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APPROACHES TO DETERMINE AND EVALUATE THE STABILITY OF PROPULSION SYSTEMS

R. C. TEAR, SQUADRON LEADER, RAF

TECHNICAL REPORT AFAPL-TR-67-75

FEBRUARY 1968

AIR FORCE AERO PROPULSION LABORATORY
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

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AFAPL-TR-67-75

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R. C. TEAR, SQUADRON LEADER, RAF

FOREWORD

This report was prepared by the Turbine Engine Division of the Air Force Aero Propulsion Laboratory, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio, with Charles E. Bentz as project engineer. The work reported herein was accomplished in-house under Project 3066, Task 306603, "Advanced Engine Studies." Analyses made during the period from September 1965 to September 1966 are reported. This report was submitted by the author October 1966.

Appreciation is expressed to North American Aviation, Inc., Los Angeles, California, for their permission to include Figures 8 through 15 in this report.

Publication of this report does not constitute Air Force approval of the report's findings or conclusions. It is published only for the exchange and stimulation of ideas.

Ernest C. Simpson
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Chief, Tubine Engine Division

ABSTRACT

A major consideration in the current design and operation of air vehicles is the stability of the total propulsion system. The derivation of stability criteria involves many evaluations and determinations at the subcomponent level to ascertain stable operating regions and the performance coefficients that are associated with the elemental margins. Painstaking compilation of the performance and stability margin for the key flight points of operation can thus be derived and a numerical value assessed with respect to stability margin. These data, for the most part, are dependent upon empirical coefficients drawn from previous weapon system developments. As such, a major effort is warranted to define alternative approaches to the problems of stability margin assessment that will provide a likely improvement in performance for the system. Development of the capability for improved performance and defined stability margin is directly dependent on an accurately determined interface between an engine and the associated airframe. Such a determination can be achieved only if there is expert liaison between and complete cooperation by the engine and airframe development companies. The interface between such companies must be strongly developed and is extremely critical in time-phased relationships between the engine and airframe development companies.

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TABLE OF CONTENTS

SECTION		PAGE
I	GENERAL INTRODUCTION AND DEFINITION OF PROBLEM	1
	1. Introduction	1
	2. General Stability	1
	3. Approaches to Stability Resolution	1
	4. Limit Concepts to Resolution	2
	5. Allocation of Destabilizing Influences	3
II	NEED FOR STABILITY DEFINITION	6
	1. Introduction	6
	2. Evaluation of Current Distortion Indices	6
	a. Introduction	6
	b. Total Pressure Distortion	6
	c. Distortion Indices Developed by Contractors	7
	3. Nonsteady Effects	16
	4. Summary and Conclusions	17
III	CAUSES AND SOURCES OF INLET DISTORTION	18
	1. Introduction	18
	2. Inlet Variable Geometry	20
	3. Buzz — Theories on Origin and Mechanism	20
	4. Inlet Bleed	20
	5. Some Contributory Secondary Sources of Inlet Distortion	21
	a. Radius of Curvature	21
	b. Ancillary Equipment	21
	c. External Boundary Layer Ingestion	22
	d. Proximity to Disturbing Flow Fields	22
	6. Inlet Control	22

TABLE OF CONTENTS (CONTD)

SECTION		PAGE
IV	FACTORS THAT INFLUENCE SURGE MARGIN ALLOCATION	32
	1. Introduction	32
	2. Acceleration and Deceleration	33
	3. Control Mode	34
	4. Fuel Control Tolerances	35
	5. Afterburner Light	35
	6. Component Variations and Manufacturing Tolerances	36
	7. Variable Stator and Inlet Guide Vane (IGV) Rigging	36
	8. Interstage and/or Interspool Bleed Effects	37
	9. Compressor Degradations Caused by Age	38
	10. Reynolds Number Effect	39
	11. Inlet Flow Maldistribution	39
	a. Velocity Distortion	39
	b. Radial Distortion Attenuation	39
	c. Circumferential Distortion Attenuation	40
	d. Effect of Distortion on Compressor Stall	40
	e. Compressor Tolerance to Distortion	41
	f. Transient Distortions	42
	g. Inlet Temperature Distortion	42
	12. Defined Surge Margin	42
	13. Engine Performance as Influenced by Compressor Stability Requirements	43
	14. Decisions Leading to Performance Compromise	53
	15. Procedures to Preclude De ·ating	55

TABLE OF CONTENTS (CONTD)

SECTION		PAGE
V	INLET SYSTEM TEST	59
	1. Introduction	59
	2. Current Limitations	59
	3. Scale Effects	60
	4. Timing	62
	5. Summary	63
VI	INSTRUMENTATION	64
	1. Introduction	64
	2. Exploratory and Advanced Development Phase	64
	3. Engineering Development	65
	4. Summary	65
VII	CONTROL SYSTEM DETERMINATION	66
	1. Introduction	66
	2. Current Controlling Procedure	66
	3. Future Control Considerations	67
	4. Description of an Advanced Controlling Parameter	68
VIII	DYNAMIC MODELING	70
IX	DETERMINATION OF STABILITY MARGIN	74
	1. Evaluation of Stability Margin	74
	2. Summary	82
REFERENCES		83

ILLUSTRATIONS

FIGURE		PAGE
1.	Typical Inlet/Engine Stability Range	2
2.	Typical Compressor Stage Characteristic	4
3.	Radial Location Factor	10
4.	Circumferential Pressure Variation	12
5.	Circumferential and Radial Profiles	13
6.	Radial Weighting Factor	14
7.	Radial Location Factors	16
8.	Threat Control Function	23
9.	Effect of Throat Positioning on Pressure Recovery	25
10.	Effect of Aircraft Pitch Attitude on Unstart Throat Margins at Mach 3	26
11.	Effect of Aircraft Yaw Attitude on Unstart Throat Margins at Mach 3	27
12.	Bypass Control Functions	28
13.	Normal Shock Parameter at Maximum Flight Speed	29
14.	Multiperformance Mode Function	30
15.	Normal Shock Parameter, Effect of Angle of Attack	31
16.	Typical Compressor Characteristic Map	32
17.	Effect of Acceleration	34
18.	Effect of A/B Light on Fan Pressure Ratio	36
19.	Effect of Manufacturing Tolerances	37
20.	Stator Angle Shift for Surge Control (Rapid Inlet Temperature Rise)	38
21.	Distortion Effect on Compressors	41
22.	Engine Stability Margin	44
23.	Cumulative Representation of Degrading Factors on a Compressor Performance Map	45
24.	12 Pressure Ratio Turbojet Combustor Characteristics	49
25.	12 Pressure Ratio Turbojet Combustor Characteristics	50

ILLUSTRATIONS (CONTD)

FIGURE		PAGE
26.	12 Pressure Ratio Turbojet Turbine Characteristics	51
27.	12 Pressure Ratio Turbojet Performance Changes	54
28.	Advanced Compressor Development General Performance History	56
29.	Compressor Testing	57
30.	Inlet Stability Testing Model and Full Scale Correlation	61
31.	Analog Versus Digital Simulation	72
32.	Inlet Map at Mach 1.9, 40,000 Ft ~ Maximum A/B	75
33.	General Compressor Surge Line Shift	76
34.	Compressor Surge Line Shift With Varying Levels of Circumferential Distortion	77
35.	Typical Circumferential Distortion Effects on Axial Compressor	78
36.	Example of Compressor Surge Line Shift With Distortion	79
37.	Compressor Derating in Reaction to Distortion	81

SECTION I

GENERAL INTRODUCTION AND DEFINITION OF PROBLEM

1. INTRODUCTION

The achievement of stability of the total propulsion system is one of the primary goals in the design of current and future air vehicles. Over the years, various aspects of this problem have been investigated, and the results of the investigations, along with test and operational data, have been documented. Generally, however, only one aspect of the many conditions — theoretical, technical, development phasing, or development procedures — which separately or together can create propulsion system instability, has received the concentrated attention of an investigator. As a result, although there is much applicable information to use as reference material in future efforts, the information is available in so many individual books, papers, and reports, that it is extremely difficult to see the problem and/or its solution as a whole.

Basically, this report is a compilation of the essence of the applicable information that is already documented. Some of the facts and related discussions presented herein, along with some of the proposed approaches that seem logical and pertinent to the effort to solve the problems of propulsion system stability, were derived from the compilation of already-available information (see Bibliography). Other facts, discussions, and suggested remedies were based on the accumulated experiences of the Air Force Aero Propulsion Laboratory and various contractors.

2. GENERAL STABILITY

The effectiveness of turbo machinery, as applied to the propulsion of an aircraft, is determined by many interrelated performance criteria. Engineering parameters such as thrust, specific fuel consumption, weight, and external dimensions have long been major influencing factors in determining the suitability of engines for given missions. These functions have not uniquely determined the operational effectiveness of a particular turbine engine design. A fifth indicator of the value of a specific design is the measure of tolerance and sensitivity of the machine to an unstabilizing environment. It can be referred to as the stability margin of the design.

The stability of any operational device is its ability to deliver a uniform output over a range of environmental conditions. In aircraft turbo machines, the most directly useful output is thrust. Since this must be regulated for effective use, the definition of turbine engine stability reduces to the fact that the engine will operate with a stable or uniform thrust output which is proportional to the throttle angle position and will transition between power level changes without significant discontinuities in flow and/or thrust output.

3. APPROACHES TO STABILITY RESOLUTION

A review of the history of turbine engine operation, as installed in currently operational aircraft systems, indicates a large variety of unstable modes of operation. These modes have different levels of impact on the operation of the system, and the priority of resolution of the modes of instability is dependent on an evaluation of the frequency and significance of a given occurrence of instability. Such a determination for the establishment of the preferred priority listing is completely dependent on the individual air vehicle system and has no validity in a general determination of approaches to the definition of overall system stability. For the purposes discussed in this evaluation, reference can be made only to determine value by quasi-averaging of the severity and frequency of the instabilities of several propulsion systems. As a determinant of methods, the proposed procedures can serve as a guide to other and more specific cases.

4. LIMIT CONCEPTS TO RESOLUTION

The approaches to evaluating stability are varied, as evidenced by the responses of the many systems currently in operational use. The ideal case can be made for evaluation of stream tube flow from a plane forward of an aircraft in flight to a plane aft of the exit plane of the nozzle and, in certain instances, the aft fuselage station of the aircraft. Such an approach must consider the value of forward fuselage boundary layer, shock formation and interaction with the boundary layer, supersonic and subsonic diffuser modulations of the flow, compression through the compressor, the heat addition process, expansion through the turbine, and nozzle flow characteristics. When these considerations are amplified by the requirements of secondary nozzle flow, fuel addition transients, and combustion kinetic stability, the problem of definition becomes insurmountable when viewed as a continuous flow process. The problem of imposing valid limits to such an equation, or series of equations, is itself an unending search for valid data. Obviously, in resolving stability, the first task must be to define such resolution according to primary cause and effect relationships involving unitized component iterations to define the performance of these components in terms of X versus Y dimensioning, up to limits which are uniquely defined for flow and/or pressure continuity within themselves, or they must be defined as modified by upstream or downstream component limitations which have an influence on the level of the continuous output used in the stability definition.

An illustration of system effects on the acceptable limits of individual component stability limits can be found in the subsonic diffuser section of an inlet. With a given divergence angle, performance of the exit of the diffuser, for stability reasons, can be determined by the radius or area which is flowing "core" or full total pressure fluid. If this parameter is plotted versus Mach number, the data shows wide stability limits as can be seen on Figure 1. When a compressor, which is sensitive to distortion, is added behind such a diffuser, the wall separation occurring in the diffuser at Mach 0.5 to 0.6 can represent a limit at which the compressor becomes unstable, and further operation at that airflow ceases. This point then becomes a limit beyond which stability definition of the diffuser is valueless.

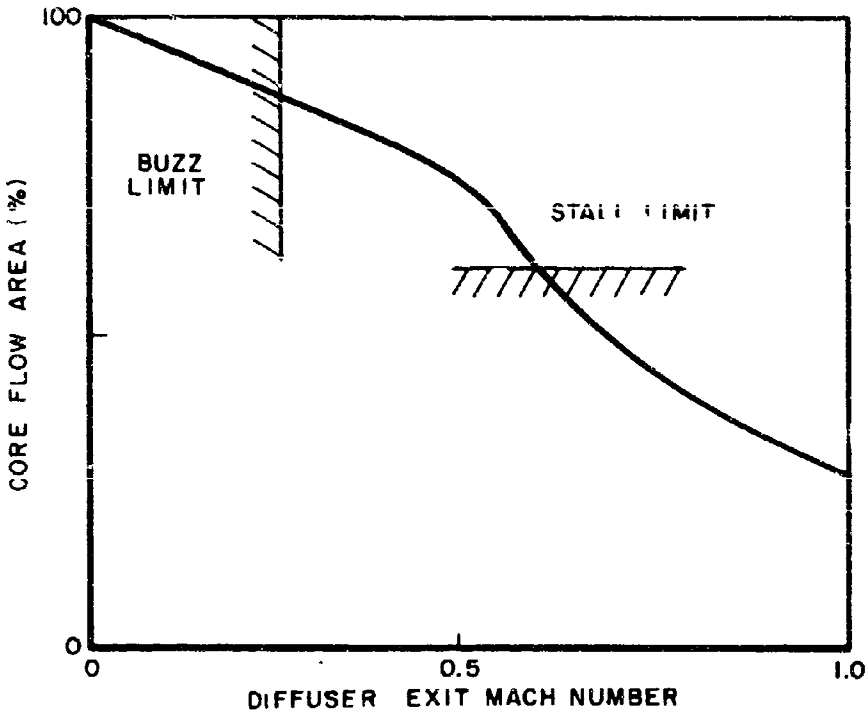


Figure 1. Typical Inlet/Engine Stability Range

With the addition of a supersonic shock system in front of the subsonic diffuser, the system becomes unstable at flow reductions toward the zero Mach number. This lower limit of acceptable flow can be determined by the buzz limit of the inlet system, for example. Thus, the simple example of the subsonic diffuser can be used to illustrate the need for three independent programs which can be used to establish the stability limits of a component. It should be further stated that dynamic interactions of either of the preceding limits must also be accounted for if a final determination of the stability limits is to represent the usage of the end product in a full aircraft propulsion system.

If any effective values are to be derived, it is important that the stability product be broken into its contributing elements. This will require categorizing the types of destabilizing factors according to basic causes and correlating the propulsion system operation with these causes. The three most significant instabilities in propulsion systems are the following:

- Aerodynamic variations in flow rate and/or pressure fluctuations due to flow separation in the compression process of the engine cycle.
- Flame or combustion discontinuity and, in particular, the combustion processes associated with low-pressure combustion, i.e., afterburners and duct burners.
- Control unresponsiveness to fluid flow phenomena. This listing also describes the impact and frequency pattern of the failure to perform in stable modes for turbo-propulsion systems.

The aerodynamic variations in the compression process are intensified by the positive pressure gradient with the result that the instabilities encountered in this process are the most severe and of the greatest consequence. As a first approach to the definition of propulsion system stability criteria, concentration on the effects, causes, and results of this system is warranted.

5. ALLOCATION OF DESTABILIZING INFLUENCES

The allocation of the contributions to the causes of instability in the pressure rise section of the propulsion system is complex and interrelated. The major sections or components of the system which determine the operational acceptability are the aircraft forebody configuration, the inlet and shock system of the aircraft, the engine, and, since it is currently common practice to connect the inlet to the exhaust nozzle system aerodynamically, the nozzle system's operation.

Consideration of the forebody influence reveals this contribution to be largely a generator of boundary layer which, when interacting with the shock system of the inlet, causes a variety of destabilizing influences in the propulsion system. In this regard, current common practice is the relatively complete diversion of the forebody boundary layer accumulation for both steady-state and maneuver conditions, and under such circumstances the major problem of definition is the boundary layer growth attendant with operation of the aircraft over its flight and maneuver envelope. In rare instances, performance trades have been made when an attempt was made to utilize the advantages of minimal boundary layer removal. Extreme care is required in the design of such inlets to insure the control and flow of this boundary layer through and into the pressure recovery component to assure any degree of stable operation. It is sufficient to observe that, when an inlet system has not been clearly defined and all or most of the external boundary layer must have been removed, then historically, stability problems have occurred.

In propulsion systems where the nozzle and inlet aerodynamics have been linked by secondary bypass flow systems, careful attention is required to determine that the transient performance of the nozzle is not unduly influencing the performance of the section which is being bled. One particular condition which has caused major problems is the low speed and takeoff performance regimes when reverse flow of heated air is exhausted into a compressor front face which is already being subjected to pressure distortion and thereby causing compressor stalls. Blocker doors and similar devices have been used to prevent this occurence. Again, however, these problems are secondary in the normal flight regime to the problems of inlet and engine compatibility.

The derivation of a stability margin for the most significant events which occur in the flight envelope of an air vehicle is dependent on the definition of three basic operational parameters. These are the time-dependent distortion of the inlet airflow as discharged by the inlet, the effects of this distortion on the stage characteristics of the compressor, and the compressor loading with respect to the stable pumping pressure ratio of the compressor. The derivation of the inlet data and the proper development and accounting for the compressor loading of a particular compressor are covered in greater detail in later sections. of this report. A general description of the overall loading characteristics is, however, appropriate at this point in the report.

