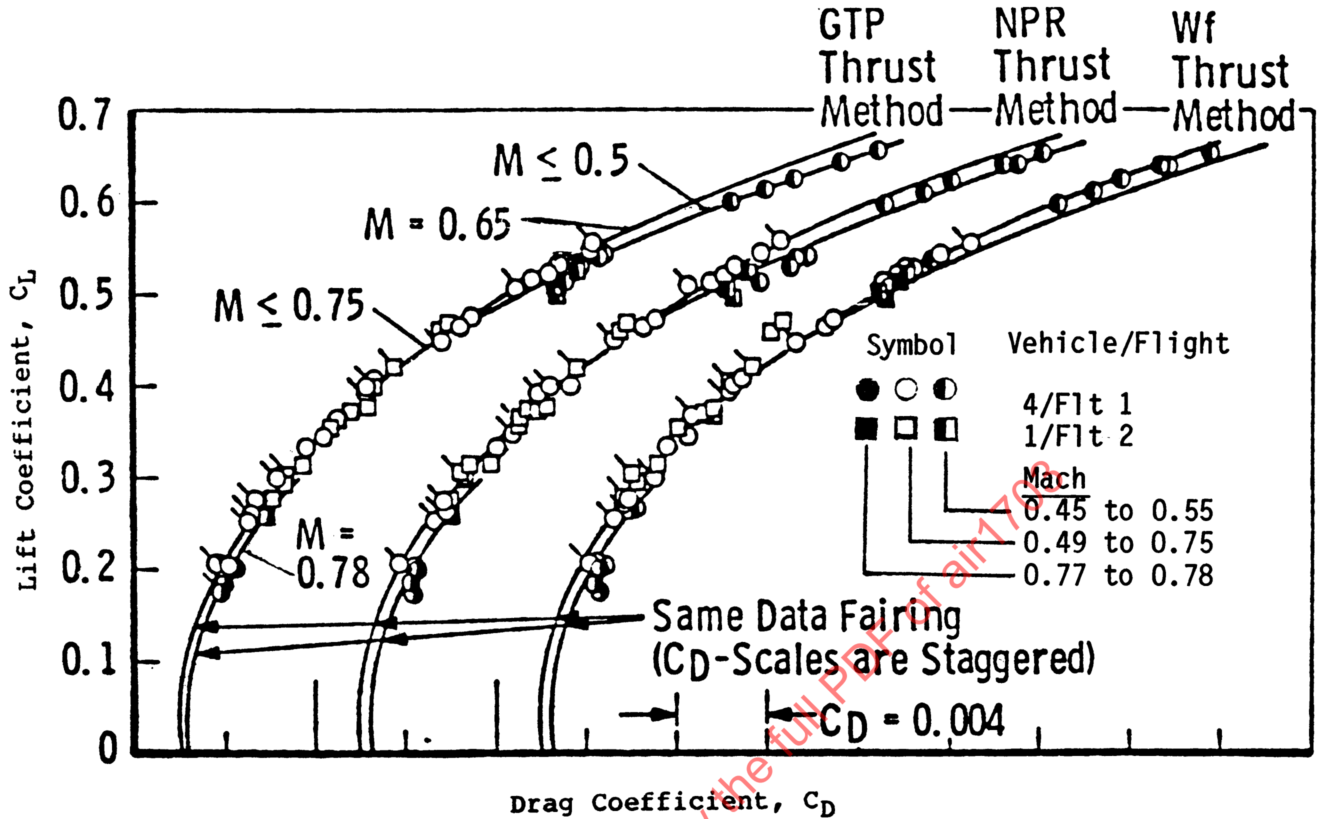


Figure 7.12 - Flight Test Drag Variation with Lift - Thrust Method Comparison

7.4.1.4 Throttle-Dependent Force Increments: The throttle-dependent force increments used for flight-test F_{IPF} and drag normalization for both in-flight thrust measurement programs were based primarily on wind tunnel information. The subcritical inlet spillage drag variation with inlet-mass-flow ratio and Mach number was obtained with a 1/3-scale aerodynamic model.

The boattail-base-drag variation with nozzle pressure ratio and Mach number were derived using a 1/3-scale powered nozzle/afterbody model. The subscale boattail-base-drag corrections were verified by pressure/area integration of boattail-base surface pressures measured during full-scale wind tunnel tests. Additional confidence in the validity of the boattail-base drag increments is achieved because of correlation in nozzle exit static pressure measurements between the flight and the full-scale wind tunnel tests.

7.4.1.5 Post-Test Evaluation of the Uncertainty Analysis: A pre-test uncertainty analysis had been conducted to evaluate the measurement process and aid in selecting preferred in-flight thrust methods.(7.12) The major sources of predicted uncertainty for in-flight thrust were the engine calibration information and the data resolution from the pulse code modulation system for transmitting engine speed. The later problem was alleviated using a relaxed spline-curve-fit procedure. The post-test evidence available from engine calibrations and validation of air vehicle aerodynamic characteristics did not appear to support as

7.4.1.5 Post-Test Evaluation of the Uncertainty Analysis (Cont'd.):

large an uncertainty band as predicted during the pre-test analysis. The extensive data included:

- Approximately 20 engine calibrations from two different ATF's
- Calibration of one engine in two ATF's and a GLTB[7.13]
- Correlation of air-vehicle drag for two completely separate in-flight thrust measurement programs.[7.9]

A review indicated that the pre-test uncertainty estimates were obtained using a limited data base of uncertainty information. At the time of the post-test review additional information was available to improve the instrumentation uncertainty assumptions. When utilized to re-estimate the overall uncertainty, the resulting uncertainty bands were significantly lower than the pre-test estimate.

7.4.2 Intermediate-Cowl Turbofan: This in-flight thrust measurement example is derived from a multi-engine transport aircraft incorporating nacelle-mounted intermediate-cowl turbofans. The procedure is based on nozzle coefficient information from scale-model, GLTB and ATF tests.

7.4.2.1 In-Flight Thrust Validation: Independent checks of the in-flight thrust calculation procedures were obtained through airplane ground-level thrust stand measurements and engine airflow correlations. The first check was obtained by measurement of static thrust on a thrust stand and comparison of the result, on a point by point basis, with the calculated thrust using the in-flight thrust calculation procedure. The results of this process are shown in Figure 7.13 for the average measured and calculated engine thrust. The agreement is better than ±1 percent of average engine thrust. This procedure is useful in resolving instrumentation and gross thrust determination methodology problems.

The second check compared in-flight engine airflow obtained from two independent sources. Inlet airflow was determined from a survey of the inlet flow using an inlet rake to measure both total and static pressures and total temperatures. Airflow was also determined throughout the program by the summation of engine nozzle flows from measured nozzle pressures and temperatures. Figure 7.14 shows excellent agreement in the airflow obtained from these two sources. This agreement adds confidence to the validity of the nozzle coefficients, which are used for both airflow and gross thrust calculations.

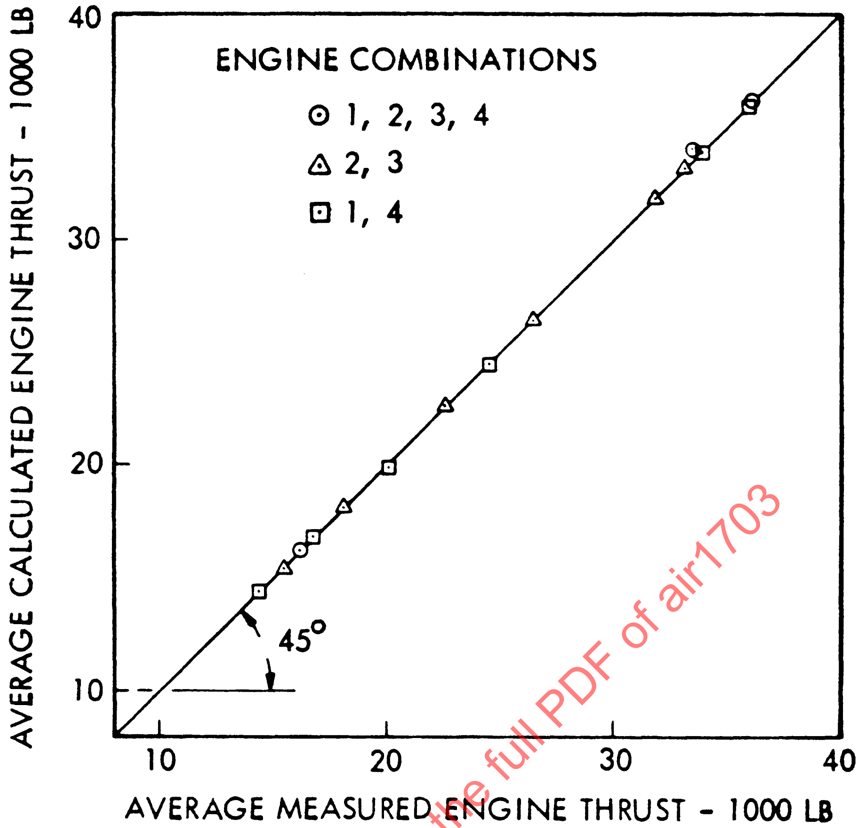


Figure 7.13 - Thrust Stand Measured and Calculated Static Thrust Comparison

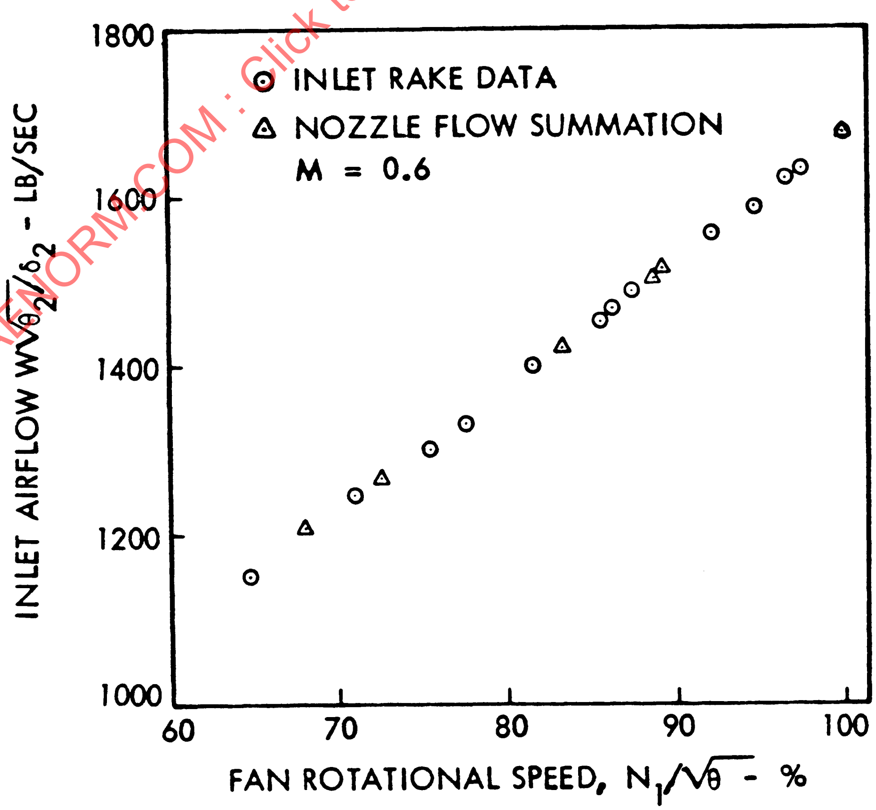


Figure 7.14 - Flight Test Comparison of Inlet Flow Calculations

7.4.2.2 Aerodynamic Characteristics: The aircraft drag polars generated from the in-flight thrust measurements are presented in Figure 7.15. Confidence in the powerplant thrust measurements is achieved because of the consistency in definition of the drag characteristics as a function of lift coefficient and Mach number. The scatter is ±3.5 percent for vehicle minimum drag with drag-due-to-lift removed as discussed in Appendix C.

7.4.2.3 Validation of Throttle-Dependent Force Increments: The computation of installed propulsive force for this aerodynamic and engine configuration comprises a synthesis of engine-airflow, net-thrust, and incremental-drag-force calculations that have been determined from the results of both model and full-scale test programs. Validation of the effects of incremental propulsion related forces (pylon and core cowl scrubbing drag, plug scrubbing drag, and nacelle forebody and afterbody pressure drags) would be difficult even with extensive pressure instrumentation and was not attempted during this flight test program. However, some confidence in the validity of the throttle-dependent drag increments can be achieved by evaluation of the consistency in calculated aerodynamic drag, which incorporates the empirical throttle related forces. This may be accomplished by comparing the normalized drag results from maneuvers using diverse engine thrust operating conditions, such as climbs, cruise and idle descents. For the example program, the data scatter for climbs and descents is ±2 percent about the drag level for cruise flight, as shown by the solid line in Figure 7.16. For the conditions evaluated, no discernible bias differences exist that might be attributable to an inaccurate accounting of the throttle dependent propulsion forces.

7.4.3 Mixed-Flow Afterburning Turbofan: The mixed-flow afterburning turbofan example pertains to a twin-engine aircraft incorporating a fully integrated propulsion system. The thrust measurement procedures are based on engine nozzle coefficients and airflow calibration data from testing of four separate engines in an ATF. Generic calibration data are used with uncalibrated engines incorporating the required instrumentation to compute in-flight thrust. Two in-flight thrust measurement procedures, termed pressure-area and airflow-temperature, were developed. Due to differences in in-flight thrust from these two procedures, effort was expended to understand the reasons, as discussed below.

7.4.3.1 In-Flight Thrust Validation Checks: Each data set was thoroughly scrutinized for normal engine operating characteristics. Evaluation of the parameters shown in Figure 7.17 (engine TSFC; fuel flow, nozzle entry pressures, and nozzle area measurement) was made to assure that all fell within expected tolerances based on historical tracking of these parameters. Any deviation from expected trends was thoroughly evaluated for its potential impact on the validity of the data set. Particular attention was given to in-flight measurement and definition of installation losses (primarily, inlet pressure recovery and compressor bleed airflow).

NOTE:
SOLID SYMBOLS ARE FOR M_{TEST} VALUES NEAREST TO NOMINAL MACH NOS.

OPEN SYMBOLS ARE POINTS CORRECTED TO NEAREST MACH NOS. ABOVE AND BELOW M_{TEST}

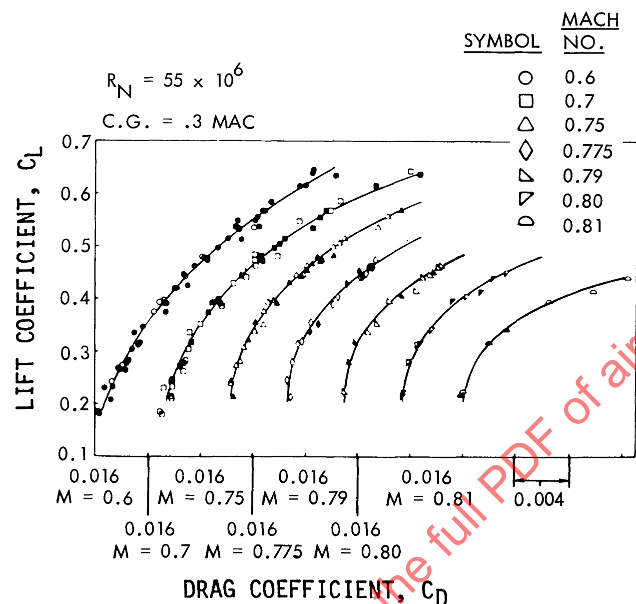


Figure 7.15 - Flight Test Drag Characteristics

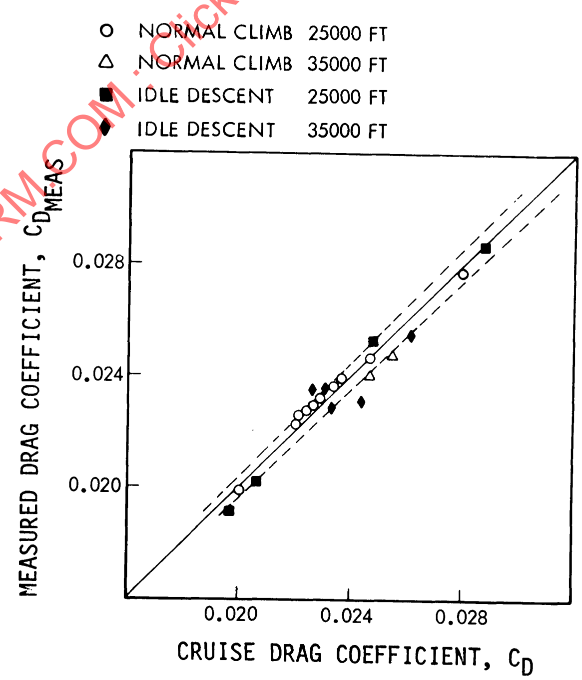


Figure 7.16 - Flight Test Drag Comparison Various Engine Thrust Operating Conditions