The stall characteristic of the compressor is determined by the stage performance capabilities. A typical performance map of a stage is the plot of flow coefficient versus the pressure coefficient. In the region marked A on Figure 2, the flow-pressure relationship is positive in terms of pumping capability, and the stage is stable. As the back pressure on the stage is increased to the maximum pressure coefficient, instabilities of flow develop, impeding and degenerating the uniformity of the flow into the next stage. Generally, the flow separations are mild and do not block the flow in succeeding stages. As the stage is constrained by back pressure to operate in the C zone (Figure 2), marked fluctuation of flow is apparent. The flow-pressure ratio discontinuity shown is characteristic of several stage designs currently operational. Other designs do not have flow discontinuities in this zone of

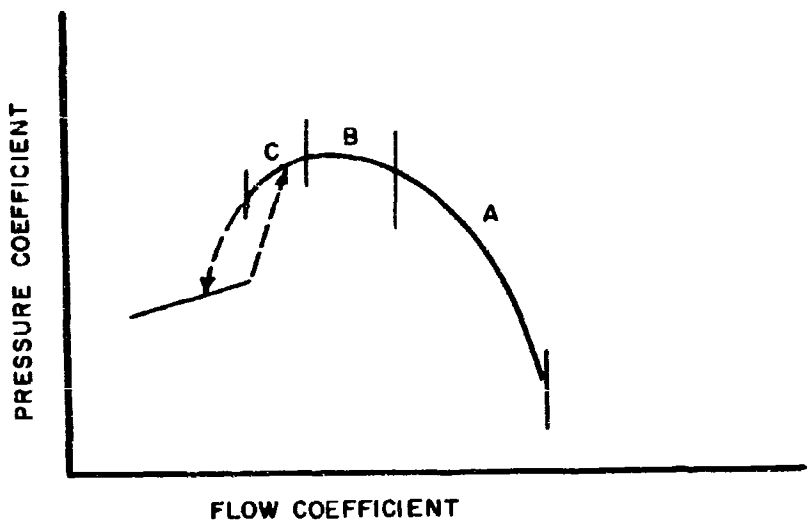


Figure 2. Typical Compressor Stage Characteristic

operation. However, all stage designs do exhibit the nonpositive flow-pressure coefficient characteristic with respect to pumping capability, and, as a result, all show total flow disorientation when constrained to operate at this low flow condition. These degrees of stage stability are referred to in multistage compressors as rotating or partial stall and complete or abrupt stall.

The unstable operation of the compressor occurs whether the pressure rise-flow relationship is caused by high back pressure or by a reduced inlet pressure resulting from inlet distortion. In either case, the result is similar in the multistage compressor. Therefore, the compressor must be loaded or stage-stacked with appropriate areas in the downstream stages to account for the expected distortion level. As is readily apparent, stage matching with large distortion tolerance acceptance can only be achieved with major sacrifices in stage pressure rise. This approach can be successful only by the addition of weight as more stages are added to the compressor, or the required overall pressure ratio is reduced markedly.

SECTION II

NEED FOR STABILITY DEFINITION

1. INTRODUCTION

The "distortion problem" has received a considerable emphasis in the recent past, resulting in a certain amount of confusion as to what the term distortion means and as to the current capability to handle distortion. The purpose of this section is to evaluate the current capability to define and predict the effects of distortion. Much of the information presented is historical in nature and taken directly from the published literature contained in the Bibliograpy at the end of the report.

The distortion problem, as it is understood at this time, can be divided into three parts: first, to establish an acceptable definition of the term distortion; second, to establish the effects of distortion on engine performance; and third, the essence of the problem, to establish the parameters or indices required to correlate the conditions in the face of engine-to-engine performance differences. The total distortion problem also includes sources of distortion and the factors that influence distortion, but these can be treated separately until more is known about the entire problem.

A general definition of distortion is, "Any deviation from a steady uniform distribution of the flow properties at the face of the engine." If this definition is used, one should understand that the undistorted condition, steady uniform flow, does not necessarily represent the ideal flow distribution condition for an engine and that some amount of distortion will represent the true condition with respect to engine performance. The definition, by implication, includes the occasionally popular distortion terms such as swirl, vorticity, and turbulence, as well as the more generally accepted time varying and steady nonuniform pressure or velocity terms. The effects of distortion are covered in other sections of this report, leaving the third area, the major problem, the correlation between flow conditions at the engine face and performance. This area has received much attention during the last dozen years and will undoubtedly receive greater attention in the future.

2. EVALUATION OF CURRENT DISTORTION INDICES

a. Introduction

In order to arrive at an understanding of the current capability of the turbine engine industry to predict performance with distorted input flow, an examination will be made of the distortion indices that have been developed and the basic flow models that have been used to arrive at these indices. The first examination will be on the use of total pressure distortion since it has served as the basis for all of the distortion indices used today.

b. Total Pressure Distortion

The total pressure of the flow entering the engine is determined in order to evaluate the performance of the engine by taking an average of several total-pressure measurements at the face of the compressor. Therefore, it was natural that the original method of describing the nonuniformity of the flow would be in terms of the variation of total pressure. Since it is the variation of the velocity vector on the leading edge of the compressor blade that results in the major change in performance, it may seem somewhat unnatural that total pressure is still the major parameter in current distortion indices. Simplified relationships can be established between the change of total pressure and the change in the velocity vector relative to the compressor blade, using standard blade velocity triangles.

If uniform static pressure is assumed to exist at the face of the compressor, there is a direct relation between the total pressure and the axial velocity, depending on the Mach number of the flow at the compressor face. With a compressor face Mach number of 0.6, a variation of 1% in the total pressure represents a 2% change in the axial velocity; at a compressor face Mach number of 0.3, the same variation in total pressure represents an 8% change in the axial velocity. (The second case might exist during a lift engine start in flight, where there is apt to be a large pressure distortion at the compressor face while the engine RPM and axial flow velocity are very low, thus representing a very severe situation.) A change of 10% in the axial velocity represents a 2- to 3-degree change in the direction of the velocity vector relative to the rotor blade at high RPM and results in a slightly greater change in angle of attack at lower RPM when the ratio of the axial velocity to the blade speed becomes higher.

The assumption of constant static pressure at the exit of the diffuser, as used in the preceding discussion, is useful when evaluating the flow conditions at the exit of a diffuser without an engine. However, when the diffuser is mated to an engine, the compressor appears to impose upstream effects which alter the flow profile received by the compressor. The amount of upstream effect is related to the pumping characteristics of the particular compressor. It appears that the compressor actually receives a more nearly constant velocity with a static pressure distribution that matches the total pressure variations and a nonaxial flow that is compatible with the static pressure gradients. If, at some station upstream of the compressor, the static pressure is uniform and the distortion can be truly expressed as an axial velocity variation, the compressor alters the flow profile by pumping harder in the low velocity region, thus lowering the static pressure and increasing the dynamic pressure. The net result is that, at the compressor face, the distortion is represented by both a total and a static pressure variation which results in a more uniform axial velocity profile with transverse components. A change in the diffuser exit velocity angle of 10 degrees from axial results in a 2- to 3-degree change in the direction of the velocity relative to the blade.

The most common expressions used to describe the magnitude of distortion are the maximum spread in total pressure or the deviation from the average total pressure nondimensionalized with either the average compressor face dynamic pressure or the average face total pressure. Although these expressions are sometimes used to specify the distortion limits for even the most modern systems, they do not define the conditions that affect the engine performance and therefore are not adequate, unless other parameters are considered, for the correlation of distortion to performance. This fact and the realization that this correlation is required to successfully develop a compatible high-performance engine-inlet system without extensive, expensive, cut-and-try methods have motivated efforts to improve the methods of evaluating distorted flow conditions.

c. Distortion Indices Developed By Contractors

Flow models developed independently by competitive companies illustrate completely different approaches that resulted in very similar distortion indices.

(1) Company "A"

Company "A" first developed a theory to explain the mechanism through which flow distortions cause a shift in the surge line and a method of estimating the magnitude of this shift. (For a more complete description of this flow model and its application, see Item 2 of Bibliography). From this theory, they later developed their method for correlating the effect of circumferential distortion on axial flow compressors. The mechanism is explained using airfoil theory based on the influence of the "starting" vortex in determining the changes in blade angle of attack which will result in separation and stall. As the blading of a compressor passes through a nonuniform flow field, the angle of attack of the blades increases and decreases. If the angle of attack exceeds the stalling incidence of the blading, separation and stalling may occur, depending on the magnitude of the velocity variation, the duration of the condition, and the initial proximity of the blade to the stalling incidence angle.

Airfoil theory suggests that whenever there is a change in an airfoil's angle of attack, a starting vortex is formed at the trailing edge of the airfoil, which temporarily counters the change in strength of the blade circulation. As this vortex detaches and is swept downstream, the airfoil achieves the new full value of circulation. The force due to the starting vortex is not normally included when the forces which act on an airfoil are examined, since it is negligible after any appreciable length of time, but the vortex phenomenon does have a considerable effect on the airfoil while in the immediate vicinity of the airfoil. The transient effect of the starting vortex is to increase the flow into the leading edge of the blade as the angle of attack is increased, which temporarily prevents the blade from aerodynamically reaching the new physical angle of attack. As this vortex is washed downstream, the airfoil circulation reaches equilibrium, and the blade aerodynamically attains the physical angle of attack. Due to this lagging response to a sudden change in the blade angle of attack to the delay in the development of the boundary layer there is a definite interval between the time the blade exceeds its physical stall angle and the onset of stall. Thus, if a compressor blade exceeds the stalling angle of attack but the angle is reduced again before stalling occurs, the airfoil may not stall at all.

Based on this explanation of the effect of flow distortion, a parameter, consisting of blade chord length, rotor speed, and angular extent of the low velocity region, was developed to represent the displacement of the starting vortex during the time interval in which the blading is passing through the low velocity region. This parameter was then used in conjunction with the total pressure distortion to arrive at an expression for the percent of reduction in surge pressure ratio. The original distortion factor or index developed from this flow model was

$$C = \frac{P_{avg} - P_{min}}{P_{avg}} \times \frac{N_{ref}}{N} \times \frac{\theta^{+} \times \theta^{-}}{360} \qquad (1)$$

where the compressor inlet's pressure defect has been considered to be representative of the inflow velocity defect. The equation is explained as follows:

$\frac{P_{avg} - P_{min}}{P_{avg}}$ = the area-weighted average total pressure minus the area-weighted total pressure below average in the sector of the velocity defect at the compressor face expressed as a percent of the overall area-weighted total pressure (P_{avg})

θ^{-} = the angle subtended by the sector of the inlet annulus in which the velocities (total pressures) are less than the area-weighted average

θ^{+} = the angle subtended by the preceding sector of the inlet annulus in which the velocities (total pressures) are greater than the area-weighted average

$\frac{N_{ref}}{N}$ = comparison of the physical rotative speed of the compressor to an arbitrary standard in order to establish, in combination with $\frac{\theta^{-}}{360}$, a factor which represents the relative period of time during which a compressor blade is subjected to low flow velocities as it sweeps the circumference of the compressor inlet annulus.

The combination of these factors represents the time during which a compressor blade is subjected to the maximum angle of attack and the total change of angle of attack as the blades sweep a compressor inlet annulus subjected to circumferential flow distortion.

Although this index (Eq. 1) does not take into account the effect of radial distortion, it did tend to correlate, within acceptable limits, the steady state distortion patterns with engine stall. This index, originally applied to an installation with a bifurcated duct, which presented an almost pure circumferential contour, later correlated acceptably with certain installations but unacceptably with others. The following indices evolved in the process of attempting to correlate the distortion contours of later installations and to reflect the effects on a fan engine where applicable.

$$C = \frac{\left(\frac{W_a \sqrt{\theta}}{\delta}\right)}{\left(\frac{W_a \sqrt{\theta}}{\delta}\right)_{design}} \left(\frac{\Delta P}{P_{avg}} \cdot \theta^{-}\right)^{0.2} \tag{2}$$

$$C = \frac{\frac{P_{avg} - P_{min}}{P_{avg}} \left(\frac{\theta^{-}}{360}\right) + 0.5 \left(\frac{P_{avg} - P_{min}}{P_{avg}}\right)\left(\frac{\theta^{-}}{360}\right)_{(OD \text{ or } ID)}}{1.5} \tag{3}$$

$$C = \left(\frac{P_{avg} - P^{-}_{avg}}{P_{avg}}\right) \frac{\theta^{-}}{360} \tag{4}$$

The most recent index to be used by company "A" is

$$C = \frac{\sum \left[\frac{P_{avg} - P_{min}}{P_{avg}}\, \theta^{-}\right]_{ring} K_{ring}}{\sum K_{ring}} \tag{5}$$

where K is the weighing factor applied to each ring of probes.

The Equation (5) index applies a weighing factor to the value computed for each ring of probes; the effect is to evaluate a distortion at the hub over two times as severe as the same distortion located at the tip. The variation of this factor is illustrated in Figure 3.

These indices proved successful for certain applications but no claim is made to a general correlation. Both analytical and experimental work is continuing but this company considers their current accomplishments to represent only the first steps toward establishing a discription of an unsteady state distortion.

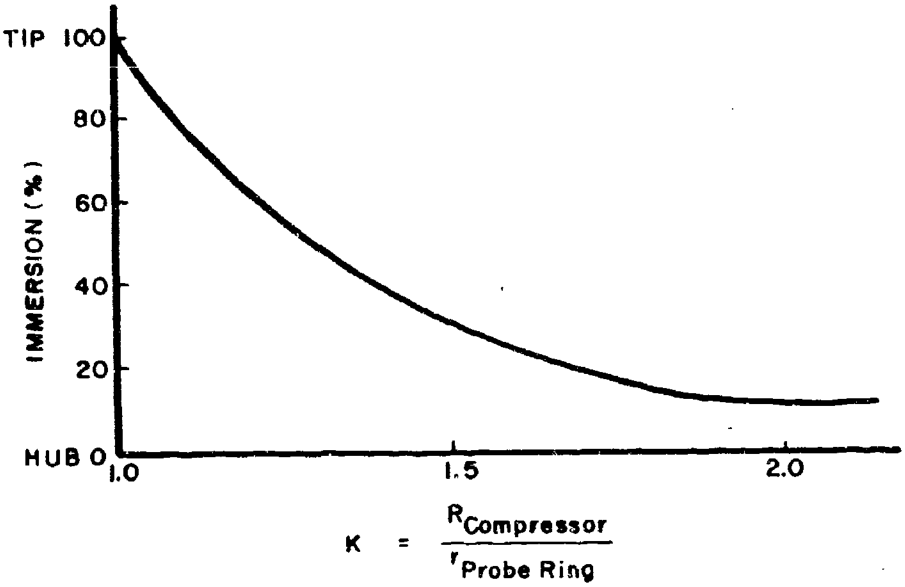


Figure 3. Radial Location Factor

(2) Company "B"

The Company "B" flow model is based on the physical concept that mass flow must move circumferentially in the axial gaps between blade rows in order to redistribute the excess flow from that sector of the compressor having an axial velocity greater than the average. *The model consists of circumferential variations in pressure with air flowing from the high-pressure regions to the low-pressure regions to "fill" the areas where the pressure is less than the average. The flow moves circumferentially over an average path length — the larger the area of distorted flow, the longer the average path length and the greater the number of stages within the attenuation length. By setting limitations on the velocity variation and on the absolute temperature change per stage, and by making certain assumptions as to the stage characteristics and the pressure drop versus the flow between the blades, an expression was derived for the number of stages within the attenuation length. (The attenuation length is defined as the axial distance from the front of the compressor for the distortion to decay to $\frac{1}{e}$ or 36.8% of the value at the compressor face.) The result of this analysis indicated that the number of front stages within the attenuation length is proportional to the square root of the product of the distorted area and the path length. The distortion index was based on the proposition that the effect of circumferential distortion on engine stall margin is proportional to the magnitude of the circumferential variation of total pressure and to the number of stages over which the distortion persists.

Company B's analysis was accompanied by an extensive amount of engine distortion testing, a wide variation of distortion patterns being used. As a result of these and later tests and

*This flow model and its application are discussed in Item 3 of Bibliography.

continued analysis, a series of distortion indices has evolved. These indices have been developed in an effort to correlate the steady state distortion index with compressor stall at various compressor corrected speeds and appropriate values of Reynolds Number Index. The indices are

$$C = \frac{P_{max} - P_{min}}{P_{avg}} \cdot \frac{\theta^{+}}{180} \cdot \frac{\theta^{-}}{180} \qquad (6)$$

$$C = \frac{P_{avg} - P_{min}}{P_{avg}} \cdot \frac{\theta^{-}}{180} \cdot \frac{\theta^{-} + \theta^{+}}{180} \qquad (7)$$

where

P_{avg} = average total pressure area-weighted over the entire annulus at the compressor face

P_{max} = maximum total pressure at the compressor face

P_{min} = minimum total pressure at the compressor face

θ^{-} = the circumferential angle subtended by the largest singly connected sector of the inlet annulus receiving flow having a total pressure less than P_{avg}

θ^{+} = the circumferential angle subtended by the largest singly connected sector of the inlet annulus receiving a flow having a pressure greater then P_{avg}

The first index to achieve a good correlation with engine data was established for an installation with a bifurcated duct which tends to present an almost pure circumferential distortion pattern with two areas of distortion per revolution. The generalized expression that resulted from the test data and the quantitative analysis of the physical flow model are as follows:

$$C = \frac{P_{max} - P_{min}}{P_{avg}} \sqrt{\left(\frac{\theta_1^{-}}{90}\right)\left(\frac{\theta_1^{+}}{\theta_1^{+} + \theta_2^{+}}\right)\left[\frac{\left(\frac{\theta_1^{-}}{90}\right)\left(\frac{\theta_1^{+}}{\theta_1^{+} + \theta_2^{+}}\right) + \left(\frac{\theta_1^{+}}{90}\right)\left(\frac{\theta_1^{-}}{\theta_1^{-} + \theta_2^{-}}\right)}{2}\right]} \qquad (8)$$

where

θ_1^{-} = the major low-pressure area expressed in angular form

θ_1^{+} = the larger high-pressure area adjacent to θ_1^{-}

θ_2^{+} = the smaller high-pressure area adjacent to θ_1^{-}

θ_2^{-} = the other low-pressure area adjacent to θ_1^{+}

The proceding distortion index is evaluated to ascertain the worst combination of two adjacent distorted areas when the compressor face contour has more than one area where the pressure is less than the average. When only one distortion area exists, this index reduces to

$$C = \frac{P_{max} - P_{min}}{P_{avg}} \sqrt{\frac{\theta^-}{90}\left(\frac{\theta^-}{180} + \frac{\theta^+}{180}\right)} \tag{8a}$$

Equation (8a) allowed an apparent discontinuity in the value of the index if a slight change in the average total pressure of the compressor face changed the distortion from a one to a two area evaluation. This is illustrated in Figure 4 where the circumferential pattern indicated by the solid line results in an index of 0.20 and a pattern modified (the dotted lines) just slightly, so that there are two areas of pressure less than the average pressure, results in an index of 0.14. This result occurs because the change from the 2 per rev to the 1 per rev pattern almost doubles the path length which results in an approximate $\sqrt{2}$ increase in the distortion index.

The next index to be used elminated the problem of the multiple areas of low pressure by evaluating each area of low pressure separately and arriving at the final index through the following formula:

$$C = C_{max} \sqrt{\frac{C_{max}}{(C_1 + C_2 + \cdots C_n)}} \tag{9}$$

where

C_{max} = the numerically-largest individual index

Equation (9) credits the distortion contours with more than one major area of low pressure but does not result in overemphasizing the effect of small secondary regions of low pressure. The index was further changed by adding a factor that introduces the average value of the pressure in each low-pressure region and by eliminating the length-of-path term.

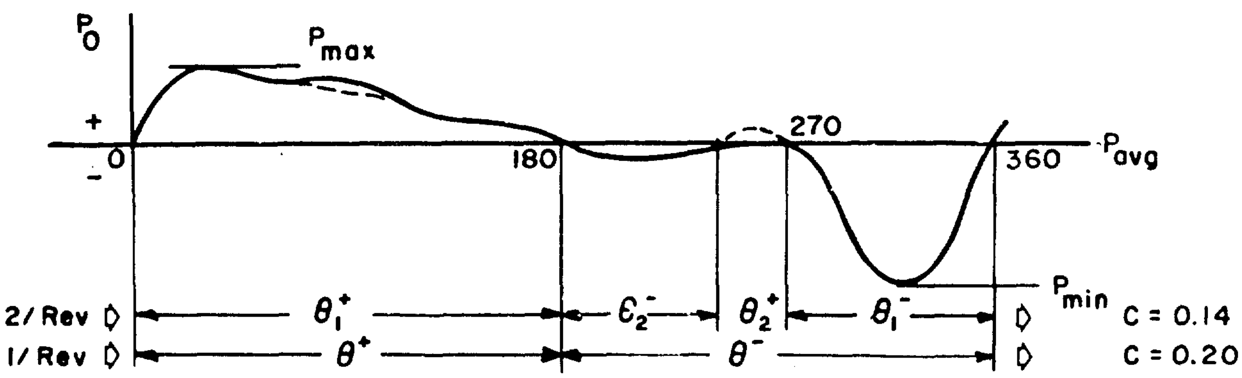


Figure 4. Circumferential Pressure Variation

The changed index is as follows:

$$C = \frac{P_{max} - P_{min}}{P_{avg}} \sqrt{(2A)\,\frac{\pi}{2}\left(\frac{P_{avg} - \bar{P}_{avg_{min}}}{P_{avg} - P_{min}}\right)} \qquad (10)$$

where

A = the ratio of an area of below-average-pressure to the compressor face annulus area

$P_{avg_{min}}$ = the area weighted average total pressure in A

The constants result in the first term becoming unity when a continuous area equal to one half of the annulus area is below the average pressure and the second term becoming unity when the pressure distortion in the area being evaluated is equivalent to a sinusoidal distribution.

The effect of the second term is illustrated in Figure 5 for several distortion hole-shapes as they might appear on either a circumferential or radial profile. The triangular pattern is 11% less severe (lower index) than the sinusoid, while the square pattern is about 25% more severe than the sinusoid. The accurate establishment of this profile requires a large number of data points (high probe density), particularly if the low-pressure area is small. Note that a pure hub or tip radial distortion would have the same index as a 1 per rev circumferential distortion.

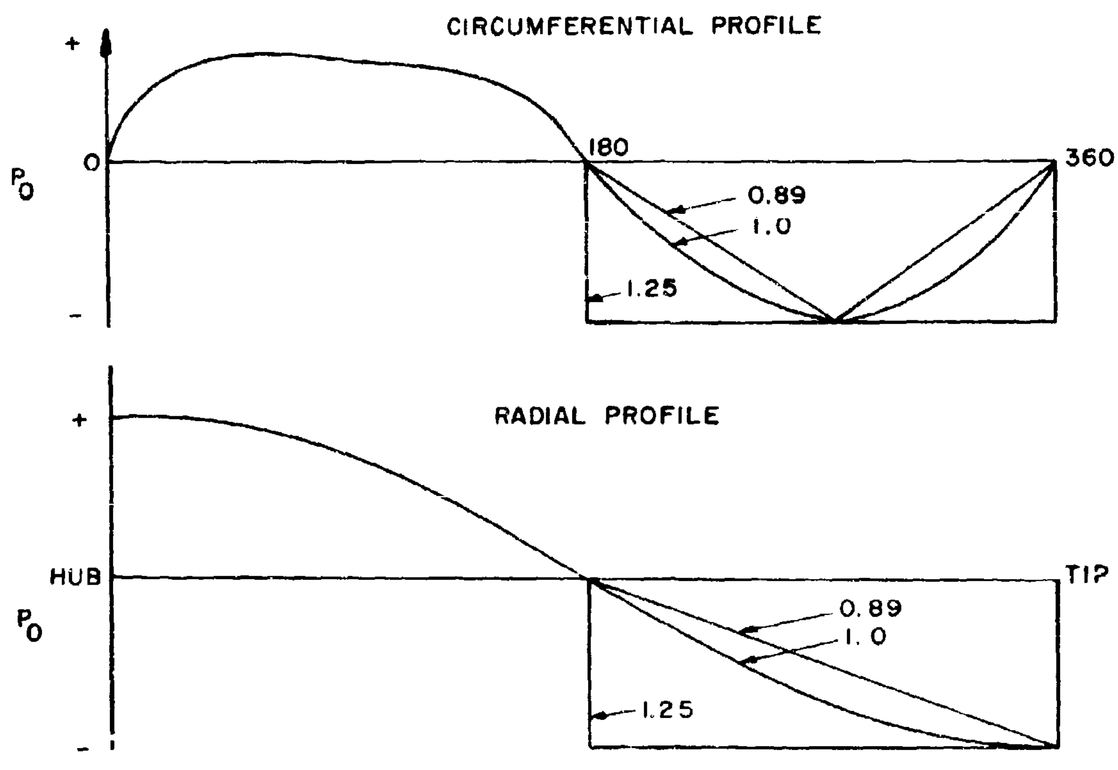


Figure 5. Circumferential and Radial Profiles

The next step in the distortion index evaluation was to include a term to reflect the radial location of the low-pressure areas resulting in this index (Equation 11):

$$C = \frac{P_{max} - P_{min}}{P_{avg}} \sqrt{(2A)\left(\frac{\pi}{2}\,\frac{P_{avg} - P_{avg_{min}}}{P_{avg} - P_{min}}\right)\left[1 + 2\,(|1 - 2AL|)\right]} \tag{11}$$

Where AL is the fraction of the low-pressure area being evaluated in the outer half of the annulus area of the compressor.

The effect of this term, as shown in Figure 6, is to express hub or tip radial patterns, $\sqrt{3}$ times more severe than circumferential patterns. The index has been recently modified to include the effect of RPM on the distortion tolerance of the engine, and an effort is currently underway to include the effects of the flow turbulence level on the distortion level.

(3) Company "C"

A third company has developed a slightly different approach to establishing a distortion index. (This approach is more fully discussed in items 4 and 5 of the Reference List.) A parallel compressor having characteristics similar to those of the main compressor is considered to be dealing with the distorted portion of the flow. Both compressors exhaust to the same static pressure, which means that the distorted section must operate at a higher pressure ratio. If this higher working point reaches the nominal surge pressure ratio, the parallel compressor will surge, and if the extent of the distorted region is sufficient, the combined real compressor would be expected to surge. This results in a simple relationship

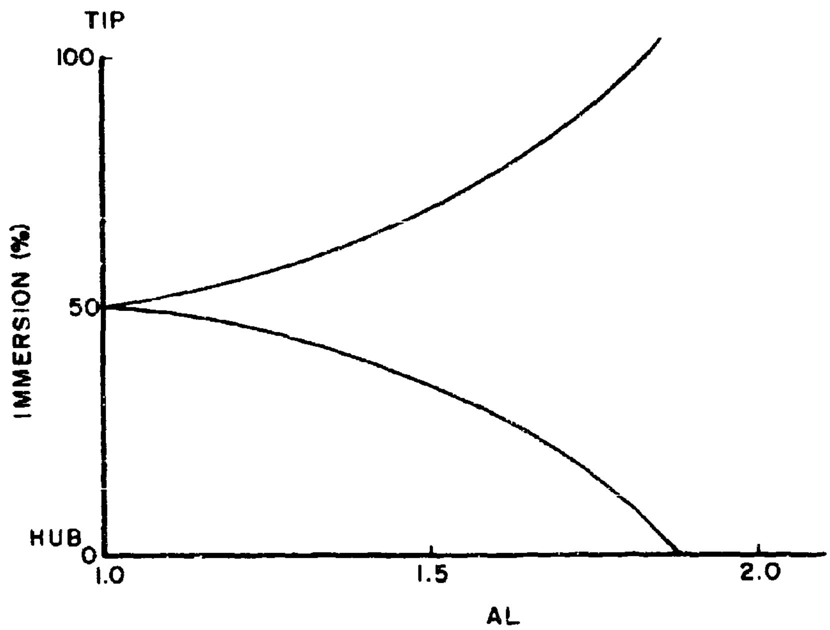


Figure 6. Radial Weighting Factor

between the pressure distortion and the change in the surge pressure ratio. To compare this loss in the surge pressure ratio to test data, it is necessary to choose an appropriate angular extent θ^- over which to determine the average total pressure loss that leads to a value for the effective pressure distortion ratio to be used in predicting the surge pressure ratio loss. Experience has shown that the lowest mean pressure over a 60-degree sector, when combined with the overall mean pressure, produces a parameter which gives a good prediction of the surge margin loss.

The preceding is useful for analyzing and comparing the results of different compressors, but the ratio is not as convenient in matching a given engine and intake as the pressure distortion coefficient, defined as the difference between the lowest average total pressure found in a 60-degree sector and the average total pressure over the compressor annulus divided by the average dynamic pressure at the compressor face. The coefficient is

$$C = \frac{P_{\theta^- avg} - P_{avg}}{q_{avg}} \qquad (12)$$

where

q = dynamic pressure

For a given inlet flow pattern, this coefficient remains essentially constant with varying compressor flow. Testing has indicated that pure radial and circumferential gradients of compressor face distortion affects the surge line in a different manner. Radial distortion appears to affect the surge line only slightly over the entire speed range, while the same amount of circumferential distortion may result in very large reductions in the surge pressure ratio. Therefore, Company "C" has established a distortion parameter that reflects only circumferential variation. A large number of compressor and engine tests have shown this index to be reliable within the above constraints.

(4) Evaluation of the Indices Developed by the Contractors

The indices developed by the three companies appear to be vastly different, but the fact that there has been some degree of success in correlating these expressions to compressor stall suggests that either these expressions are related or the compressor families must be unrelated. Certainly, there are differences in the design philosophies of the companies, but there is still a relationship between the indices. This is to say, that the expressions indicate there is some agreement in the first order effects of certain types of distortion patterns even though the magnitude of the values specified as the acceptable index limit to the engine is scattered from 0.05 to over 1400. As an example to show this trend, a comparison can be made with one element of the most recent indices developed by Companies "A" and "B."

The indices of both companies include a method of weighting the radial location of the low-pressure area, and the effect of this location on the index has been shown in Figures 3 and 6. If we superimpose these graphs, in Figure 7 we see that both companies agree that a hub distortion, for the engines and installations where these indices are used, is about two times as severe as a similar distortion located elsewhere on the compressor blade. This point is made only to demonstrate that, although the expressions appear to be vastly different, there are still major areas of agreement between them.

We emphasize, that since each index has been tailored to some degree to a particular type of engine or compressor design and may also be tailored to a particular type of installation or distortion pattern, a general comparison between the indices may lead to an erroneous conclusion.

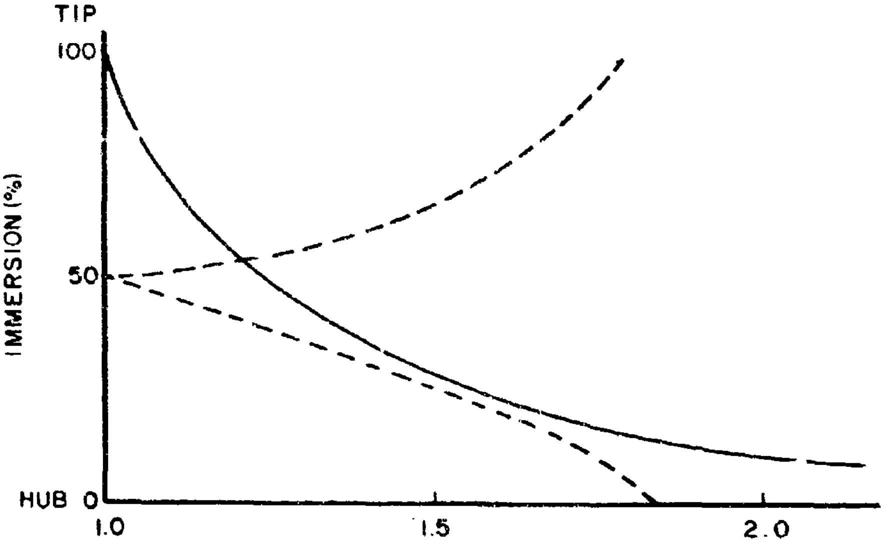


Figure 7. Radial Location Factors

3. NONSTEADY EFFECTS

Although care was taken in the early part of this section to define distortion so as to include true velocity and nonsteady phenomena, all of the indices presented have been based on steady-state total pressure, and there are no provisions in any of the current indices (Equations 1 through 12) to include these nonsteady effects. Part of the explanation for this apparent omission is that the effects of turbulence, vorticity, and swirl appear to be secondary to the main steady-state distortion effects, but they may be reflected in the next evolutionary cycle of improved distortion indices.

The engine "recognizes" as steady certain disturbances that are unsteady with respect to the airframe. This is true if the period of the disturbance is close to the time for one revolution of the compressor. If the duration of the distortion is long enough for all of the blades to pass through the disturbance, then certainly the steady-state criteria will apply and probably there is some minimum fraction of the total number of blades that must be disturbed before the compressor is affected. At any rate, it appears that instrumentation capable of responding to a minimum frequency equal to the reciprocal of the maximum RPM is required to evaluate just the steady-state conditions of the compressor.

In addition to these quasi-steady-state disturbances there are other dynamic phenomena, both transient ramp and oscillatory, that influence the stability of the engine-inlet system. Altitude changes, inlet or engine geometry changes, throttle changes, etc., are examples of ramp type transients. These are the types of disturbances normally analyzed by the various types of simulators for inlet and engine dynamics. Viscous effects within the inlet or on a surface upstream of the inlet also account for dynamic disturbances due to shock-boundary layer interaction, periodic transitory separation in the duct, or vortex shedding. There is a high frequency disturbance inherent to the rotating machinery of the turbine engine and, in addition, there may be cyclic disturbances due to rotating compressor stall, combustor instability, aerodynamic mismatch between the compressors, interaction between the engine and the engine control, or compressor surge. Another class of disturbance is acoustic oscillation, the propagation and successive reflection of low-amplitude pressure waves traveling at the speed of sound within the engine-inlet cavity, normally associated with organ pipe resonance.

Any one of the dynamic conditions mentioned may have an effect on the performance of the propulsion system, and, when they are occurring simultaneously or coupled to each other, the magnitude of the disturbance may be quite large and must be considered in determining the stability of the engine-inlet system. Analytical prediction of many of these disturbances currently is impossible and testing the engine and the inlet separately may identify as insignificant a disturbance which is of major consequence when reinforced by the interaction of the engine-inlet dynamics. Thus, even with major improvements in the capability of correlating engine face conditions with performance, there will be a continuing requirement for early realistic component test programs on the inlet and engine and, if a high-performance propulsion system is to be achieved before flight, an engine-inlet test will be required early in the development of the system.

The major accomplishments to date in accounting for nonsteady effects have been achieved through the use of inlet-engine dynamic simulators. There has been a considerable amount of work done to define the inlet-engine dynamics, using complex analog and digital computer programs. The programs are covered elsewhere in this report, but it should be mentioned here that, to date, the known dynamic simulation techniques are limited to a one-dimensional approach. The distortion indices are derived from the two-dimensional representation at the compressor face, but these have not been evaluated through the compressor. Even the steady-state capability is limited to an axisymmetrical solution.

A true three-dimensional solution of the compressor performance will be an extremely important step forward in determining the actual response of the compressor to distorted flow and in establishing the design tradeoffs between the inlet-engine stability margins and performance. The three-dimensional solution, along with a three-dimensional dynamic simulation of the engine-inlet system would be invaluable in predicting the behavior of a compressor subjected to both steady and nonsteady distorted flow conditions. Fortunately, the above problems are being attacked independently by several groups.

4. SUMMARY AND CONCLUSIONS

The efforts of the turbine engine industry to develop a parameter by which to correlate the flow conditions at the entrance of the engine with engine performance have resulted in a series of similar steady-state distortion indices. There has been a degree of success in correlating distortion patterns with compressor stall, but the trend is still for the engine manufacturers to set a very conservative limit to the amount of distortion the engine will accept.

Some distortion indices are tailored to a particular installation, and others are so general that an attempt to determine the acceptability of an engine in a different or modified installation is impossible, or at least cannot be made with any degree of confidence. As more of the factors that influence engine performance are included in the correlating expressions, the single-number index is bound to evolve into a series of curves, expressions, or perhaps into volumes of performance tables and graphs. The influence of swirl, vorticity, and turbulence must be established, and more representative parameters derived before an improved correlation with engine performance can be achieved. In addition, work must continue to enhance the analytical capability of the designer to predict the distortion tolerance of proposed engine designs so that early trades between the stability margin and performance can be accomplished.