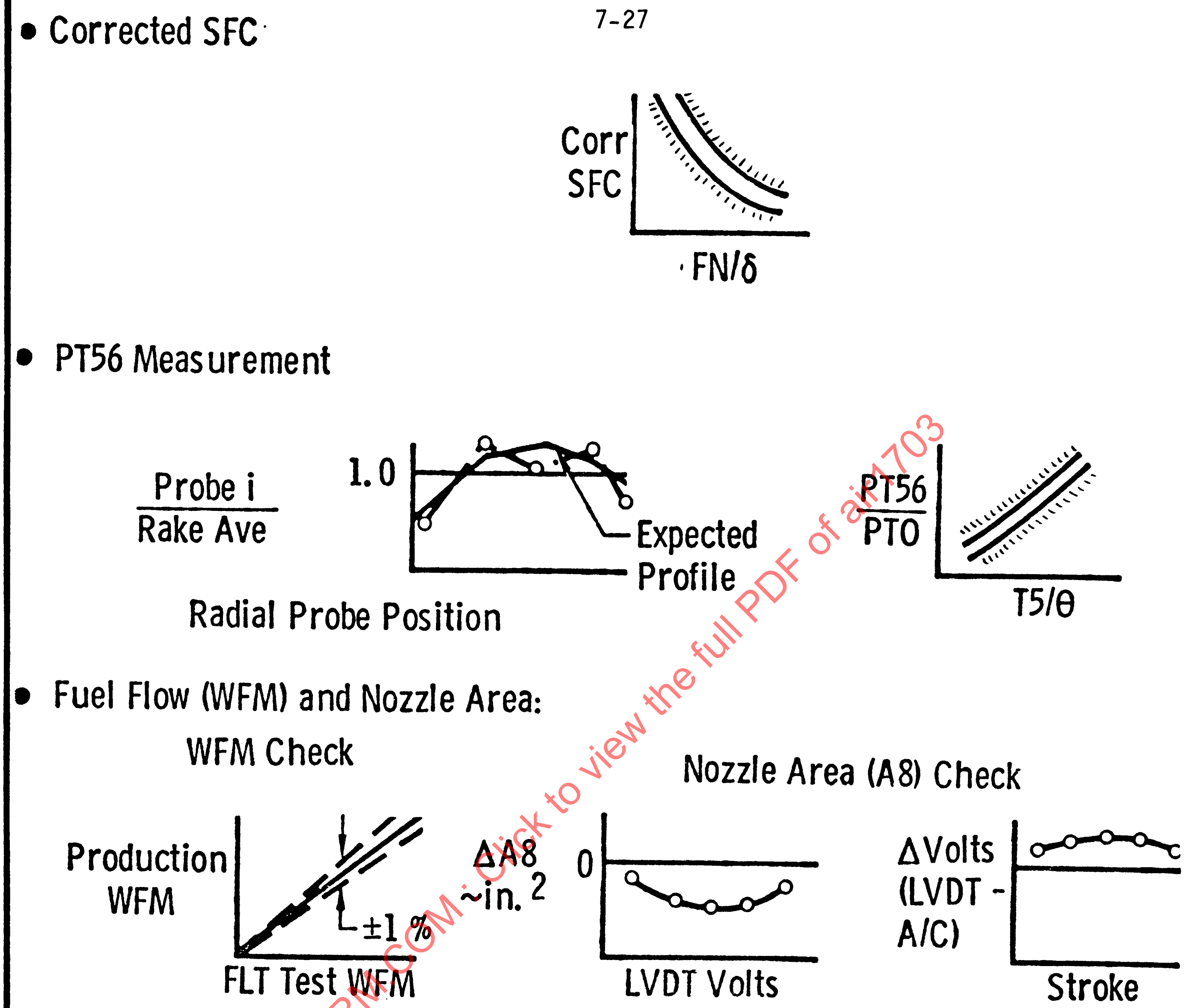


Figure 7.17 - Data Validity Checks - Steady-State Maneuvers

7.4.3.2 Area-Pressure In-Flight Thrust Method: Early air vehicle aerodynamic characteristics evaluated using this approach would not generalize as a function of engine thrust level (correlation in drag for steady-state and quasi-steady-state maneuvering could not be established). Investigation of the error sources in this method involved expansion of the engine performance data base from two to four engines, including an engine removed from a flight test aircraft. Incorporation of the refined pressure-area method based on the expanded ATF engine sample is shown in Figure 7.18 and compared to the steady-state drag polar defined by the airflow-temperature thrust measurement method. The data indicate good agreement between steady-state and acceleration data at medium and high lift coefficients. At low lift coefficients, scatter in the acceleration data prevents accurate definition of the air vehicle

7.4.3.2 Area-Pressure In-Flight Thrust Method (Cont'd.):

minimum drag level. This disparity could be caused by uncertainty in turbine discharge pressure measurements at high Mach number, low altitude conditions (upper end of instrumentation pressure range limit). The drag values for the 10,000-ft steady-state data (shaded squares in Figure 7.18) are questionable because of uncertainty in nozzle area measurements caused by disengagement of the nozzle flap actuation rollers from the roller cam surface at low pressure ratios (NPR less than 2.5). The data are included for comparison with the airflow-temperature method in Figure 7.19.

7.4.3.3 Airflow-Temperature In-Flight Thrust Method: Investigation of the error sources in the pressure-area in-flight thrust method was facilitated through use of an alternate in-flight thrust method which did not require accurate measurement of the nozzle area, a critical parameter for accurate pressure-area thrust measurement. The alternate method was based upon fan corrected airflow correlations, fuel flow measurements, and heat balance calculations to determine nozzle temperature and airflow. The results of this analysis, utilizing the same data sets as presented for the pressure-area method in Figure 7.18, are presented in Figure 7.19. The results show excellent agreement between the steady-state and low and high altitude quasi-steady-state (acceleration) maneuvers at all lift coefficients. However, considerable data scatter is introduced at higher lift coefficients when the low altitude maximum thrust accelerations are included. One possible explanation is that the airflow-temperature method (particularly the tailpipe temperature model) may be inaccurate due to the limited ATF test data available pertaining to these low-altitude operating conditions.

7.4.3.4 Validation of Throttle Dependent Forces: The subcritical inlet spillage drag variation with inlet mass-flow-ratio and Mach number was derived from a subscale aerodynamic model. These test results were assumed to be applicable to the full-scale vehicle. The nozzle/afterbody drag variation with nozzle pressure ratio and nozzle exit area were obtained from a subscale powered nozzle/afterbody model. Validation of the throttle dependent nozzle/afterbody force increments was accomplished through correlation of static surface pressures between the subscale model and the flight vehicle, as discussed in Appendix C. Correlation of these pressures at selected longitudinal locations is shown in Figure 7.20. The correlations are in good agreement, except where influenced by discontinuities on the flight article, and provide confidence in the accountability for the throttle-dependent forces acting in the vicinity of the nozzle and afterbody.

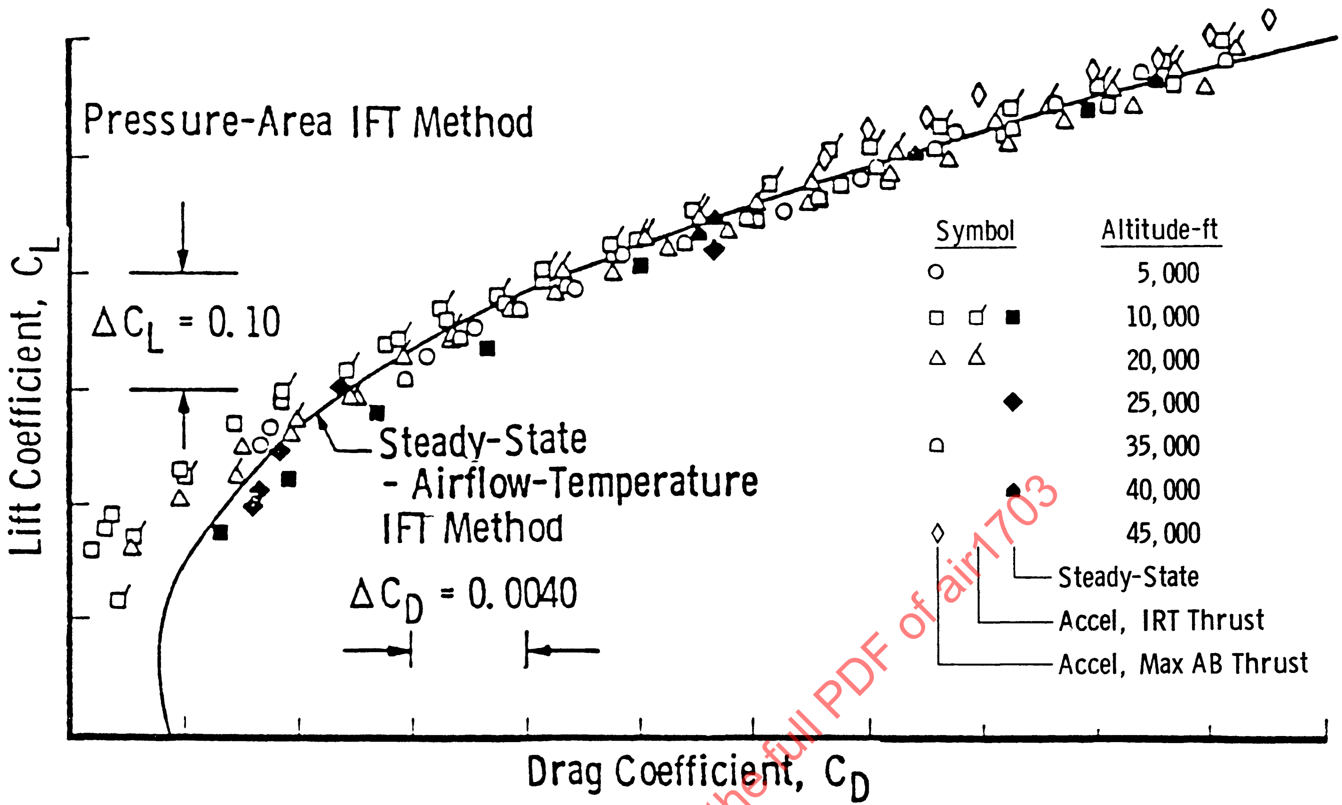


Figure 7.18 - Flight Test Drag Characteristics-Pressure-Area IFT Method

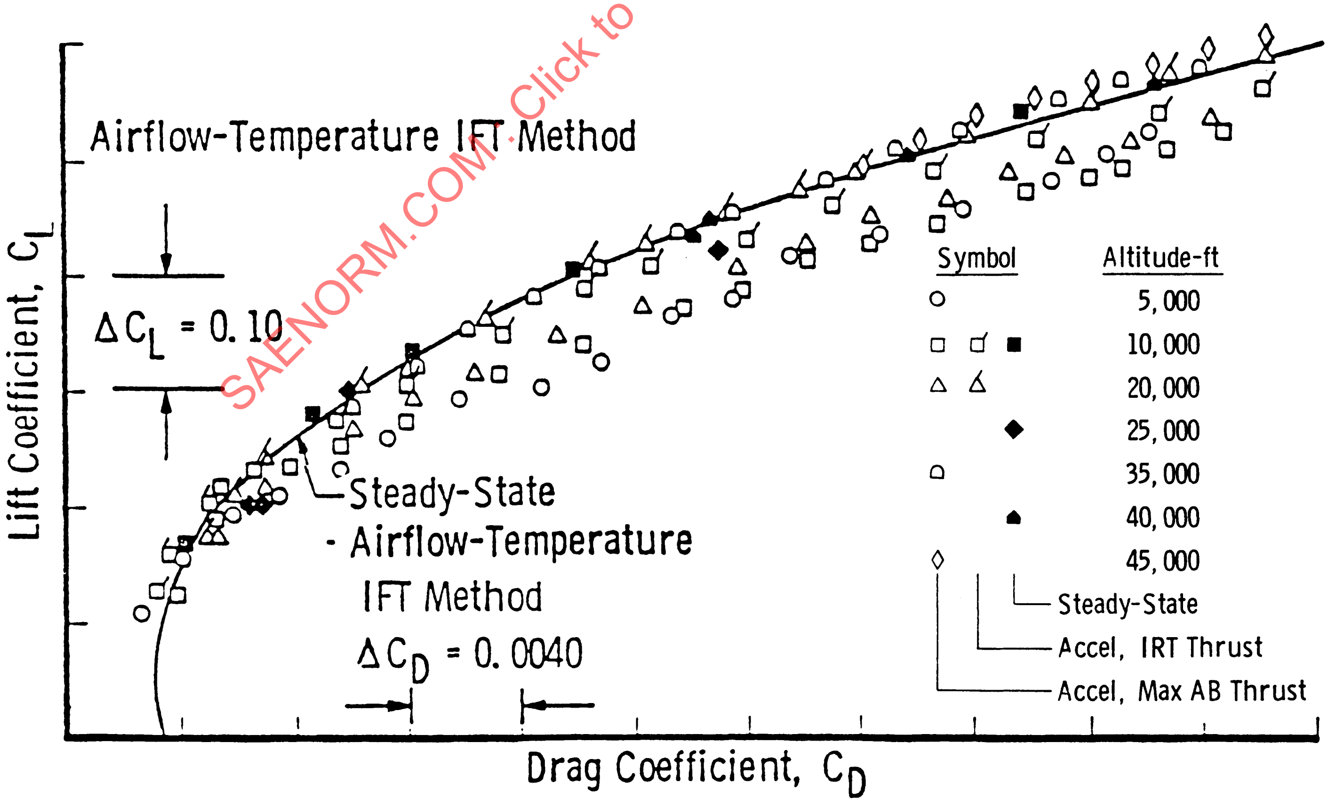


Figure 7.19 - Flight Test Drag Characteristics-Airflow-Temperature IFT Method

Conditions: Mach 0.9, 1 g, 30,000 ft.

Legend:

FS - Fuselage Station

CP - Pressure Coefficient

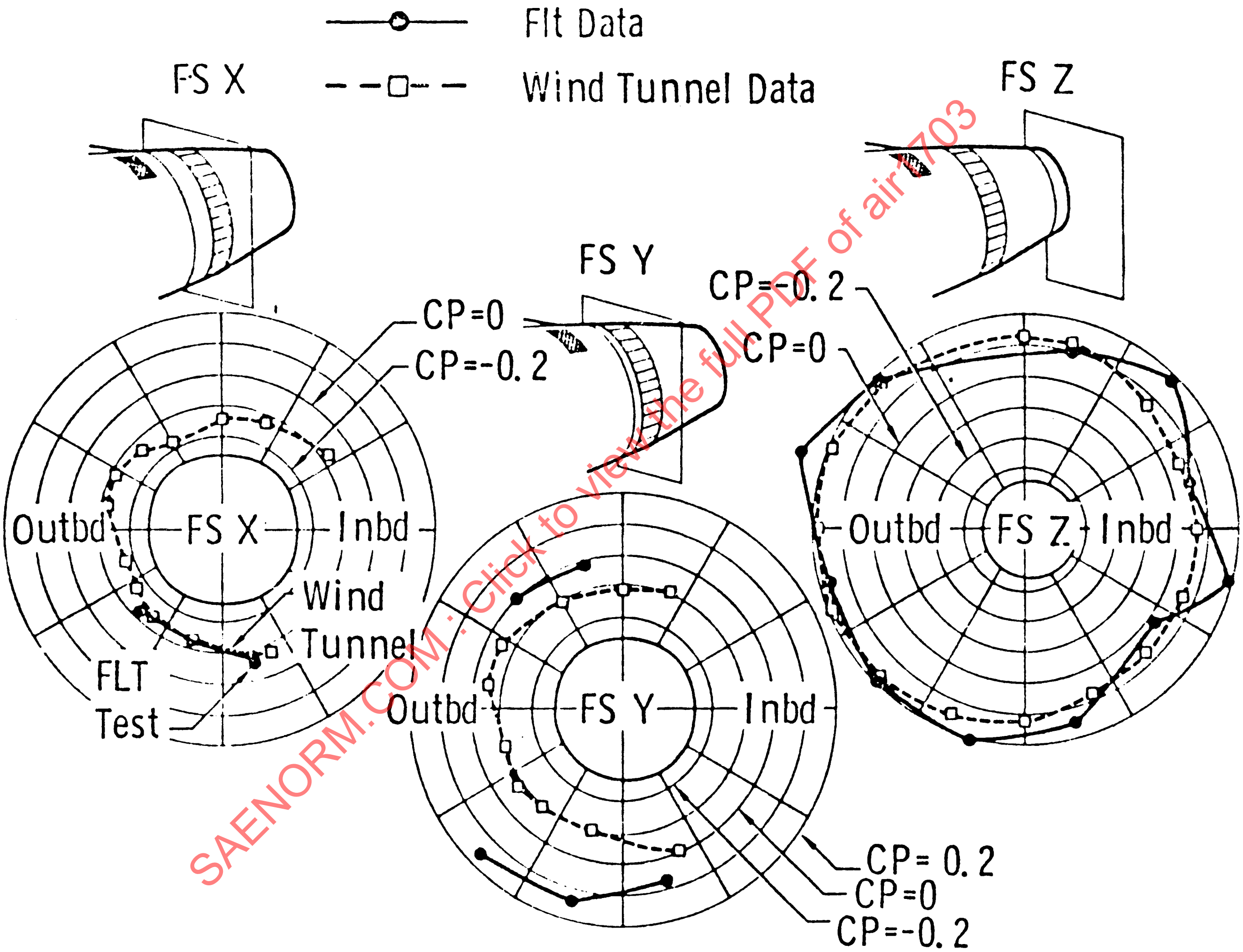


Figure 7.20 - Comparison of Afterbody External Static Pressures

8. TEST PLANNING GUIDELINES:

8.1 Approach to Planning: A systematic approach to advance planning is required for the effective and efficient conduct of a flight test program. Since the flight test program is an integral part of the overall development process, the data acquisition systems, data reduction procedures, and test plan matrices must be structured to meet the overall program objectives. The success achieved from the test program is directly related to the degree of attention given to details early in the planning phase of the overall development program.

A properly planned program will necessitate the integration of:

- Model, rig, and full-scale propulsion system test programs
- Airframe wind tunnel test programs
- Airframe and propulsion system test instrumentation
- In-flight thrust computational methods
- Data handling procedures used for both the propulsion system ground tests and airframe/propulsion system flight tests
- Data accuracy requirements and assessment capabilities.

Early involvement of both propulsion system ground test and flight test specialists will enhance the potential for successfully accomplishing the desired ground and flight test goals.

8.1.1 Task Force Concept: A successful approach for integrating the planning and prosecution of the overall development program, with its various test programs, has been the use of a task force concept involving all the various specialists of all organizations associated with the development, test, and evaluation of both the propulsion system and the airframe/propulsion system combination. An integrated planned approach should be formulated by the task force and responsibilities assigned for prosecution of the plan. Periodic meetings of the task force should be held to review progress and update the plan. The organization responsible for the combined airframe/propulsion system performance would normally be expected to:

- Formulate the task force
- Provide overall leadership in the formulation of an integrated plan which meets the program objectives with optimum utilization of available resources
- Ensure each specialist understands how his particular function fits into the overall plan

8.1.1 Task Force Concept (Cont'd.):

- o Ensure there are adequate interfaces between related functions
- o Assess the success/failure of each phase of the plan.

8.1.2 Integrated Program Elements: The following summarizes a systematic approach to test planning and program prosecution:

- o Establish overall program objectives
- o Assess available resources
- o Formulate task force
- o Set test program goals
- o Establish data accuracy goals
- o Devise an integrated, comprehensive program plan
- o Assign task responsibility
- o Conduct uncertainty analysis of candidate instrumentation/data acquisition systems and in-flight thrust computational methods
- o Assess prior success/failure of candidate computational methods with same/similar propulsion systems
- o Select primary and alternate computational methods
- o Select appropriate instrumentation and data acquisition systems
- o Devise integrated detail test plan matrices for the ground and flight test programs
- o Update uncertainty analysis as new information/data become available
- o Assess the success/failure of each phase of the program plan
- o Modify program plan and individual test plan matrices as required
- o Assess the validity of the data generated during the ground and flight test programs, and publish results.

Several critical aspects to be considered in the selection of in-flight thrust computational methods are discussed in Paragraph 8.2, and an outline for an integrated program plan is presented in Paragraph 8.3.

8.2 Selection of Method: The selection of a method (or methods) for computing in-flight thrust is predicated on the particular circumstances of the specific program. Factors which should be considered in this selection are:

- o The scope and nature of the flight test program, as dictated by the applicable flight test goals and program objectives
- o The economics of the particular situation, as dictated by available resources relative to the desired test goals and program objectives
- o The known state-of-the-art accuracies obtainable from each of the candidate methods which are appropriate to the type of propulsion system (turbojet, turbofan, nozzle type, etc.).

The goals of the flight test program are predominant factors which will influence the selection of a method for computing in-flight thrust. It is paramount that these goals be established prior to any attempt to select a computational method. General test program goals, applicable to both the ground and flight test programs are:

- o To substantiate that the design performance of the propulsion system and propulsion system/airframe combination has been met
- o To provide information for assessment of departures from design goals and identification of the causes of departures
- o To provide information to predict performance variations resulting from design changes which may be made during the operational life cycle of the airframe and propulsion system
- o To provide information to the designer to substantiate and enhance theoretical estimation techniques
- o To provide a data base for the improvement of future designs.

A performance prediction of the propulsion system/airframe design is based on airframe drag estimates and propulsion system thrust estimates. The overall performance of the aircraft can be substantiated by the flight test program without specific knowledge of propulsion system thrust and airframe drag. However, if the goal of the flight test program is to provide information for the explicit assessment of departures from the design goals, then an in-flight thrust computational method which will provide accurate in-flight thrust information is required. The cost involved in the development and utilization of one particular in-flight method compared to another must be weighed against the fidelity/uncertainty of the data obtainable from each of the methods and related to the accuracy requirement of the desired departure assessment. The necessity for providing highly accurate information for the enhancement of theoretical estimation techniques, and the size and quality of the data base to be generated for the assessment of design changes and improvement of future designs, must be considered.

8.2 Selection of Method (Cont'd.):

Selection of in-flight computational methods and the associated instrumentation/data-acquisition systems will be influenced by the desired accuracy and the resources available for achieving this accuracy. Uncertainty analysis of the proposed instrumentation and computational methods should be conducted early during program planning to establish the predominant parameters which influence accuracy and to highlight areas where the expenditure of resources would be most beneficial. The uncertainty data can be used to:

- Aid in the selection of a preferred in-flight thrust computational method
- Assess the necessity for alternate computational methods and aid in their selection
- Ensure that the model, rig and full-scale propulsion system ground test programs are adequate to achieve the desired flight test accuracies
- Enhance the confidence level associated with the validity of the data obtained during the flight test program.

Uncertainty analyses should be initiated early in the program planning process and updated as additional information becomes available.

Any complete in-flight thrust program should include backup method(s) for thrust determination. Although backup methods may appear to add program cost, having other method data can minimize retest and schedule slippages.

8.3 Example of an Integrated Program Plan: The following program plan outline is presented to illustrate the various functions that should be considered during the early planning phase of a development program. The plan is divided into sections related to the development of the propulsion system, the development of the air vehicle, and the conduct of the flight test program. Most of the functions in the first two sections can be conducted concurrently, but some of the results from the air vehicle tests are required to complete certain of the propulsion system functions.

8.3.1 Propulsion System Development:

- Conduct nozzle tests on scale models to determine momentum losses due to friction and non-axial flow at nozzle exit, static pressure distribution over the nozzle length, and thrust and flow coefficients. Adjust results to full scale.
- Conduct analyses to determine anticipated nozzle mass flow variations and/or leakage.
- Conduct rig tests to determine frictional (and blockage) and total pressure losses from the pressure measurement station to nozzle inlet. Conduct analyses to determine additional total pressure and momentum losses resulting from leakage and the addition of heat, if applicable.