SECTION III

CAUSES AND SOURCES OF INLET DISTORTION

1. INTRODUCTION

Improved performance and stability characteristics of turbojet or turbofan powered aircraft over an expanding flight regime can be achieved only if each component of the propulsion system is operating at the highest possible performance level that is consistent with a given overall system stability requirement. With increasing emphasis being placed on multimatch point missions, the stability and performance requirements have tended to precipitate propulsion system configurations in which these requirements apparently are inconsistent with each other. The induction system that supplies air to the engine, the engine's net addition of energy to the airflow, and the exhaust mechanism utilized for conversion of this net energy addition to thrust are the major components that must operate at high-performance levels over the expanding operational band while simultaneously maintaining a positive propulsion system stability.

The success or failure of a particular installation will depend on the degree to which the individual component performance and stability characteristics have been utilized when matched. Although extensive efforts have been expended in prior weapon system developments, initial successes in the area of aircraft-propulsion system compatibility have been somewhat limited to date. The problems associated with inlet-engine compatibility have required detailed definition and solution during early flight test of the weapon system in some instances. The forestalling of compatibility resolution until the flight test program has sometimes resulted in the continued assignment of aircraft and engines beyond aircraft number 5 to this problem when data on compatibility was required for operation of aircraft number 1. Table I gives an indication of some of the stability problems and resolutions that have occurred on some Air Force operational systems from about 1954 to the present time.

When the airflow capacity of an inlet does not equal the air demand of the engine, penalties to the propulsion system performance, stability, or both, result. If the inlet is too small, the engine is starved for air, and a thrust reduction occurs. This condition can also increase the inlet's distortion by virtue of the engine having induced too great a flow rate through the inlet. This increased flow rate and distortion characteristic could impose stability problems on the propulsion system in addition to the performance penalty mentioned. If, on the other hand, the inlet is too large, the excess air must be either discharged through a bypass system or spilled externally over the cowl, resulting in drag that is chargeable to the propulsion system. Since the engine demand normally does not vary over the mission profile to the degree that the inlet capacity varies over the same profile, inlet-engine-nozzle matching is generally not achieved at all flight conditions. Usually, the system will be designed to match at the vehicle's cruise condition where maximum range is desired. As a rule, this results in excess inlet capacity at other than design conditions and hence penalizes the propulsion system by causing an excess of bypass or spillage drag at these off-design conditions. Since emphasis is being placed on multimatch point missions, it becomes apparent that conditions previously regarded as off-design points assume increased importance and therefore the latitude for performance and stability compromises at these off-design points is being drastically curtailed.

This section of this report presents a rudimentary review of inlet-engine matching problems as they generally contribute to performance/stability and to delineate some of the causes and sources of inlet distortion. This will be followed by a discussion of a specific case study of the inlet stability of a particular model.

TABLE I

STABILITY HISTORY OF SOME OPERATIONAL USAF SYSTEMS

OPERATIONAL SYSTEMS		INSTABILITY PROBLEMS	FIXES
Fighter	I	Engine stall — level flight and maneuver	Compressor and inlet configurations
	II	Inlet instability, engine stall during maneuver	Compressor and inlet configurations
	III	Engine stall — weapon release, maneuver, inlet cross-feed	Inlet configuration, engine scheduling plus rematch
	IV	Inlet instability — engine stalls during afterburner (A/B) lights	Compressor changes plus rematch
	V	Inlet instability during A/B shutdown	Inlet control
	VI	None	None
	VII	Engine stall — weapon release	Engine and control scheduling
Bomber	I	None	None
	II	High power at sea level static and crosswind takeoff	Inlet configuration
	III	None	None
	IV	Minor inlet instability	Inlet control
Trainer	I	Engine stall with maneuver	Engine control changes
Cargo	I	None	None
	II	None	None
	III	Crosswind takeoff	Inlet configuration

2. INLET VARIABLE GEOMETRY

A survey of previous efforts indicates that the majority of experimental work on propulsion system drag generally has been restricted to drag studies of complete airplane configurations. This means that a wide variation of inlet geometric factors was not considered, and, more importantly, that a relatively small error in measurement of complete configuration drag could create a large inaccuracy in the drag of the propulsion system. For example, if spillage drag is 10% of the total vehicle drag, an error of only 1% in total drag measurement precludes meaningful spillage drag studies of many inlet geometric variations. Some recent work has been accomplished in the area of additive drag and is available to the aircraft and engine industry in general.

The additive drag discipline deals with performance achievement primarily and is influenced by geometric configurations such as ramp angle, cowl shape, fuselage boundary layer removal plates, etc. These geometric shapes, their scheduling, and their placement can readily influence inlet distortion patterns. This is particularly true if these geometric shapes and their variability at off-design points have had to be compromised by their placement on the aircraft and by weight-reduction considerations.

3. BUZZ — THEORIES ON ORIGIN AND MECHANISM

Inlet buzz has been defined as unstable, subcritical operation associated with fluctuating internal pressures and a shock pattern oscillating about the inlet entrance. These phenomena are intolerable structurally due to the high frequency and pressure amplitudes in both the duct and the engine. These fluctuations are operationally prohibitive because they produce periodic oscillation of thrust and may result in engine surge and/or burner blowout. Therefore, buzz and the threshold of buzz should be considered very important items in any investigation of inlet distortion and engine operational response thereto for any supersonic propulsion system.

The buzz cycle starts with the terminal shock located at its limiting downstream position which may be either at the lip of the inlet for an external compression inlet or just past the throat for an external-internal compression inlet and some disturbance which causes the shock to move to an unstable upstream condition. As the shock is expelled from its stable position and moves upstream, spillage and overall drag increase. In the limiting upstream position, the shock may detach from the centerbody, resulting in a large change in the amount of air flowing into the inlet. The shock reattaches instantaneously and moves, sometimes at a slower rate than when flowing out of the inlet, to its limiting downstream position, and the cycle is repeated. This motion is random but can be identified by an average frequency and amplitude.

Although no single theory has been advanced that satisfactorily explains all of the characteristics of buzz, a number of theories, some based on analogies to other forms of oscillations, have been proposed in an effort to explain this phenomenon. A few of these theories and their applications are explained in References 6 through 9.

4. INLET BLEED

Inlets that are designed for critical operation near maximum supersonic aircraft speeds may be required to operate subcritically with a detached shock at transonic aircraft speeds because of engine airflow requirements. Subcritical operation of such an inlet causes an increase in drag which, in turn, results in a penalty in aircraft performance. The most promising of proposed methods for reducing propulsion system drag that results from subcritical operation is a combination bypass-bleed system.

Tests of perforated diffusers have shown that perforations act as automatic valves which pass small amounts of flow for supersonic Mach numbers within the duct and large amounts of flow for subsonic Mach numbers within the duct. Perforations may be used to obtain subcritical operation without causing a detached shock in front of an inlet by spilling flow from inside the inlet. In most of the tests of perforated inlets reported to date, the perforations were drilled so that they were perpendicular to the axis of the model. Recent studies have indicated that perforations slanted in a rearward direction were employed to obtain a higher momentum recovery of the flow spilled through the perforations, and subcritical operation could be realized with less drag penalty than that resulting from detached-shock operation. Since the perforations act as automatic valves, the use of slanted perforations could provide some drag savings obtainable from the bypass system during subcritical operation without requiring the mechanical complexity of a bypass configuration.

Highly subcritical operation of most inlet types results in an unsteady flow phenomenon known as inlet buzz, which was previously discussed. If a bypass system is used, buzz can be prevented by spilling the required flow through the bypass. However, the control for this system would be extremely complex if the bypass nozzle were forced to open and close with every change in airflow. On the other hand, perforations fulfill the same function as the bypass system in preventing buzz and will operate automatically with no moving parts.

One disadvantage of an inlet configuration which has perforations in the throat to provide a stable subcritical operation is that some flow will be spilled through the perforations when the shock is located downstream of the perforations. However, this flow spillage does not represent a complete loss in engine thrust because the flow spilled is primarily boundary layer, and boundary layer bleed at the throat of a supersonic diffuser generally increases inlet pressure recovery. Due to the use of these perforated bleed systems for control of shock positioning, their orientation and quantity will affect the inlet's flow-distortion characteristics. Although they are primarily a control function utilized to influence performance, the perforated bleed systems will also affect inlet flow distortion by the boundary layer removal and buzz prevention mentioned in the immediately preceding discussion. Perhaps a better presentation of this distortion influence is to suggest that one consider the flow characteristics exiting a supersonic inlet if the bleed system were incorrectly positioned or of insufficient flow capacity.

5. SOME CONTRIBUTARY SECONDARY SOURCES OF INLET DISTORTION

a. Radius of Curvature

When an inlet is applied to a fuselage, it can become extremely difficult to maintain the smooth aerodynamic flow paths desired. Length and weight restrictions can shorten an inlet to the point where the area-contraction ratios and expansion ratios are attempted in too short an axial distance. This naturally leads to extreme radii of curvature with the inherent danger of wall boundary layer separation. These separations of flow will form an effective area blockage with resultant degradation of inlet performance, distortion level, or both.

b. Ancillary Equipment

The placement of the air scoops that supply such equipment as air conditioning, etc. within the inlet flow area can cause an effective inlet area mismatch during other than design point operation (that is, the design condition for a combination of the inlet, engine, and ancillary equipment). This mismatch can impose both stability and performance degradations on the propulsion system. If the mismatch is sufficiently large, the adverse effects can make the entire propulsion system installation suspect, however; this would represent an extreme situation, and this is generally not the case. Another illustration of adverse influence of ancillary equipment would be, for instance, the rejection of the oil-cooling heat load directly into the intake system. The above illustrations will both generate a local pressure depression immediately downstream of their location but, in the case of the oil heat rejection system, would also result in a local temperature inversion.

c. External Boundary Layer Ingestion

This source of inlet distortion is somewhat self-explanatory in that whenever the core flow of an induction system that supplies the engine and/or an ejector nozzle ingests a large amount of the low-energy fuselage boundary layer, the resulting low-pressure level can cause both stability and performance degradations. Boundary layer removal scoops are incorporated in most current high-speed high-performance aircraft and, therefore, due caution is required in the design and operation of the scoops. The scoops can and will become one of the major influences if sufficient attention is not given to the interface between the propulsion system and the aircraft, especially when operation across a broad operational spectrum is required.

d. Proximity to Disturbing Flow Fields

External flow fields that affect the entire propulsion system include the wing and pylon flow field; whereas the tail surface flow field influences the nozzle and ejector, etc. Aft-mounted engines, arm-pit installations, etc. can become extremely sensitive to these flow fields when proximity to such flow fields is not taken into account and given careful consideration in the design phase.

6. INLET CONTROL

The performance of installed engines is based on the highest level of performance that can be obtained at each flight Mach number without any degradation due to control tolerance. Because of inlet control tolerances, an inlet can not always be operated at an optimum point, and additional performance losses are incurred. For external plus internal compression inlets, the inlet control system must control inlet geometry (throat area) and the bypass door position if a door is utilized. If the throat area is too small, the throat Mach number will be too low, or if the bypass door is closed too far, forcing the normal shock up past the throat, the inlet will "unstart" causing rapid decreases in pressure recovery, increase in drag, and unstable operation which may result in engine flameout and severe disturbances to the aircraft. To minimize the chance of unstart, the throat area must be increased from the minimum allowable, and the bypass door must be opened to spill more flow and move the normal shock downstream away from the minimum area section. The resulting lower pressure recovery and increased drag further degrade engine performance. (A more complete discussion on supersonic inlet control is presented in following pages of this section, along with a specific illustration and discussion of an inlet-engine system designed to operate to Mach 3.) To attain the inherent advantages of internal shock compression at high flight speeds, the variable geometry type internal or mixed internal/external compression inlet becomes attractive but introduces the problem of shock expulsion or unstart. The inlet and duct pressure transients associated with unstart can become violent at high flight speeds and not only cause engine instability or malfunction but also result in undesirable interactions with the flight vehicle. The avoidance of these situations imposes stringent requirements on the inlet control system.

Where the inlet is of fixed capture area and is sized for the maximum flight Mach number, the excess air at intermediate flight speeds must be bypassed to avoid performance loss. The bypass system, itself, must contribute as little as possible to aircraft drag yet at the same time it may have to be of large airflow capacity to eliminate buzz and to assure inlet restart conditions during engine failure. To use a large capacity bypass system for accurate shock position control, designers must provide additional sophistication to separate the high-rate airflow requirements from those where slower rates are acceptable. This can be done by a separately actuated trimmer control on the main bypass. During shock control operation, the main bypass may operate at a low rate corresponding with aircraft acceleration and deceleration, while the trimmer bypass, of small capacity, has a fast response to handle the higher-rate smaller-amplitude transients. The two will only assume an equilibrium position at steady-state conditions. In one such system, the main bypass also can switch to a high rate during the restart and buzz elimination cycle to effect a more rapid recovery.

The control of a variable throat on a mixed compression inlet may be handled by selecting a closed loop system using the appropriate parameters or, if no performance compromise is involved, may be arranged on one or more scheduling parameters. Inlet performance may not be greatly affected at low to intermediate flight speeds by small changes in the throat position and can be scheduled in a linear manner from maximum available throat area at takeoff to an appropriate position for inlet starting at the starting Mach number. Requirements of scheduling and accuracy are not necessarily too critical although the flight mission, of course, will dictate the inlet performance required. Once the inlet is started, extremely fine accuracy and response are required of the throat control to maintain high performance. After starting, the throat area is reduced rapidly to increase the throat contraction ratio and improve performance. A typical schedule is shown on Figure 8, where the throat area is controlled by a signal from a local Mach number sensor.

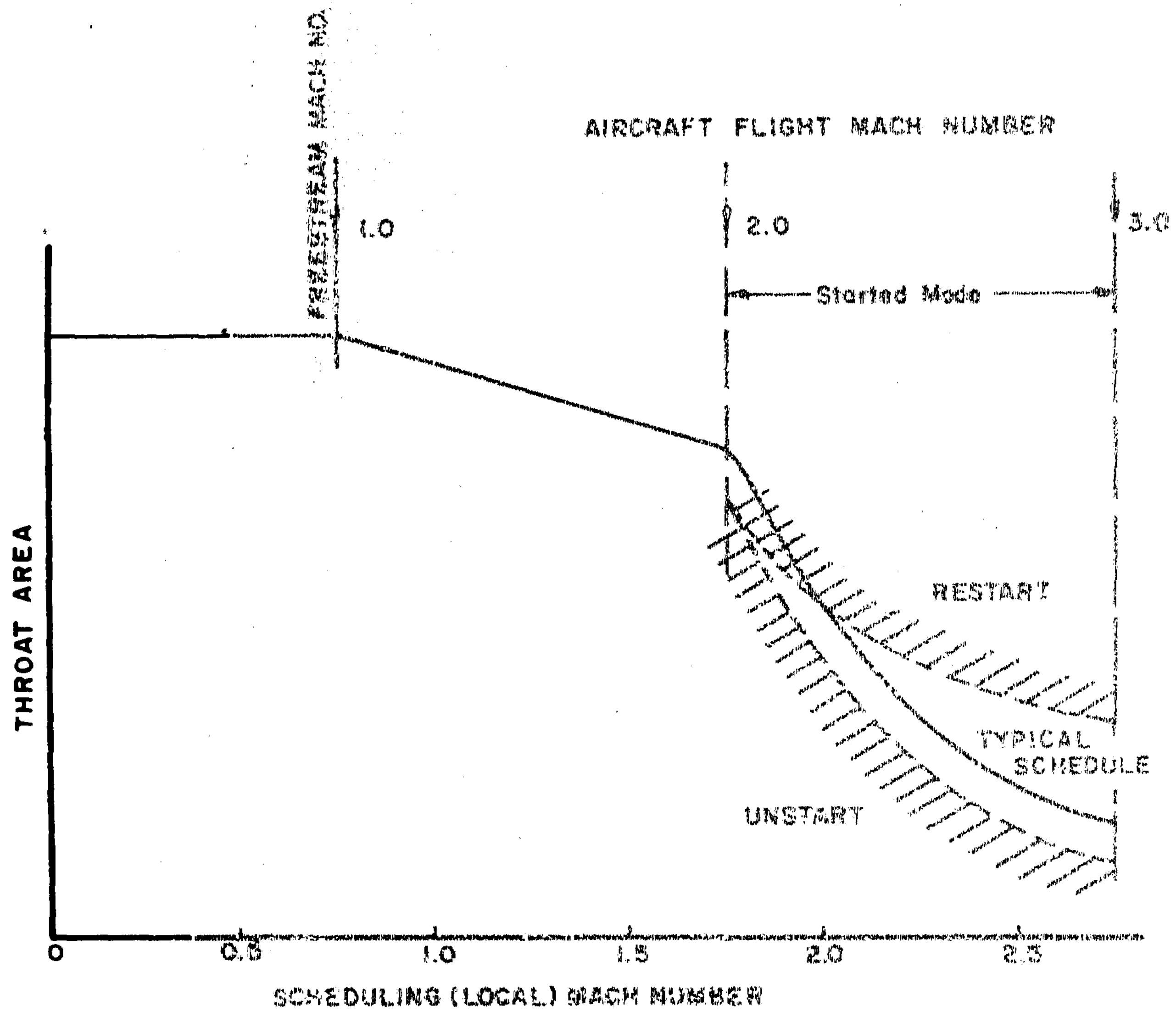


Figure 8. Throat Control Function

The importance of accurate throat positioning as it affects pressure recovery during the started inlet mode is shown in Figure 9. The left-hand vertical line represents the choking limit of the throat, i.e., an unstart occasioned by the inability of the inlet to pass all the entering air. The upper limit line represents the terminal shock unstart limit due to excessive upstream movement of the normal shock past the effective throat station, caused in the classical case by a diffuser outflow reduction resulting in a temporary mismatch with inflow. Significantly, these unstarts may also be induced at constant shock position by a throat-area increase, causing the initial or most forward stable normal shock position to move downstream as a result of the aft movement of the effective throat as the supersonic diffuser section Mach number increases. The slope of the normal shock unstart limit as a function of throat position becomes greater with increased Mach number within the started range. As the illustrations show, lines of constant shock position intersect both unstart limits, and the region between the limits becomes smaller as the normal shock moves upstream towards improved recovery and high performance. This emphasizes the importance of accurate and repeatable throat positioning to attain optimum performance, particularly at flight speeds at Mach 3 and above.