8.3.1 Propulsion System Development (Cont'd):

- o Update engine thermodynamic-cycle computer model based on component rig, scale-model, and engine test results.
- o Review recent experience and state-of-the-art accuracies obtained with similar propulsion systems and candidate in-flight thrust computational methods.
- o Initiate uncertainty analyses of candidate in-flight thrust computational methods and instrumentation/data acquisition systems.
- o Select primary and alternate in-flight thrust computational methods, and establish computation procedures/computer routines to be used to analyze full-scale GLTB and ATF data.
- o Select and standardize instrumentation for GLTB, ATF and flight programs using information developed from method and uncertainty analyses. Establish instrumentation calibration accuracy requirements.
- o Establish flight test and GLTB/ATF test plan matrices.
- o Conduct engine ground calibrations, including the effects of inlet distortion, ram recovery, and shaftpower and bleed extractions.
- o Analyze GLTB and ATF data, updating in-flight thrust computational routines, and thermodynamic-cycle computer models, as appropriate. Update uncertainty analyses to highlight areas of concern.
- o Review early flight test results to ascertain the necessity for additional ground testing and/or revision to the in-flight thrust computational routine.

8.3.2 Air Vehicle System Development:

- o Conduct wind tunnel tests on aerodynamic scale models at selected reference conditions and adjust data to full scale.
- o Conduct wind tunnel tests to determine throttle-dependent force increments.
  - (a) Determine inlet spill drag characteristics from inlet drag model.
  - (b) Determine aft-end/nozzle drag characteristics from jet effects model.
  - (c) Assess other force increments, if appropriate.
- o Determine inlet-duct total pressure recovery.
  - (a) Determine inlet recovery from wind-tunnel test data using aerodynamic or inlet-drag model.

8.3.2 Air Vehicle System Development (Cont'd.):

(b) Establish full-scale flight test instrumentation requirements and flight-data handling/default substitution criteria, based on uncertainty analyses.

- Establish horsepower extraction relationships based on full-scale rig testing.

- Determine bleed airflow extraction requirements and measurement technique:

(a) Establish bleed duct instrumentation configuration

(b) Calibrate full-scale bleed ducts/instrumentation

(c) Incorporate this information into flight-vehicle uncertainty analyses.

- Conduct analyses to determine drag increments associated with engine-bay cooling airflow, inlet-ramp boundary-layer bleed flow, and other systems.

8.3.3 Flight Test Program:

- Conduct preflight performance model updates:

(a) Update installed-engine-performance model (thrust, fuel flow, and airflow as functions of altitude and Mach number) based on GLTB, ATF and inlet pressure recovery data

(b) Update airframe-system-drag model with available wind tunnel data.

- Establish instrumentation/data-acquisition system calibration procedures, tolerance limits, and special calibration jig requirements based on uncertainty analysis; and establish data handling procedures.

- Establish flight-test program goals and construct test plan matrices. Provisions should be made in the flight program to:

(a) Verify the wind tunnel inlet recovery data

(b) Verify the installed engine performance model

(c) Verify the airframe system drag and investigate the throttle-dependent drag increments (inlet spill and aft end/nozzle drag)

(d) Verify that the performance models accurately calculate aircraft performance throughout the flight envelope

8.3.3 Flight Test Program (Cont'd.):

(e) Provide flexibility and redundancy that will permit the investigation of unanticipated results and the isolation/identification of causes or error sources.

- Analyze flight test results for consistency and accuracy. Update analytic models as required.

- Publish flight handbook performance data.

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APPENDIX A

FUNDAMENTALS OF THRUST/DRAG ACCOUNTING

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SYMBOLS

Roman	Description	US Common Units	SI Units
A	Projected area normal to the freestream flow direction	ft^2	m^2
D	Drag	lb	N
$\vec{F}$	Absolute force vector	lb	N
F_{EX}	Excess thrust	lb	N
F_G	Gross thrust or gauge stream force	lb	N
F_G^*	Modified gross thrust	lb	N
F_{IPF}	Installed propulsive force	lb	N
F_N'	Net thrust	lb	N
F_N^*	Modified net thrust	lb	N
F_N	Overall net thrust	lb	N
F_{net}	Axial component of net (gauge) force	lb	N
ΔF	Incremental force bookkept as part of F_{IPF}	lb	N

SYMBOLS (CONTD.)

	Description	US Common Units	SI Units
$\vec{n}$	Unit vector normal to surface element dS directed outward		
NPF	Net propulsive force	lb	N
P_s	Static pressure	lb/ft^2	N/m^2
$\overline{P_s}$	Area weighted static pressure	lb/ft^2	N/m^2
P_t	Stagnation (total) pressure	lb/ft^2	N/m^2
$\dot{S}$	Surface area	ft^2	m^2
t	time	s	s
v	Volume	ft^3	m^3
$\vec{V}$	Velocity vector	ft/s	m/s
V_n	Outward component of velocity vector normal to surface area element, dS	ft/s	m/s
V_x	Axial component of velocity vector	ft/s	m/s
$\overline{V}$	Axial component of mass weighted velocity	ft/s	m/s
W	Mass flow rate	slug/s	kg/s
Greek			
θ	Local surface inclination relative to freestream flow direction	deg, rad	rad
ϕ	Net (gauge) axial force	lb	N
ρ	Density	$slug/ft^3$	kg/m
Σ	Summation		
$\vec{\tau}$	Local shearing stress vector	lb/ft^2	N/m^2, Pa

SYMBOLS (CONTD.)

	Description	US Common Units	SI Units
Subscripts			
0	Upstream infinity		
1	Inlet highlight		
2	Inlet discharge, engine front face		
7	Engine discharge exhaust nozzle inlet		
9	Engine exhaust nozzle discharge		
00	Downstream infinity		
AB	Afterbody		
AFS	Airframe system		
EXH	Exhaust		
FB	Forebody		
INL	Inlet		
inst	Installed		
int	Intrinsic		
isol	Isolated		
meas	Measured		
nac	Nacelle		
op	Operating		
post	Post-exit		
pot	Potential flow		
pre	Pre-entry		
quies	Quiescent, "wind-off"		
REF	At reference conditions		

APPENDIX A
FUNDAMENTALS OF THRUST/DRAG ACCOUNTING

A.1 INTRODUCTION: The material of this appendix proceeds from basic fundamentals to outline the principles of thrust, drag, and force accounting for an axially aligned propulsion system, as indicated schematically on Figure A.1. The relationships derived herein are consistent with those described in Paragraph 2.2 of the main report. This appendix is intended to illustrate only the basic principles involved.

The objective of the thrust/drag accounting method described in Paragraph 2.2 and developed herein is the production of a throttle-independent aircraft drag polar based on in-flight thrust determination. This thrust/drag accounting method is specifically applicable to conventional (CTOL) aircraft incorporating highly integrated (buried) engine installations in situations requiring the detailed bookkeeping of thrust and drag over a wide range of engine throttle settings at a given flight condition (e.g. flight at Mach 0.3 with the throttle set for Mach 3.0).

In discussing the fundamentals of thrust/drag accounting, it is convenient to begin by considering an isolated propulsion system, such as a nacelle removed from the flow field of the remainder of the aircraft. Isolation of the propulsion system (if in fact it is physically possible) will, in most cases, result in changes in the forces exerted on both aircraft and propulsion system surfaces. If the axial components of these changes do not cancel, the result is interference drag which may be either favorable or unfavorable. Lift is also likely to be affected, and therefore the interference drag at constant lift may include incremental changes in induced drag and other aerodynamic effects that are beyond the scope of this AIR.

A.2 Force Acting On A Surface: The force exerted on a surface by a flowing fluid is a vector quantity and is equal to the surface integral of the normal (pressure) and shearing (friction) stresses acting on the surface. Using vector notation, (See Figure A.2a)

$$\vec{F} = \int_{surface} (-P_s \vec{n} + \vec{\tau})\, dS \qquad (A.1)$$

Where, $\vec{F}$ = the total force acting on the surface
= the hydrostatic buoyancy plus the incremental force due to the relative motion of the fluid.

P_s = the static pressure acting on the surface element, dS,

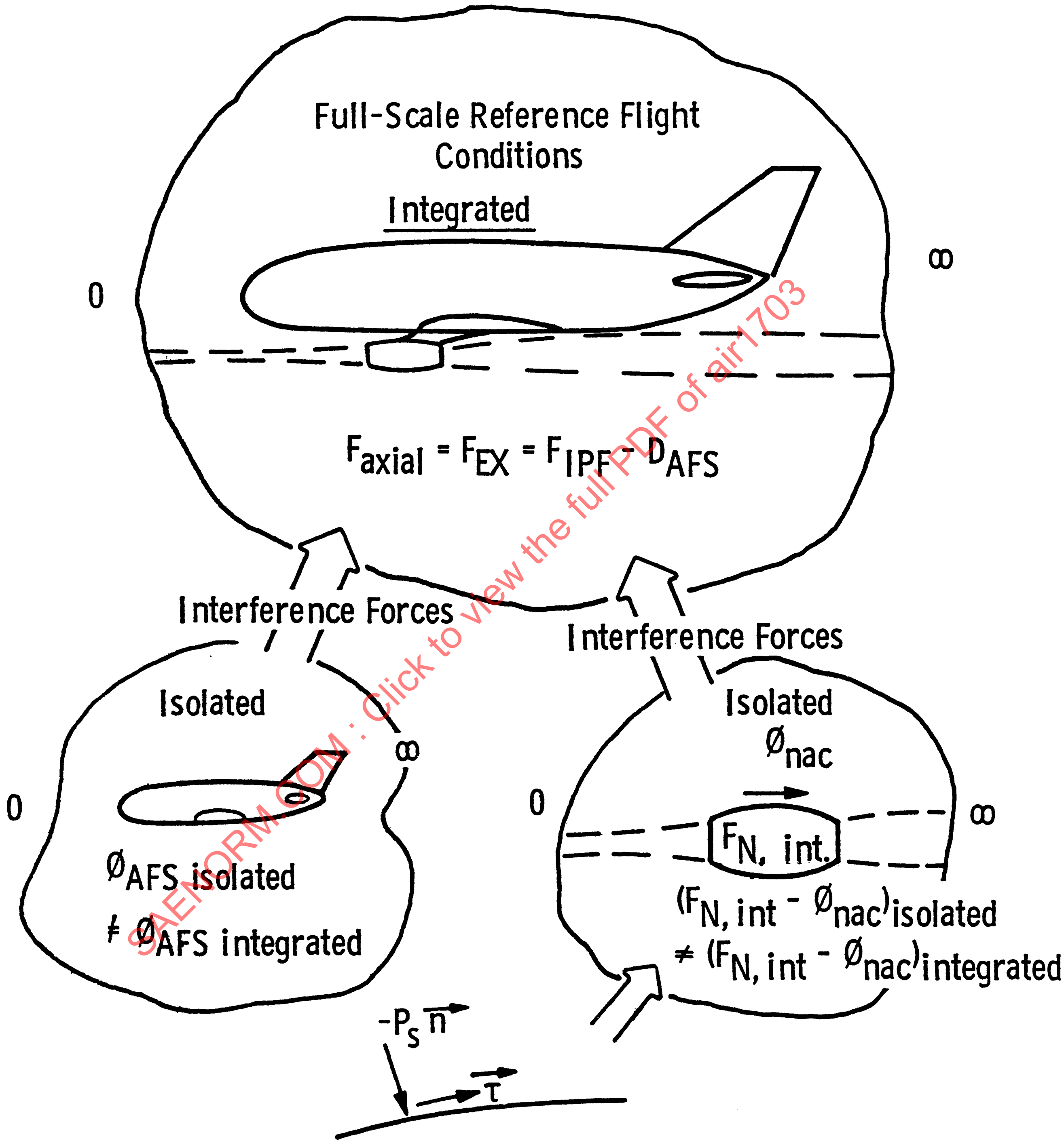


Figure A.1 - Definition and Bookkeeping of Propulsion Related Forces

A.2 Force Acting On A Surface (Cont'd.):

$\vec{\tau}$ = the shearing stress acting on dS, and

$\vec{n}$ = the unit outward normal vector (i.e. vector of magnitude 1, normal to dS and directed into the fluid).

In the case of an axisymmetric or two dimensional flow the axial (parallel to the freestream flow and directed downstream) component of this force is represented by the expression (See Figure A.2b).

$$F_{axial} = \int_{surface} (P_s + \tau \cot \theta)\, dA \qquad (A.2)$$

where, $dA = \sin\theta\, dS$ = the area of the surface element dS projected on a plane normal to the free stream flow direction, and

θ = the acute angle between the freestream flow direction and a plane tangent to the surface element dS ($\sin\theta$ always positive).

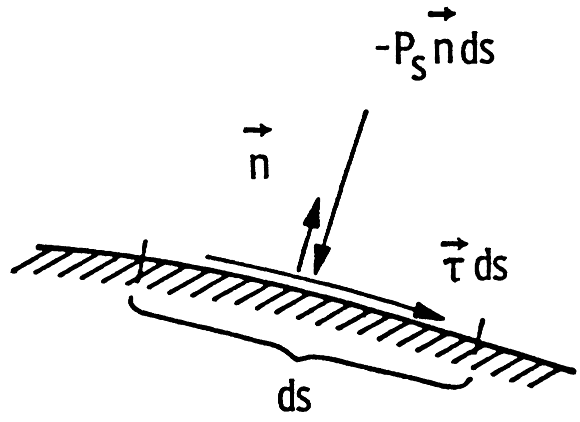
The sign convention adopted is such that the axial component of the force acting on a surface element is considered positive when directed downstream. Thus, the static pressure, P_s, is given a positive sign for forward facing surface elements and a negative sign for aft facing elements. If the axial component of the local velocity is directed upstream, the shearing stress is given a negative sign.

Finally, if hydrostatic buoyancy forces are neglected, the axial component of the net (gauge) force (i.e. the force due to the relative motion of the fluid) is given by

$$\phi = \int_{surface} \left[(P_s - P_{s0}) + \tau \cot \theta\right] dA \qquad (A.3)$$

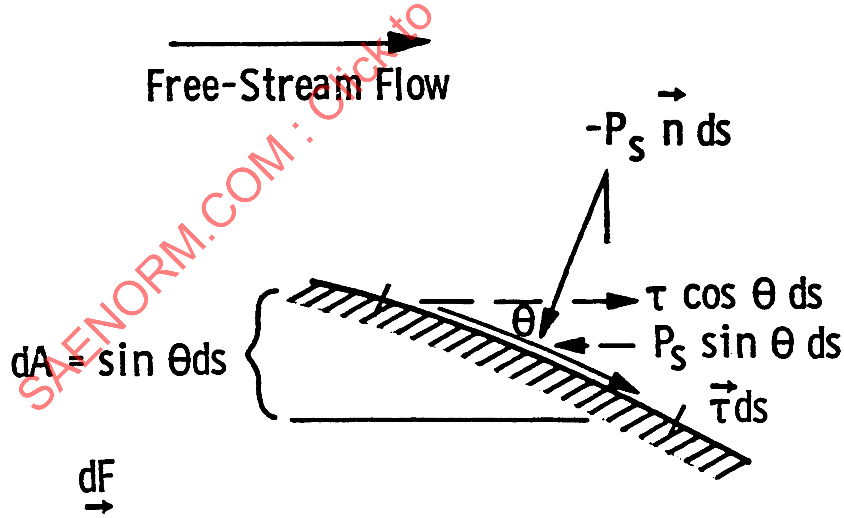
A.3 Momentum Equation: The momentum theorem provides a means of relating the forces exerted on a fluid within a control volume to the rate of change of linear momentum within the volume and the net outflow of linear momentum. For a fixed reference system (for all practical purposes one moving at constant velocity relative to the earth's surface) and using vector notation, the relationship is:

$$\sum \vec{F} = \frac{d}{dt} \int_{\substack{control\\volume}} \rho \vec{V}\, dv + \oint_{\substack{control\\surface}} \rho \vec{V} (\vec{V} \bullet \vec{n})\, dS \qquad (A.4)$$



$$dF = (-P_S \vec{n} + \vec{\tau})ds$$

a. General case



$$dF_{axial} = (-P_S \sin \theta + \tau \cos \theta)\, ds$$

b. Two-dimensional or axisymmetric flow
(Lateral component of shearing stress vector = 0.)