In the case of one particular system, at Mach 3, Figure 9 shows the steep reduction in unstart limited peak pressure recovery above approximately 110% of the unstart value of the throat position. Hence, for best performance, the throat schedule is set at 5% above unstart with control tolerances set on a maximum steady state positioning error of ±2% of unstart with an additional ±1% allowance for transient error. The allowable transient error and throat actuation rates are based upon average aircraft maneuvering rates in pitch and yaw.

Inlet flow characteristics change with variations in aircraft attitude in pitch and yaw, depending upon the inlet type and location. Entering Mach number, the relative positions of the oblique shocks and the quantity of air entering the inlet vary with attitude, thus affecting the average throat Mach number and its distribution and thereby affecting the unstart or choking throat limit. These changes must also be accounted for in determining the scheduled throat area variation and they also require some additional safety margin between unstart limits. This may require the schedule to be arranged for the normal shock to occur further downstream and compromise the performance in terms of pressure recovery.

Figure 10 illustrates the shift in unstart throat limit with angle of attack for one particular flight vehicle and shows a typical scheduled throat variation with the scheduling parameter — in this case, a local Mach number sensor outside the intake. An increase in angle of attack reduces the local Mach number which increases the throat area. The schedule level is based upon providing a 5% throat margin above unstart at $\alpha = 3°$. This arrangement provides a good margin above 2 degrees but none at $\alpha = 0°$. When the throat is not varied with angle of attack (as in the case of a variant where the scheduling parameter is flight Mach number rather than local Mach number), the smallest margin occurs at high angles of attack. In this case, for a 5% margin at 3 degrees, the inlet will unstart at $\alpha = 6°$. This example illustrates the importance in defining the best scheduling parameter in terms of safety margin and reference setting in relation to aircraft attitude changes and rates anticipated. It may become even more critical as flight speeds and maneuverability requirements impose additional problems and will call for more sophisticated control modes to satisfy stability, safety, and performance aspects. The scheduling parameter is, of course, compounded when inlet flow conditions change in a rather more complicated fashion than just as a function of angle of attack.

Figure 11 shows the combined angle of attack and yaw unstart limits for both the local and flight Mach number scheduled throats; again, both are based on 5% unstart margin at $\alpha = 3°$. The local Mach number scheduled throat provides a much greater yaw capability at $\alpha > 1°$. The aircraft of this example has side fuselage ramp type inlets and provides another illustration of the interface factors in that the selection of the local Mach number sensor for throat

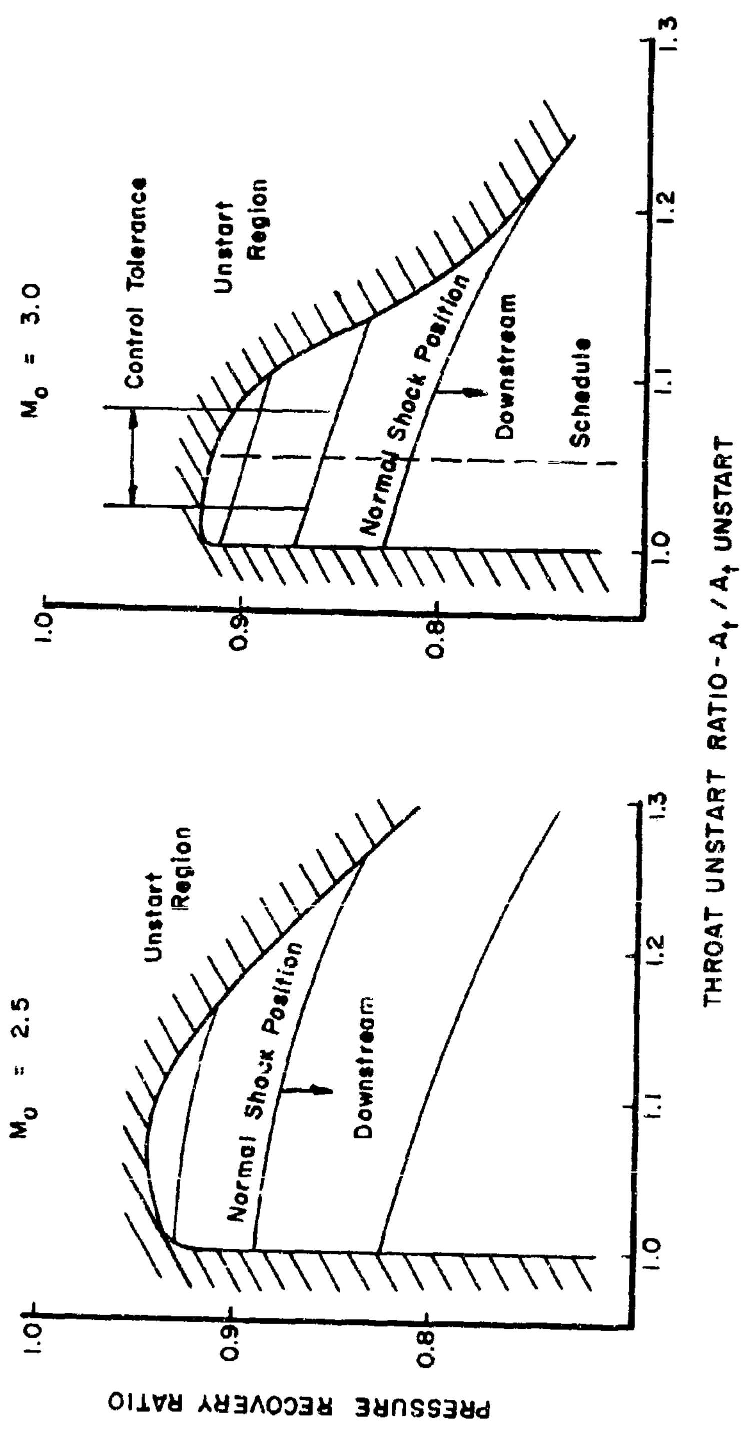


Figure 9. Effect of Throat Positioning on Pressure Recovery

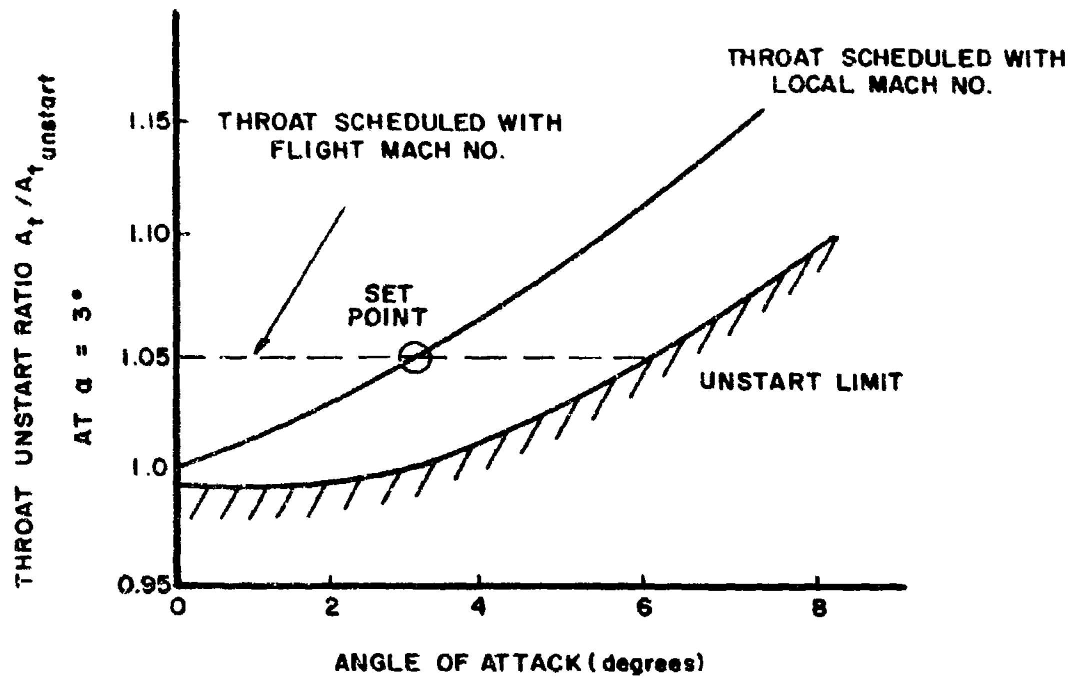


Figure 10. Effect of Aircraft Pitch Attitude on Unstart Throat Margins at Mach 3.0

scheduling permits the maximum unstart margin to be on the windward side at high angles of attack. In the aircraft installation just discribed, the local Mach number sensor satisfies aircraft controllability considerations since it is better to accept an unstart on the lee side and at low angles of attack, where the resulting forces then become restoring. Control of the bypass is accomplished as indicated on Figure 12. An optimum schedule for a standard day is shown for speeds that are less than those at which the inlet starts. In the aircraft considered, this function is replaced by an open loop step function with manual inputs to prevent duct buzz if engines are shut down. The step function causes some loss of performance at intermediate Mach numbers which may not be acceptable in other applications. At speeds above Mach 2, a closed loop shock positioning control is used initially to start the inlet and maintain the terminal normal shock at the desired location aft of the throat. The inlet is started by a command signal from the local Mach number sensor which activates the closed loop shock control. As flight speed increases, the bypass closes as required to maintain a constant shock position.

The shock position parameter employed in the example is a pressure ratio that varies continuously as the normal shock moves near the control point. An important requirement for this control parameter is that its gain within the control region be nearly linear or increasing as the normal shock moves upstream. This requirement can be satisfied by appropriate selection of control parameters and instrumentation techniques

After selecting the throat position for control, optimum inlet performance occurs with the normal shock in its extreme upstream position just prior to unstart. A performance back-off must be made for control tolerances. In the example illustrated by Figure 13, a total system tolerance of ±2% of the unstart corrected airflow means a minimum performance loss of 1-1/2% in pressure recovery at maximum flight speed. The total tolerance includes ±1/2%

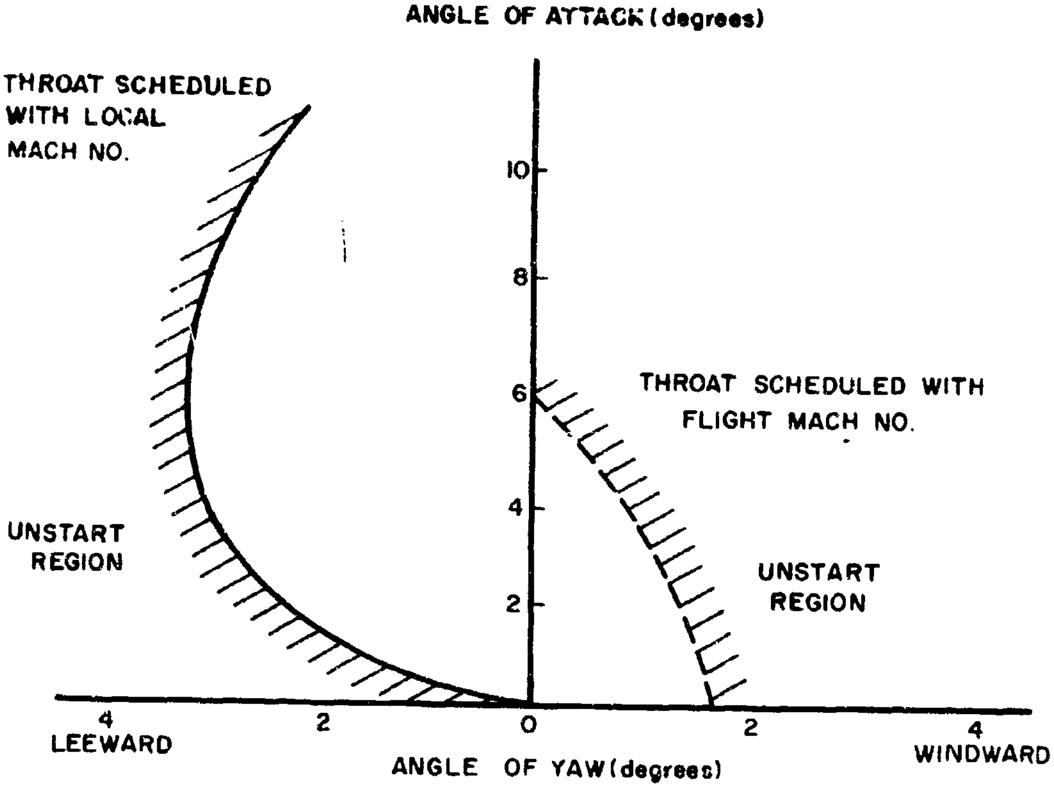


Figure 11. Effect of Aircraft Yaw Attitude on Unstart Throat Margins at Mach 3.0

allowance for steady-state positioning of the normal shock. This can be expressed in terms of the sensed parameter and its gain. The 1-1/2% allowable transient error applies to airflow change rates of 4-1/2% per second which do not result in normal shock excursion beyond the quasi-steady-state equivalent of 1-1/2% in corrected airflow. The transient requirement will be determined largely by aircraft maneuvering rates and also will depend upon the number of engines behind each inlet and engine throttle response rates, afterburner light-off allowances, and shutdowns. A high-performance aircraft will certainly impose stringent requirements upon the control system which, in turn, will need extremely sophisticated, accurate, and responsive control techniques, particularly with regard to transient tolerances, to avoid large compromises in performance potential while satisfying large transients.

The technique used in the example considered here, one employing a large capacity bypass, is to remove the large main bypass doors from the transient loop by reducing their rate and controlling transient disturbances by a high-rate, small-size, trimmer bypass. The airflow capacity of the small, trimmer bypass limits the magnitude of the airflow transients that can be controlled automatically. Since large amplitude transients are mostly pilot-initiated, they can be anticipated, and a higher inlet stability margin can be selected prior to the transient. This multiperformance feature is described in References 10 and 11 and illustrated in Figure 14. The feature provides the options of selecting high, medium, and low performance in terms of pressure recovery while allowing small, medium, and large stability margins, respectively. Such a system could be made automatic but would rely upon extremely fast-response control and anticipatory signals. In addition to performance compromises to allow for control system tolerances, an additional back-off may be necessary if the peak values of

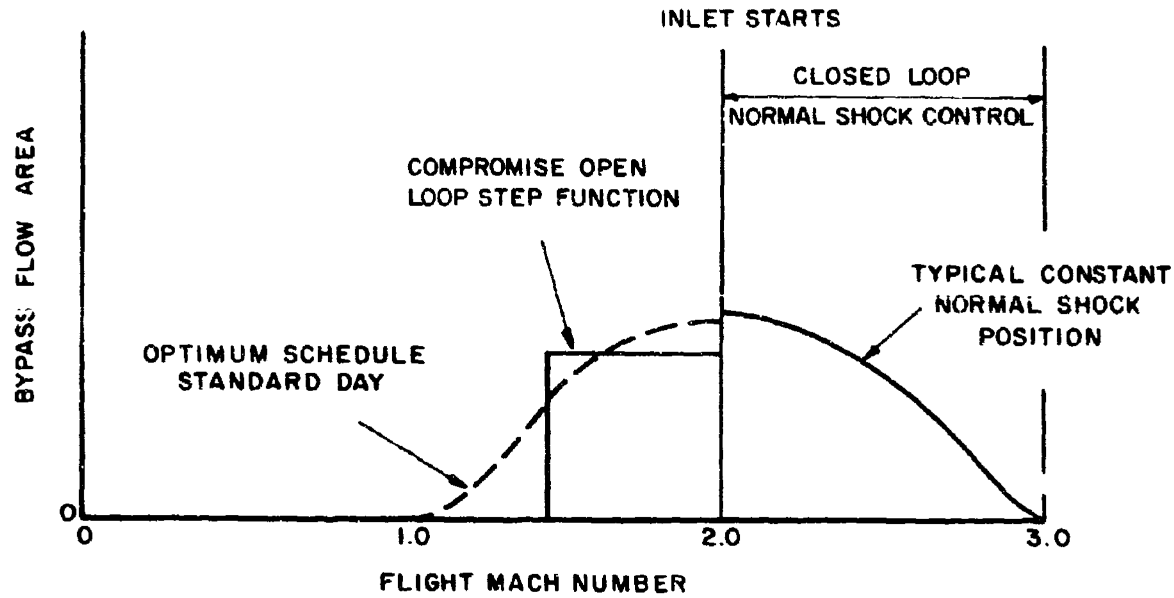


Figure 12. Bypass Control Functions

the shock position parameter vary with geometry and aircraft attitude. Figure 15 depicts a typical shock parameter and its peak value variation with angle of attack. Yaw angle changes show a similar effect. The decrease in peak value at high angle of attack is caused by a rearward shift of the effective throat. The example shows, in the shaded area, the performance decrement necessary to provide adequate margin up to $\alpha = 8°$. The selected control point is based upon design back-off at $\alpha = 8°$. At a lower (more typical) flight angle of attack, the inlet operates further below peak. Depending upon inlet type and location, the variation in the most forward normal shock position with aircraft attitude may be quite large and account for a significant performance decrement due to tolerances of the control system. Such considerations are most important in achieving high performance and should be emphasized. They must be thoroughly investigated in the early definition of any new weapons system.