Figure A.2 - Force Definitions

A.3 Momentum Equation (Cont'd.):

Where, $\sum\vec{F}$ is the vector sum of all forces acting on the fluid within the control volume and is conventionally considered positive when acting in the downstream direction. The forces comprising $\sum\vec{F}$ consist of:

Body forces such as gravitational, magnetic, and electrodynamic forces which are often negligible or non-existent, and

Surface forces resulting from normal (pressure) and shearing stresses.

$\frac{d}{dt}\int_{\text{control volume}} \rho\vec{V}\,dv$ represents the time rate of change of fluid momentum within the control volume and is zero for a steady flow. ρ and $\vec{V}$ are the local density and velocity within an element of control volume dv.

$\oint_{\text{control surface}} \rho\vec{V}(\vec{V}\bullet\vec{n})\,dS$ represents the net outflow of linear momentum through the closed control surface. In this case $\vec{n}$ is the unit normal directed outward from the control surface.

For a two dimensional or axisymmetric, steady flow with body (gravitational) and hydrostatic forces neglected, the axial component of the net force acting on the flow is given by:

$$\sum F_{net} = \oint_{\substack{\text{control}\\ \text{surface}}} \rho V_x V_n \, dS \tag{A.5}$$

where, V_x is the axial component of velocity and V_n is the outward component of velocity normal to dS.

$\sum F_{net}$ is conventionally considered positive when acting on the flow in the downstream direction; its reaction will therefore be positive when directed upstream. In applying the momentum theorem, it should be noted that surface forces evaluated according to the sign convention of Section A.2 will be positive in the downstream direction, and their contribution to $\sum F_{net}$ will be $-\sum\phi$.

A.4 Definition of Drag: The term drag is frequently used to represent forces or terms in the momentum equation which, in a strict sense, are not drag. The terms ram drag (momentum), and additive (or pre-entry) drag (force) are examples of this practice. A similar comment can be made with regard to thrust.

A.4 Definition of Drag (Cont'd.):

For the purpose of this appendix, drag is defined as follows:

> The drag of a surface is the difference between the axial component of the net force, ϕ, acting on the surface in a real flow and the axial component of the net force, ϕ_{pot}, which would act on the surface in a potential flow. The latter force, ϕ_{pot}, is called potential flow buoyancy.

The drag, as defined above, of a closed surface may be equated to an axial momentum defect at downstream infinity. This is consistent with drag being a consequence of energy dissipation occurring in the external flow. Given that the force acting on a surface is known, the drag of the surface can be known only if either its potential flow buoyancy or the momentum defect attributable to the surface is known.

A.5 Application of The Momentum Theorem:

a) The force exerted on a closed, isolated, non-lifting, body immersed in a flowing fluid:

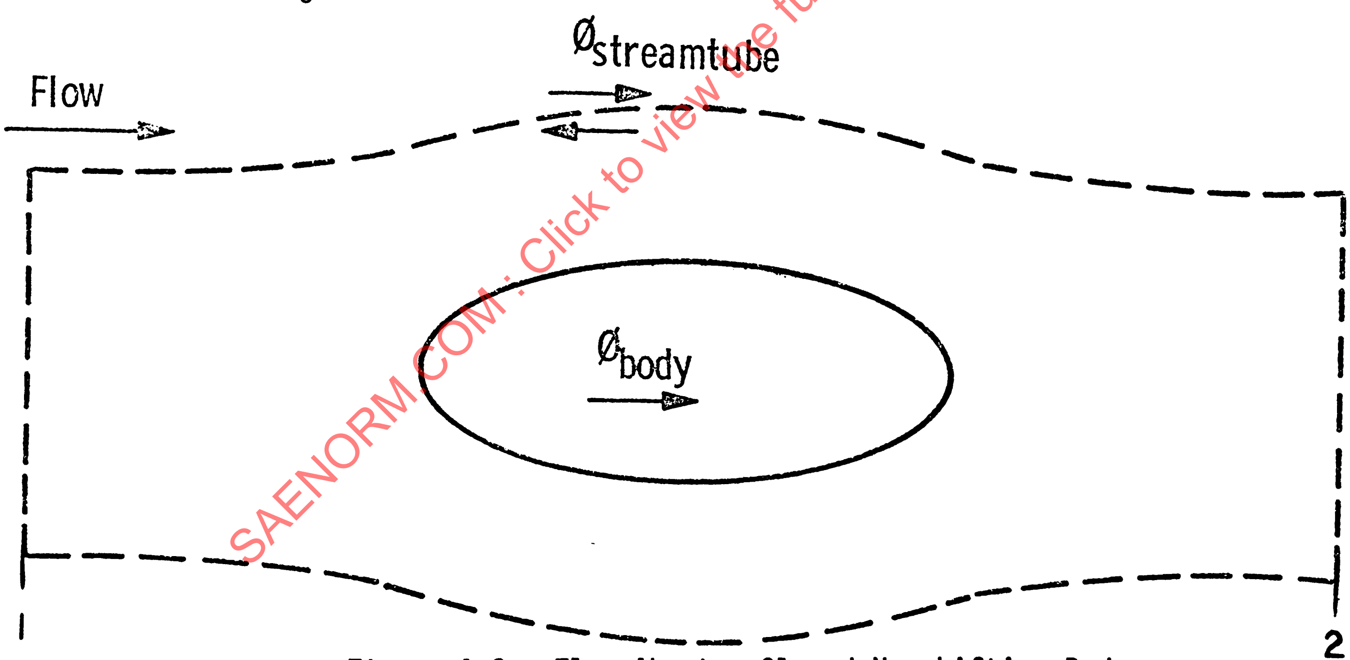


Figure A.3 - Flow About a Closed Non-Lifting Body

The lateral portion of the control surface between stations 1 and 2 (Figure A.3) consists of a streamtube which contains the entire wake of the body. The net force exerted by the flow on the surface of the body is:

$$\phi_{Body} = \oint_{\substack{\text{body}\\ \text{surface}}} \left[(P_s - P_{s0}) + \tau \cot \theta \right] dA \qquad \text{(A.6)}$$

A.5 Application of The Momentum Theorem (Cont'd):

The net force exerted by the streamtube surface on the flow within the control volume is:

$$\phi_{Streamtube} = \int_{\substack{\text{streamtube}\\ \text{surface}}} (P_s - P_{s0})\,dA \qquad (A.7)$$

Letting

$$\overline{V} = \frac{\int V_x \, dW}{W} \quad \text{at stations 1 and 2} \qquad (A.8)$$

and

$$\overline{P}_s = \frac{\int P_s \, dA}{A} \quad \text{at stations 1 and 2,} \qquad (A.9)$$

the momentum equation is

$$(\overline{P}_{s1} - P_{s0})A_1 + \phi_{Streamtube} - \phi_{Body} - (\overline{P}_{s2} - P_{s0})A_2 = W(\overline{V}_2 - \overline{V}_1) \qquad (A.10)$$

or

$$\phi_{Streamtube} - \phi_{Body} = F_{G2} - F_{G1} \qquad (A.11)$$

where, $F_G = W\overline{V} + A(\overline{P}_s - P_{s0})$ is called gauge stream force (see reference A.1). (A.12)

If the distance from the body to the streamtube is sufficiently great that $P_s = P_{s0}$ at the streamtube surface and if stations 1 and 2 are located sufficiently far upstream and downstream that $\overline{P}_{s1} = \overline{P}_{s2} = P_{s0}$, then:

$$F_{G1} = W\overline{V}_1 = WV_0 \qquad (A.13)$$

and

$$F_{G2} = W\overline{V}_2 = W_{wake}\,\overline{V}_{wake} + W_{\substack{\text{outside}\\ \text{of wake}}}\,V_0 \qquad (A.14)$$

Therefore

$$-\phi_{Body} = F_{G2} - F_{G1} = W_{wake}\,(\overline{V}_{wake} - V_0) \qquad (A.15)$$

Furthermore, in a potential (isentropic) flow there is no wake (or alternatively the velocity at station 2 is everywhere equal to V_0), and the force, $\phi_{Body,\ pot}$, is the potential flow buoyancy and is equal to zero.