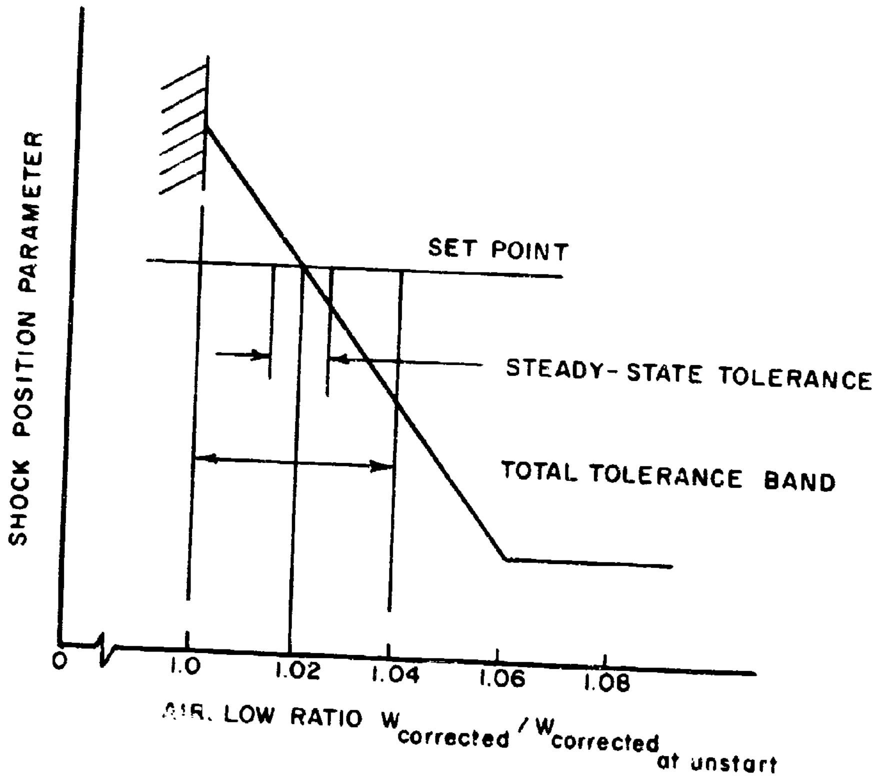
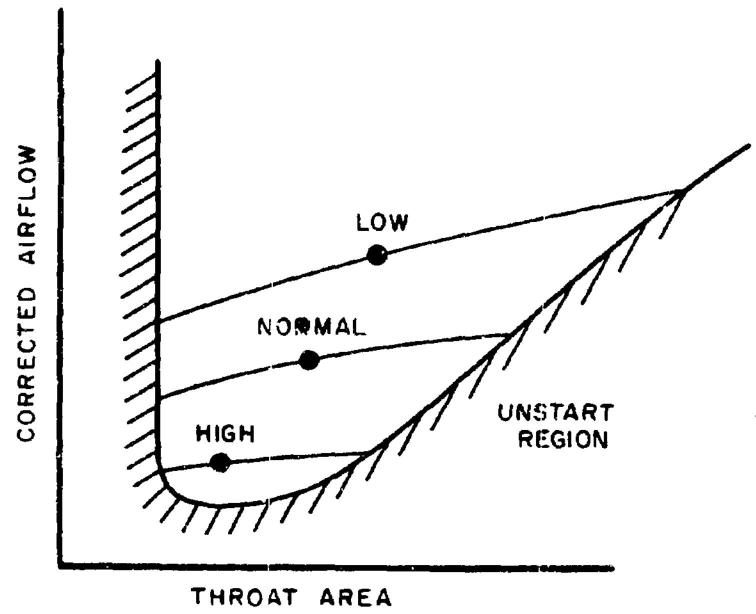


Figure 13. Normal Shock Parameter at Maximum Flight Speed



TYPICAL MARGINS

M_o = 3.0

PITCH = 3°

YAW = 0°

PERFORMANCE MODE	THROAT MARGIN $A_t/A_{t_{unstart}}$	SHOCK MARGIN $W_c/W_{c_{unstart}}$	PRESSURE RECOVERY H_2/H_0
HIGH	1.05	1.03	0.87
NORMAL	1.12	1.08	0.81
LOW	1.20	1.20	0.72

Figure 14. Multiperformance Model Function

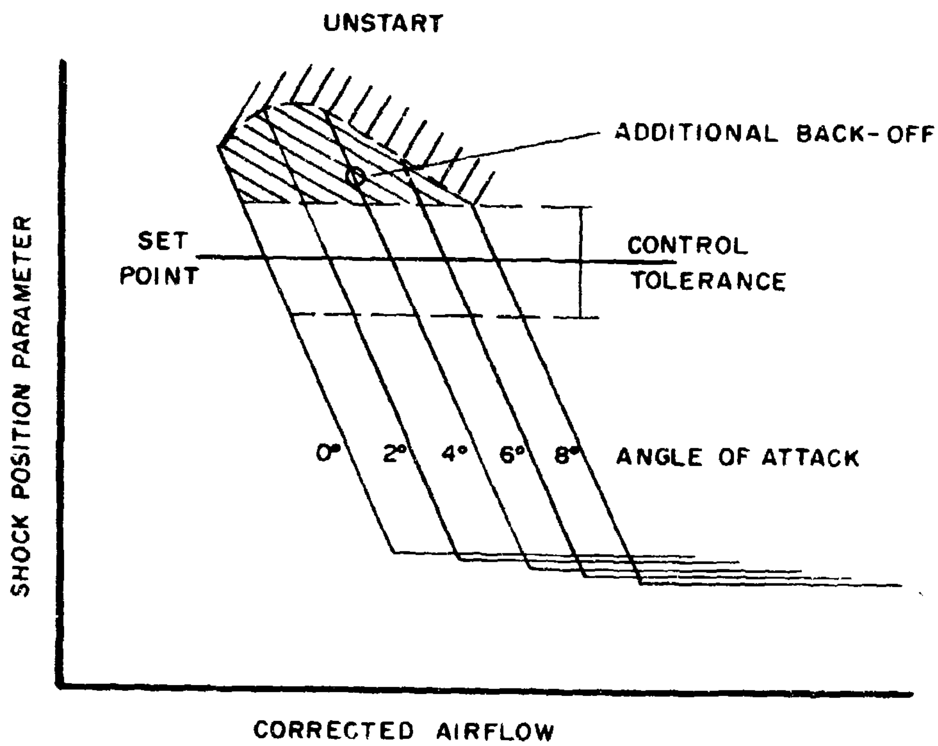


Figure 15. Normal Shock Parameter, Effect of Angle of Attack

SECTION IV

FACTORS THAT INFLUENCE SURGE MARGIN ALLOCATION

1. INTRODUCTION

Surge margin in a compressor is that allowance made between normal steady-state operation, over the speed range, along a matched working line and the limits of machine flow stability. Surge margin is usually defined as

$$\frac{\text{PR surge} - \text{PR operating}}{\text{PR operating}}$$

and is usually measured at a constant corrected airflow, corresponding to particular equilibrium operating conditions. Surge margin is illustrated by reference to the typical compressor characteristic map of Figure 16. Depending upon the machine characteristics and mode of operation, sometimes it may be more meaningful to define surge margin on some other basis, i.e., on the basis of a ratio of pressure ratios along a constant corrected speed line. In such a case, the flow modulation ratio between steady state and surge also may be a significant measure of machine capability. In any case, the basis of surge margin definition must be unambiguous.

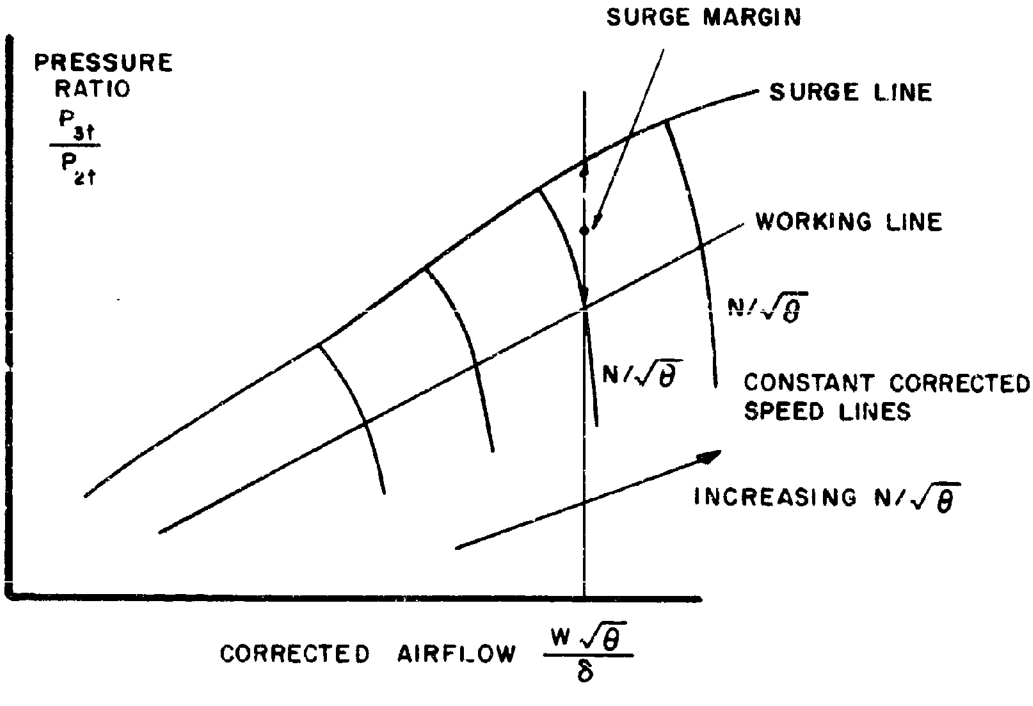


Figure 16. Typical Compressor Characteristic Map

For a machine of certain basic stability characteristics as determined by rig test, the best and most economical engine combination is made by attaining the highest overall operating pressure ratio. This means that, for life and reliability considerations which determine a maximum physical speed, high-pressure ratio operation can be attained at the expense of surge margin, thereby frequently compromising compressor stability. Because of the inherent danger in degrading stability characteristics, it is most important that the required surge margin for a particular application be defined absolutely. For such a definition to be valid, the detailed examination of the component parts or elements that together make up a satisfactory surge margin becomes an essential evaluation factor. It is no longer acceptable to make an arbitrary surge margin selection as has often been done in the past. Thorough and complete awareness of the constituent elements is a mandatory requirement. The various factors influencing surge margin selection are listed as follows:

ELEMENTS OF SURGE MARGIN ALLOWANCE

Control Mode

Acceleration/Deceleration

Afterburner Light

Fuel Control Tolerances

Component Variations and Manufacturing Tolerances

Variable Stator and IGV Rigging

Interstage Bleed Effects

Compressor Age Degradation

Reynolds Number Effect

Inlet Flow Maldistributions

2. ACCELERATION AND DECELERATION

In order to effect an acceleration from a given steady state point on the operating line, the fuel control system responds to throttle lever demand by injecting a metered excess of fuel. The excess fuel instantaneously raises turbine inlet temperature (T_{4t}) and the ratio T_{4t}/T_{2t}. Lines of constant T_{4t}/T_{2t} can be cross plotted on the compressor map as shown in Figure 17. The effect of an acceleration demand from a speed N_A corresponding to an equilibrium point A is to suddenly increase T_{4t}/T_{2t} which, in turn, provides excess turbine work to accelerate the rotating assembly. The increasing airflow then tends to reduce the rate of increase of T_{4t} until the machine returns to equilibrium at point B, at the demanded speed N_B, corresponding to the new power lever position and higher fuel flow and T_{4t}. The excursion of the operating point traces out a typical path AXB. The extent to which the operating point diverges from the equilibrium working line for a fixed geometry engine is very largely determined by the rate at which acceleration fuel is scheduled. This is normally a function of the time necessary for engine acceleration and, over the operating range of corrected speeds, will define a region or band above the working line. At no condition should the limit of this acceleration band encroach upon the compressor surge line.

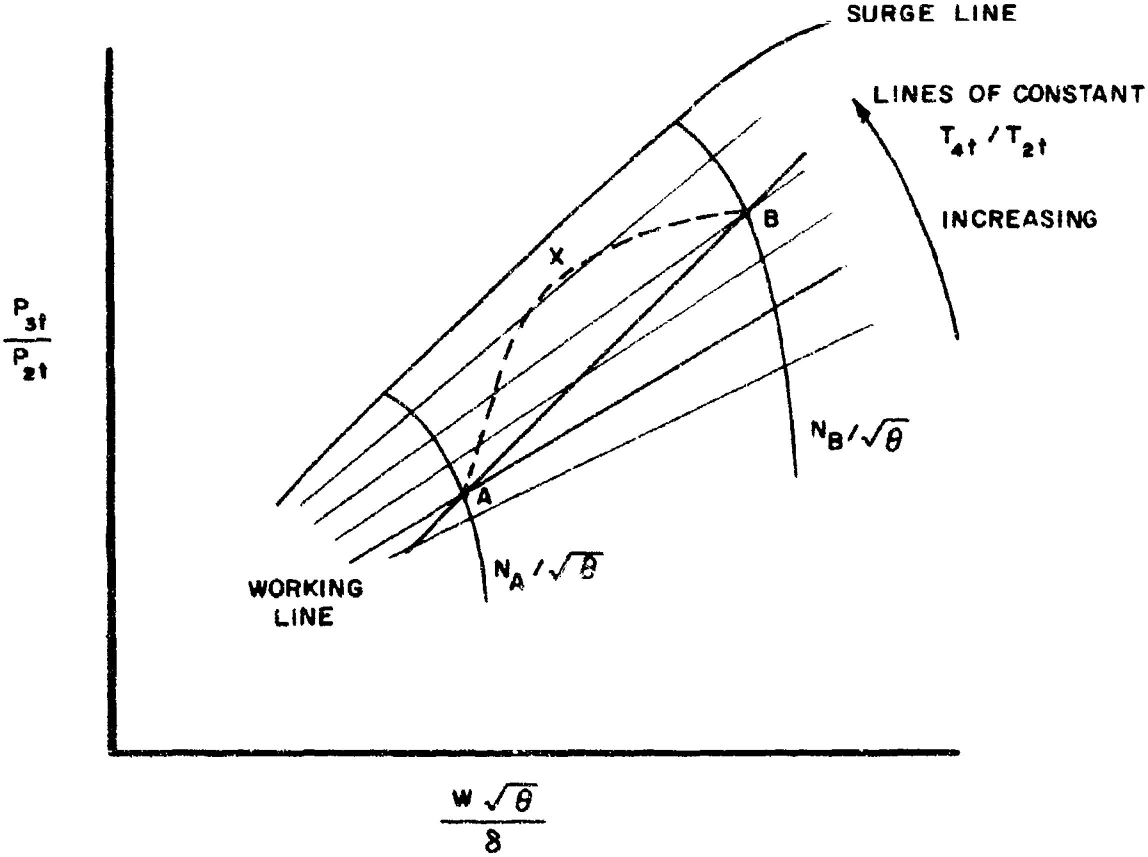


Figure 17. Effect of Acceleration

For the single-spool compressor considered here, the effect of a deceleration is to trace out an operating point excursion in the opposite sense, i.e., below the equilibrium working line. The problem in fuel scheduling for deceleration of the engine in the simplest case of the turbojet engine is imposed by the requirement to avoid the weak limit of combustion. In the case of a two-spool compressor, the deceleration case may additionally impose a stability constraint in terms of surge margin, since the operating point excursion traces a more complex path and may approach the surge limit of one of the compressor spools. This possibility is the result of speed mismatching between spools during transients and the inertia split between rotors. An absolute determination of the deceleration transient is necessary in allocating the appropriate surge margin within the critical speed range. It may be that the deceleration transient, at all conditions, falls within the allowance made for accelerations, in which case no additional surge margin will be required.

3. CONTROL MODE

The fuel control system is not normally a critical factor in the determination of engine flow stability characteristics, except where these may be compromised by combustion instabilities, particularly in the region near the rich and weak extinction limits. Present hydromechanical, prescheduled (open loop) type fuel control technology is sufficiently well advanced to provide the necessary accuracy in fuel scheduling and speed holding characteristics for most normal situations. However, in future applications, where less conventional techniques become increasingly employed and system complexity grows, the fuel control can become a most significant contributor to flow instabilities. The more complex propulsion systems will require more control functions and will include significantly more variables than the simple, fixed geometry, single-spool turbojet. Selection of appropriate control mode(s), interactions between various control loops, and characteristics of electronic and fluidic type amplifiers are all factors requiring detailed examination in terms of stability characteristics. The effectiveness of a future control system will also depend largely upon correct selection of control parameters and their sensing accuracy and response. A comprehensive control mode study becomes an essential requirement during the definition stage of any new system. The adequacy of the selected control mode in performing all necessary functions, biased by consideration of stability influences and surge margin allowance, must be specified along with any control loop interactions and the allowances made for sensing accuracy and response. This may be done best by utilization of advanced techniques in dynamic representation. The surge margin allowance for the selected control mode must be specified for all relevant conditions and influences.

4. FUEL CONTROL TOLERANCES

The control mode study discussed above, if properly conducted, will describe the engine response under all environmental conditions, including the worst conditions of imposed transients. The study should define those areas where stability compromises, due to control mode, may be effected by engine operating point excursion beyond the allowances made for normal accelerations and decelerations previously discussed. The appropriate, additional allowance must then be stipulated for fuel control system tolerances. Ideally, this should be defined by conducting a comprehensive tolerance analysis as an adjunct to the control mode studies. The tolerance analysis should be conducted on the basis of available test data and design information. The result of such a study, later supported by actual test data as component hardware becomes available, will be to define the areas, in terms of the compressor characteristic map, where additional surge margin is required to satisfy system stability aspects of fuel control component inaccuracies and manufacturing variations.

5. AFTERBURNER LIGHT

Future high-performance propulsion systems require large afterburner augmentation ratios, and an ability for the augmentor system to be fully modulating throughout the projected flight spectrum. These prime requirements tend towards pressure losses that are somewhat higher than those associated with current augmentor systems to maintain acceptable stability characteristics over the complete operating range. Large pressure losses in the augmentor system to meet full flight envelope performance and stability requirements will tend to make the engine more sensitive to the dry pressure loss when operating subsonically without afterburning. This sensitivity to dry losses has a tendency to nullify the advantages of higher engine pressure ratios and bypass ratios, particularly when the mission requirements dictate significant periods of subsonic flight with engine dry operation. In such applications there will be a temptation to minimize pressure losses in order to meet mission guarantees, which could easily compromise afterburner stability characteristics. It is most important, therefore, that afterburner stability be examined in detail.

Afterburner stability aspects will include all considerations of flame holding, resonance, flame vibration, weak and rich flame extinction, fuel to air ratio limits, and pressure (altitude - Mach) effects. In terms of surge margin allowance, one significant aspect is the case of afterburner light-up where the initial choking effect imposes a compressor (or fan in the case of a duct burning turbofan) flow reduction transient and a rise in pressure ratio along a speed line. This increase in pressure ratio is illustrated in Figure 18 for a duct burner behind a fan. In an ideal situation, it is possible to counter the effect of an afterburner

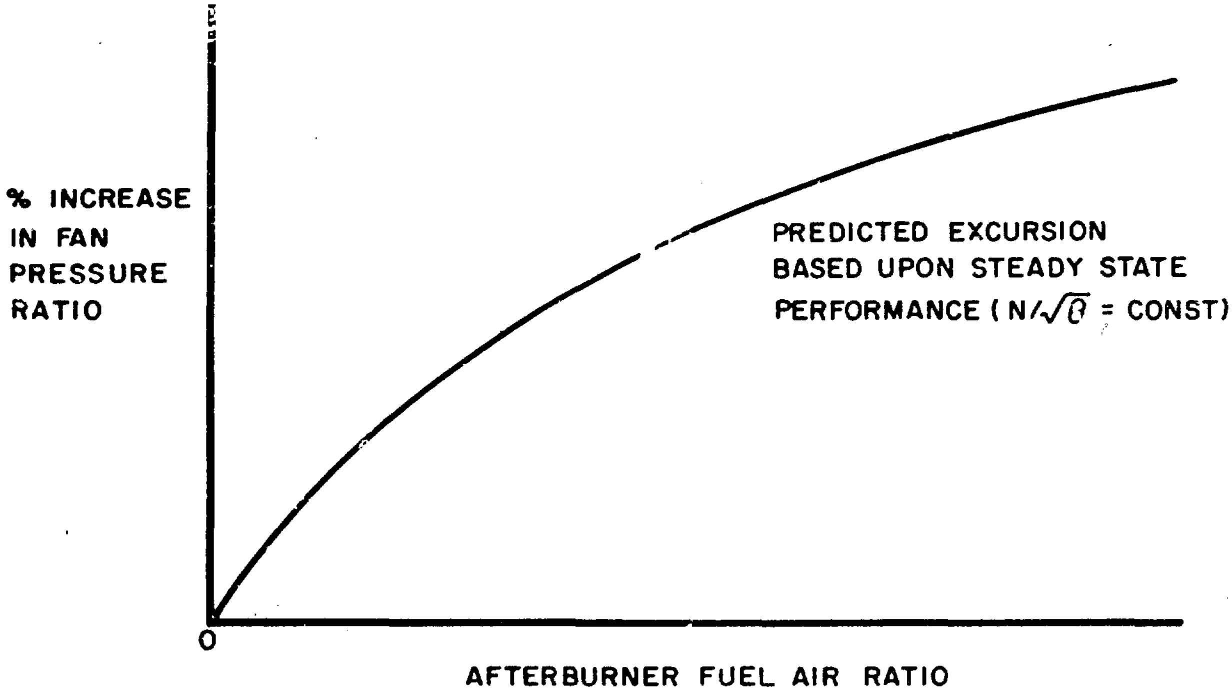


Figure 18. Effect of A/B Light on Fan Pressure Ratio

light transient by arranging for the operating point to shift to a corresponding pressure ratio reduction and avoid stability compromise by anticipating the light-up and increasing the area of the nozzle exit. In practice, it is most difficult to arrange for the two effects to balance exactly, and the appropriate allowance must be made in terms of surge margin for the actual case.

6. COMPONENT VARIATIONS AND MANUFACTURING TOLERANCES

Manufacturing tolerances, component to component variations, and engine to engine variations will influence individual engine stability characteristics to some extent. This is illustrated by reference to Figure 19, which shows the nominal or average engine's surge and operating characteristics and the perturbations due to tolerances, etc. about the norm. The illustration shows that the limiting case occurs with the worst compressor (or fan) and the highest pressure ratio matched set. The degradation in the nominal surge margin may not be large, but it is significant enough to require definition and the appropriate allowance. It should be possible to resolve this aspect analytically during preliminary engine design and then verify it by inspection of similar engine configurations.

7. VARIABLE STATOR AND INLET GUIDE VANE (IGV) RIGGING

Variable angle stator blading is frequently used in modern turbine engines as a means of shifting the surge line to a more favorable position for certain conditions with respect to the steady-state operating line. Variable stators are an effective means of matching the forward and rear stages of a compressor and providing improved machine stability characteristics over a wider operational speed range. Variable stator geometry, as a means of enhancing surge margin, is also employed to reduce the sensitivity of an engine to inlet flow maldistributions and the effects of rapid inlet temperature rise due to weapon firing or release. The latter case is illustrated in Figure 20 which shows the two surge lines corresponding to a nominal and reset stator angle and the two operating points. The

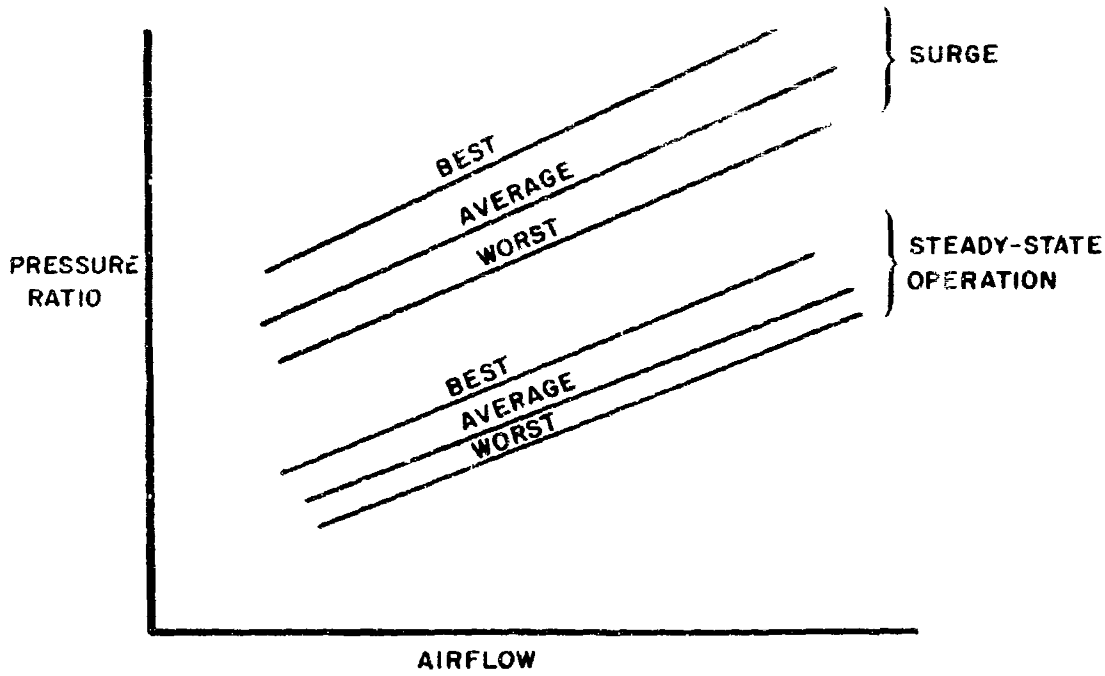


Figure 19. Effect of Manufacturing Tolerances

transient excursion of a rapid inlet temperature rise to 20% above steady-state causes surge at the nominal stator setting, whereas a higher, rapid temperature rise can be accommodated by resetting the stators to shift the surge line relative to the transient operating point excursion. It is necessary for the stator shift to be scheduled optimally and, in the case considered, to lead the temperature rise. This can be done by an anticipatory command just prior to weapon release. Stator scheduling to compensate engine stability for certain conditions of inlet flow maldistribution appears to be feasible within limitations. It is also possible to limit the extent of, or to prevent, the excursion of the engine operating point during accelerations or decelerations by lagging or leading the stator schedule as appropriate or in conjunction with temporary operating line adjustment by means of nozzle area variation.

Depending upon the mode of stator variation, the effectiveness of this means of surge control will be determined by such factors as rigging accuracy, response times,etc. Such factors must be identified, and their combined effect translated to the surge margin requirement. An accounting must also be made of the effect of rotor speed shift with compressor variable geometry.

8. INTERSTAGE AND/OR INTERSPOOL BLEED EFFECTS

In the case of a single-spool compressor, interstage bleed is frequently applied, as in the case of variable stators, to effectively rematch the front and rear stages at off-design speeds, thereby improving the range of stable operation. Since continuous interstage bleed imposes a performance penalty in terms of both thrust and specific fuel consumption (SFC), it is more usually used at low to intermediate corrected speeds and then as a method of allowing stable accelerations from part power conditions. The off-design performance of multistage axial compressors may be further improved by arranging for the compression process to be

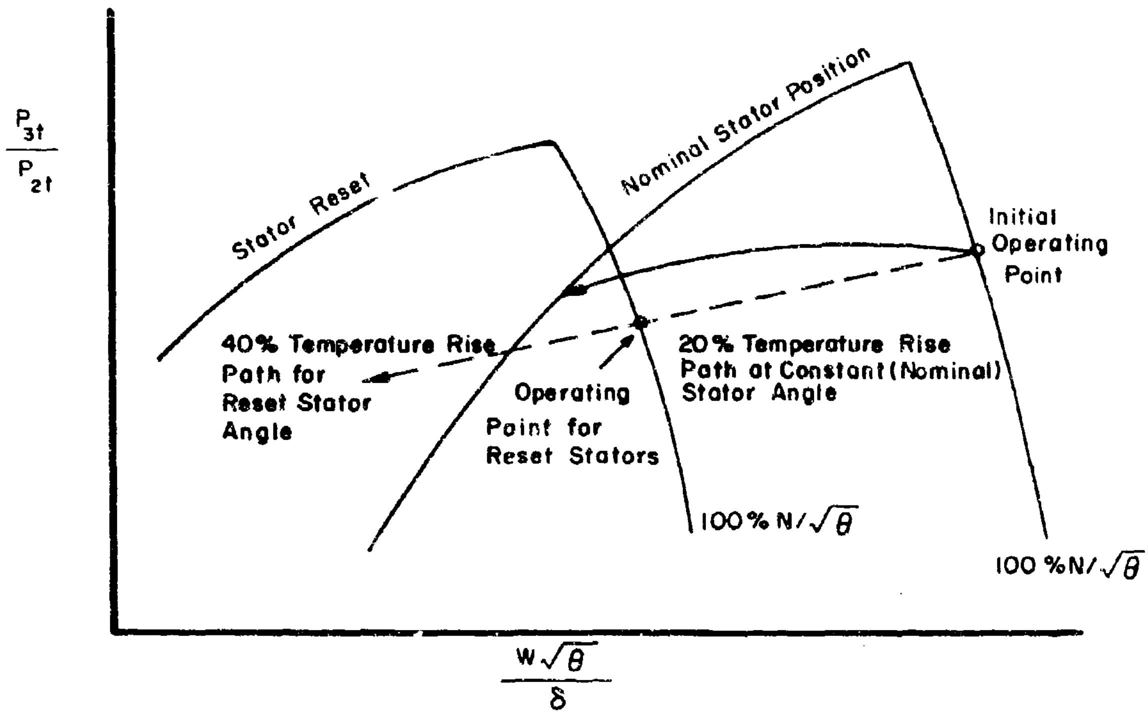


Figure 20. Stator Angle Shift for Surge Control (Rapid Inlet Temperature Rise)

performed in a split compressor with two spools operating independently. The improved off-design performance is attained by dual-spool speed matching over a suitable range, but the low and high speed limits of operation are ultimately reached with the onset of low- and high-pressure spool rear stage stall, respectively. In this case, a further extension of the "stable" speed range may be attained by bleeding between spools. For a high flight speed application where ram temperature rise forces engine operation at low corrected speeds, there may be a temptation to use interspool bleed continuously, with its attendant performance penalties, of course. The corollary to this is that the incentive to avoid performance compromise may tempt an engine manufacturer to exploit the stability tradeoff and disguise or ignore the fact. The analysis of bleed flow scheduling must consider this aspect when assessing the validity of the surge characteristics claimed for the particular design.

One other consideration in terms of engine stability and bleed is that of engine response times, particularly in the case of higher bypass ratio turbofans, where inertia coupling effects between spools may limit acceleration and deceleration times. The engine response times must be evaluated with respect to aircraft mission and engine requirements.

9. COMPRESSOR DEGRADATIONS CAUSED BY AGE

An additional small surge margin allowance generally must be made for the effects of age degradations where individual stage and overall stability characteristics may vary with dirt accumulation, changing tip clearances, seal leakages, etc.

10. REYNOLDS NUMBER EFFECT

The nominal compressor surge margin, as determined by normal sea level rig test, will shift as the Reynolds number changes in the flight environment. The appropriate element of the surge margin must be defined to account for the variation in Reynolds number index.

11. INLET FLOW MALDISTRIBUTION

a. Velocity Distortion

The velocity fields existing at the compressor face of an installed engine may be quite uneven, having components in axial, radial, and tangential directions. This nonuniformity may exist as a steady-state condition, but it usually occurs as a time-varying condition which can range to several hundred cycles per second. Engine behavior under the influence of cyclic input distortion becomes a function of the maximum pressure amplitude up to a frequency which is equivalent to the engine RPM. This critical frequency may be as high as 150 CPS for large engine designs. Sensitivity to transient conditions of nonuniformity became evident during recent engine distortion simulation tests. Engine test cell results could not be correlated with flight test findings until transient distortion levels to 100 CPS could be discerned and measured in flight and simulated as steady-state in ground engine testing.

In practice, a velocity pattern is usually represented by a total pressure distortion, total pressure being a more measurable quantity. The time-phased and positional variation of total pressure, therefore, is a measure of the engine input flow distortion. The total pressure may vary across the length of a blade while remaining essentially constant at any radius within the flow annulus, or it may vary around the annulus. The former case is known as radial distortion, and the latter as circumferential distortion, although in most actual cases the true pattern is a mixture of the two.

The compressor in the case of a turbojet, or the fan and compressor in the case of a turbofan, generally are the most critical engine components in terms of flow stability. An imposed inlet flow distortion will always tend to degrade the flow stability characteristics of fans and multistage axial flow compressors. The amount of stability degradation depends upon the flow pattern and the sensitivity of a particular component design, the geometric and aerodynamic design parameters to some degree determining the tolerance of a particular machine. While the exact mechanism by which a distorted flow affects the stability characteristics is not exactly definable, the net result is to effectively lower the undistorted fan or compressor stall limit line. The effect of this upon a fixed-geometry component is to reduce the margin between a steady-state engine matched operating line and the fan and/or compressor stall limit. This reduction of surge margin over the range of engine speeds will limit the safe area to accommodate accelerations, decelerations, transient disturbances, control inaccuracies, and tolerances. The surge margin loss can restrict the engine operating envelope and limit the stable operating range to the extent that the propulsion system might require extensive redesign for satisfactory operation.

b. Radial Distortion Attenuation

In the case of a radial pressure distortion at the inlet to the compressor, the degree to which the flow velocity profile is affected in its passage through the compressor will determine the pressure and temperature profiles at the combustor and turbine stages. A change in temperature profile at the first turbine stage, from the ideal design profile, can affect operation of the highly stressed turbine component in terms of life. The extent to which the flow profile is changed from stage to stage depends upon the compressor design. Aerodynamic design features, including stage loading, aspect ratio, and solidity, have been identified as affecting the degree of flow profile attenuation or amplification. More absolute correlation is required in this area. The position of the low-pressure region relative to the blade span will affect the blade loadings, the stage stall limits, and the stage matching characteristics of the compressor, although the design can conceivably be made to accommodate this aspect if the expected radial inlet distortion is specified early in the design.

c. Circumferential Distortion Attenuation

Again, depending upon the design features of the compressor, a circumferential inlet pressure distortion may be attenuated through the compressor, the machine tending to pump to an essentially common pressure around the circumference. The segment of the compressor immediately behind the low-pressure region of the distortion pattern will tend to operate at a higher overall pressure ratio than the segment behind a high-pressure region, and the outlet stage circumferential pressure may remain partially uneven, the effect depending upon the magnitude of the inlet distortion and the attenuation characteristics. For a uniform inlet temperature profile, the segmental pressure ratio differences will, with the exit circumferential pressure variation, combine to produce a distorted temperature profile which, in turn, will influence the combustion system pattern factor and may result in high peak temperatures at the turbine section. This again can be a critical factor affecting turbine life. Its quantitative effect will depend upon the magnitude of the inlet circumferential pressure distortion and the internal compressor aerodynamics. A qualitative result, for instance, may be the following. If the flow distortion persists through the compressor and manifests itself as a degradation of the combustor exit pattern factor, it may be necessary to de-rate the engine to a lower average turbine inlet temperature in order to achieve the desired engine life. In terms of steady-state engine performance, this, of course, will result in a reduction in thrust and probably in a specific fuel consumption penalty.

d. Effect of Distortion on Compressor Stall

Perhaps the most significant aspect of inlet flow distortion is its effect upon the compressor stall limit line. A secondary effect may be a loss in compressor efficiency as shown in Figure 21. A reduction in the stall limit will reduce engine acceleration/deceleration capability, limit the range of engine speeds for satisfactory operation, and impose a penalty in terms of high altitude, subsonic capability, and possibly high supersonic flight speed performance.

The exact mechanism by which the compressor stall limit line is affected by radial distortion pattern is somewhat complex. However, two gross effects have been observed during compressor testing. The first effect is generally to reduce the stall limit line, over the complete airflow range, to a lower pressure ratio, although in some cases, depending upon the magnitude and location of the flow deficiency, the stall limit line may be improved at certain speeds. Generally, the second effect is to extend the range of speeds over which rotating stall occurs up to a higher limit. Since radial distortion appears to have no effect on an engine matched steady-state operating line, the net result is to reduce the margin between this and the distorted stall limit line. This reduces engine acceleration capability and safe response to transient flow disturbances. The extension of the rotating stall region to higher speeds imposes limitations on the intermediate speed range, and in the extreme case, may prevent engine acceleration through this region. It also may introduce blade vibration problems if the higher frequency of rotating stall resonates with a blades natural frequency. Radial and inverse radial pressure distortions and the effects of blade hub, tip, or intermediate section stall on the compressor stage require correlation.

As previously discussed, under conditions of imposed circumferential pressure distortion, that segment of the compressor behind a low-pressure region will tend to pump a higher pressure ratio, for a given speed, than the average pressure ratio over the annulus. Depending upon the level of distortion and the subtended angle of the low-pressure, area-weighted

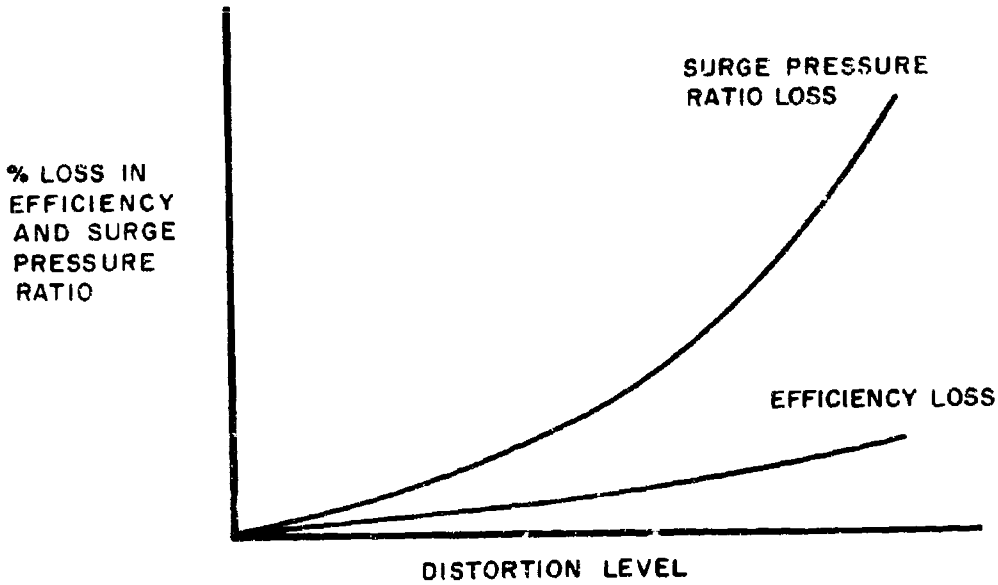


Figure 21. Distortion Effect on Compressors

segment, the segmental overall pressure ratio may reach the undistorted stall limit line. Past experience has indicated, that despite a possibly large segment of the compressor annulus operating at a relatively low overall pressure ratio, compressor stall or surge has occurred where the high-pressure ratio segment reaches the pressure ratio corresponding to the stall limit line. The average pressure ratio, at this condition, may be well removed from the stall limit. This particular phenomenon is quite well recognized. Such distortion indices as exist make use of factors which include pressure distortion, on an area-weighted basis, the annulus area, in terms of a segment angle for low- and high-distortion pressures and in some instances, term which relate engine speed. As a result of this generally accepted phenomenon, the effect of an inlet circumferential pressure distortion can be shown on the compressor characteristic performance map as a lowering of the stall limit or surge line. As in the case of radial distortion for a set operating line corresponding to fixed turbine and nozzle flow areas, the effect of a circumferential distortion will be to reduce the margin available for accelerations and to accommodate transient disturbances. The magnitude of the stall line shift will depend upon the individual compressor design and mode of operation. The effect, in gross terms, of a circumferential pressure maldistribution is usually greater than that of a radial distortion, although the degree of tolerance to each is some function of compressor design parameters.

e. Compressor Tolerance to Distortion

In the past, various attempts have been made, within industry, to obtain some correlation between compressor design and tolerance to inlet pressure distortion. These attempts have not met with any notable degree of success for the completely general case, and the usual method of ensuring satisfactory engine performance has been to set the steady-state operating line far enough away from the stall limit line. As will be appreciated, this margin is necessary to allow satisfactory accelerations, to accommodate transient disturbances, fuel control system accuracy, response and tolerance, and to allow for engine-to-engine variations and manufacturing tolerance. Frequently, the additional margin allowed for distortion effects is so poorly defined that the gross result is a matched engine which does not attain its complete performance potential. Apart from those other stability criteria requiring more exact definition, a precise knowledge of inlet distortion effects upon compressor stall limit could allow a sizeable reduction in the arbitrary margin, thereby regaining engine performance potential which otherwise is wasted.

The whole gamut of inlet distortion problems is thus seen as an integrated, though most significant, part of the approach to achieve an overall propulsion system flow stability. The overall flow stability approach may introduce the complete range of variables and could conceivably identify those system parameters most sensitive to disturbances in an optimization study of an engine performance. Techniques which include the more sophisticated aspects of dynamic modeling have shown a capability for handling such an optimization program.

It is most significant to note that variations in inlet pressure pattern can be handled satisfactorily only in an analytical study in which a fully representative dynamic model has been used to examine the discrete elements of the overall system and the particular time-phased processes occurring during an imposed transient.

f. Transient Distortions

The major portion of this discussion has been with respect to quasi-steady state operation under an inlet distortion pattern measured by total pressure variation over the face of the compressor. Additional stability problems are introduced when the distortion pattern becomes a function of time. The pattern may change or may remain essentially the same, but be alternately imposed and relieved with some particular frequency. The above situation, which is certainly a transient phenomenon, will occur with the operation of an engine inlet diffuser at off-design conditions. This condition is known as "buzz" and is associated with high-frequency engine inlet pressure oscillations. Buzz amplitude and frequency depend upon the particular inlet design and upon the operating conditions of the engine. The effect of buzz upon engine flow stability will depend upon the frequency and amplitude of the oscillation and the engine dynamic characteristics and stall limit. Such a transient situation can be handled by the development of a sophisticated mathematical model of the inlet-engine characteristics.

g. Inlet Temperature Distortion

The effect of an imposed compressor inlet temperature maldistribution has been demonstrated as causing a steady-state operating point to shift in the direction of the compressor stall limit. Methods which have been used in the past have generally been based upon a quasi-steady-state analysis wherein investigators have assumed that compressor pressure ratio departed from steady-state because of a change in the compressor's corrected speed combined with a change in turbine inlet temperature due to the airflow reduction at constant fuel flow accompanying the reduced equivalent speed. This method has introduced large errors in the limiting inlet temperature for stall and has, of course, no capability for analyzing the flow mechanism associated with a transient inlet temperature distortion imposed by partial reingestion of hot exhaust gases or weapon firing/launching effects. The techniques of dynamic representation with a comprehensive mathematical model will provide a more quantitative evaluation of inlet temperature ramps and the effect of nonuniformities upon engine stability. Such techniques require further perfection.

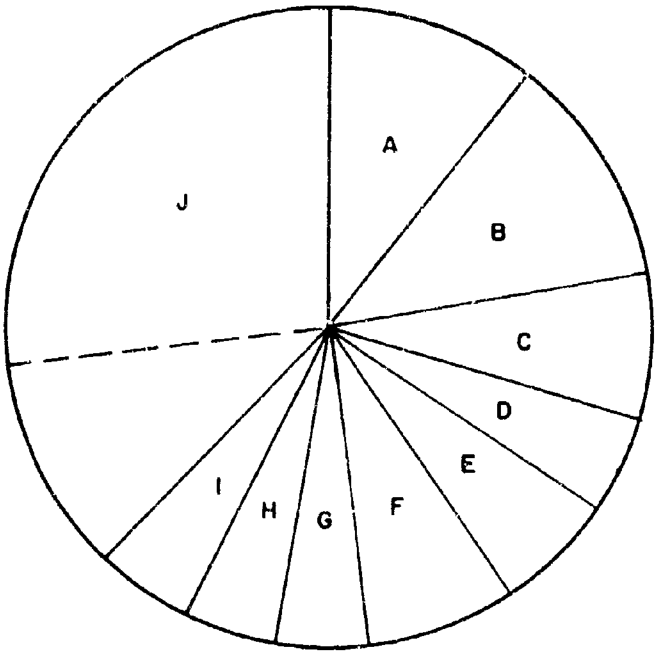
12. DEFINED SURGE MARGIN

The foregoing discussion on each of the considerations previously listed on page 33 (Elements of Surge Margin Allowance) has attempted to point out briefly and in gross terms how each factor influences engine stability criteria. More detailed inspection of individual factors can be found in many published articles and reports. The influences of the various factors are often interrelated in a complex fashion, and it is not always possible to isolate and treat them individually. However, for the purposes of surge margin definition, the effects of each item can be regarded as cumulative, and a properly defined analysis of surge margin will evaluate each factor, the various cross-influences and interactions, and fully account for each in the allocation of stability margin. The engine stability considerations, in terms of a properly defined surge margin, are represented by Figures 22 and 23. Figure 22 gives an indication of the relative importance of each item in terms of surge margin allocation, and Figure 23 is a qualitative representation of these items on a representative map.

13. ENGINE PERFORMANCE AS INFLUENCED BY COMPRESSOR STABILITY REQUIREMENTS

From the discussion on factors influencing surge margin allocation, the reader can realize that the amount of compressor surge margin that must be incorporated into an engine is dependent upon many items. Some method must be derived to distinguish the effects of each item on the overall engine performance characteristics. These stability-performance influence coefficients then define the compressor-engine working (operating) line which, in turn, implies the amount of surge margin inherent within the compressor in that particular application. No attempt will be made within this text to define absolutely these individual influence coefficients, since their precise derivation is primarily an empirical determination resulting from extensive testing of the unique article as tailored for the operational environment. Indications are that one of several situations has arisen when one inspects the final performance of an engine after "due" consideration has been given to all of the items discussed previously. A frequent observation is that there often is a significant difference in the surge margin required of two different engines even though they are called upon to perform over similar operational regimes. This can and often has been attributed to the internal behavior of the compressor or engine or the adverse installation or operational environment to which it is subjected. The problem becomes one of adequately discerning the major contributor to the requirement for the large difference in surge margin utilized in the two engines. An immediate observation is that an engine with a reduced surge margin is one that has had a rapid acceleration requirement relaxed. This can be a correct observation when considering earlier engines; the requirements of weapon systems, however, have generally tended to demand faster response time and will probably become even more stringent in the future. Therefore, attributing a substantial amount of the reduced surge margin to this extended acceleration factor can not be a consideration for latter day engines. The internal corrective measures for an engine, such as interstage or interspool bleed and variable geometry, are design tools that have been applied effectively to engine compressors to relieve the part speed, low corrected speed, stalling problems primarily associated with the front compressor staging. Some measure of the reduced surge margin match of the latter day compressors can be attributed to these factors. There still exists, however, a significient amount of surge margin difference that remains unexplained, and it is the perturbations about this difference in terms of engine performance that will be treated within this section.

It can be shown by appropriate treatment of the cycle parameters affecting specific power output and specific fuel consumption that for every flight Mach number, altitude, and turbine inlet temperature, the Brayton cycle will deliver a minimum specific fuel consumption at some reduced level of specific power. The prime cycle parameter that affects this minimizing of specific fuel consumption is the compressor pressure ratio, again, providing the turbine inlet temperature and the flight Mach number and altitude are held constant. The maximization of the specific power output of the Brayton cycle at some reduced level of pressure ratio and increased level of specific fuel consumption is explainable by the shift of turbine energy from compressor work input to energy available downstream of the turbine either for direct conversion to thrust or power extraction in terms of shaft power. The compressor pressure ratio is an operator that affects the thermal efficiency of the Brayton cycle, and its upgrading



A CONTROL MODE

B ACCELERATION / DECELERATION

C AFTERBURNER TRANSIENT

D FUEL CONTROL TOLERANCES

E COMPONENT VARIATIONS AND MANUFACTURING TOLERANCES

F VARIABLE STATOR AND IGV RIGGING

G INTERSTAGE / INTERSPOOL BLEED

H COMPRESSOR AGE DEGRADATION

I REYNOLDS NUMBER EFFECT

J INLET FLOW MALDISTRIBUTIONS
1. STEADY STATE
2. TIME VARIANT

Figure 22. Engine Stability Margin

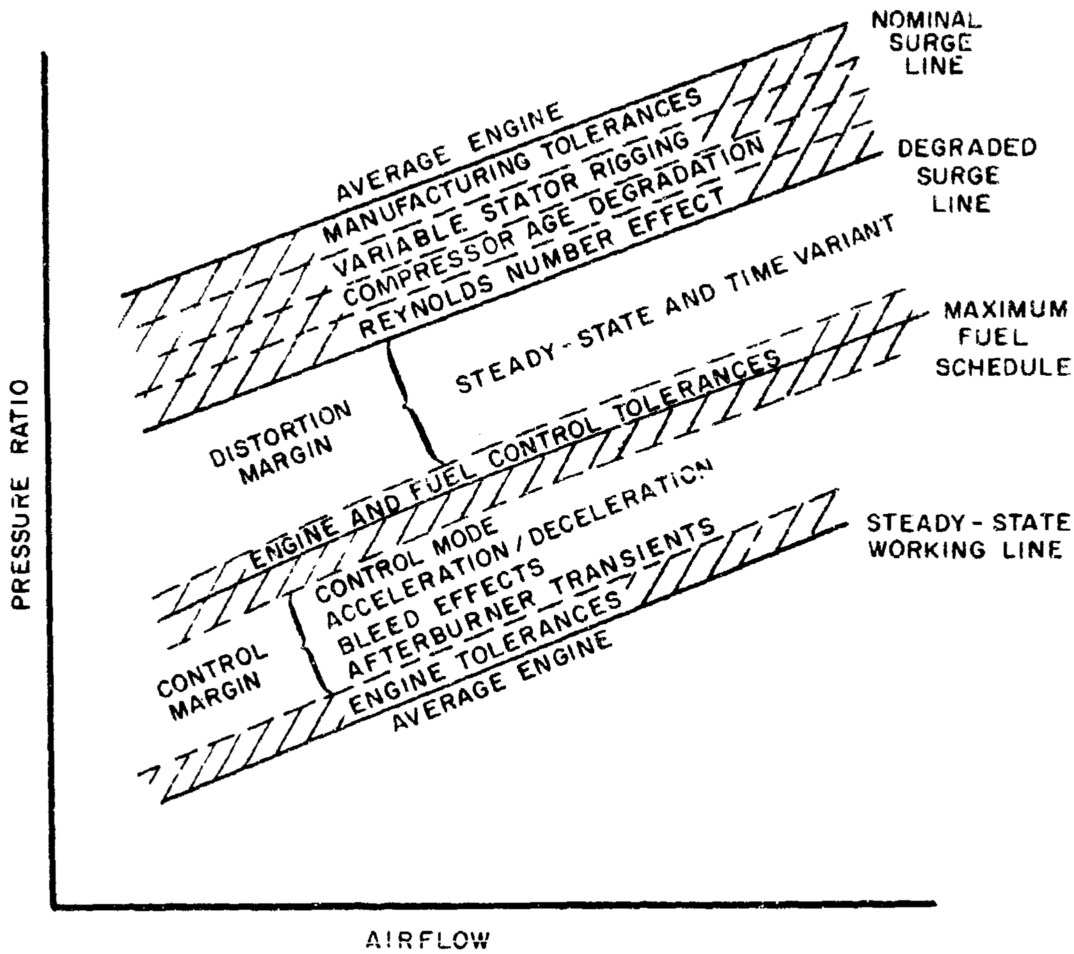


Figure 23. Cumulative Representation of Degrading Factors on a Compressor Performance Map

has been pursued actively since the advent of the Brayton cycle for the gains to be made in specific fuel consumption. Treatments of this analysis abound in the textbooks, and they all indicate that, for the Brayton cycle, neglecting the propulsive efficiency gains of the bypass principle for subsonic applications, the pressure ratio capability of turbo machinery has generally lagged behind the turbine inlet temperature capability up to flight Mach numbers in the neighborhood of 2.5. Flight Mach numbers in the vicinity of 3 to 3.5 require elevation of the current turbine inlet temperature capability. However, compressor materials and turbine cooling air temperature limitations start to have an influence on the usable cycle pressure ratio selection rather than on just the ability to achieve compressor pressure ratio. Having thus established the ground rules and desire for pressure ratio upgrading for flight Mach numbers to 3.5, it remains to establish quantitatively the effects of current surge margin matching practice and the reasons for the effects.

In one of the preceding paragraphs of this section, the statement was made that there is a large difference in the surge margin allowance afforded two different engines for the same general application. A performance perturbation has been made based upon given compressor maps and their respective operating lines. The intent was to establish the baseline performance of the engines with the given compressor performance characteristics and to compare an upgraded (reduced surge margin) operating line in the first engine, with the investigated baseline performance and degraded (increased surge margin) operating line of the second engine. For a representative operational band, the two engines' performances were investigated at sea level static at military power — Mach 1.9, 40,000 feet altitude at military power and Mach 0.8, 40,000 feet altitude at cruise power setting. The turbine inlet temperature for the military power setting was 2260°R and 1980°R for cruise power settings on both engines. The two engines investigated were a pressure ratio 9 turbojet and pressure ratio 12 turbojet. The pressure ratios mentioned above are for the military power setting at sea level static.

For the first performance perturbation, the 9 PR turbojet and the 12 PR turbojet utilized their compressor map match pressure ratio and efficiency combined with representative performance levels for the other components of the engines. The 9 PR turbojet was then investigated with an upgraded operating line and the 12 PR turbojet investigated with a degraded operating line. In both the upgraded and degraded case, the performance levels of other components were held the same as were those in the base (normal) operating line case. This was done to isolate the performance changes caused only by compressor influences.

Table II shows that for the 9 PR turbojet operating at military power at sea level static, a 4% increase in pressure ratio yields a 1.13% thrust reduction and a 1% decrease in specific fuel consumption. At military power, Mach 1.9, 40,000 feet altitude, a 6% increase in pressure ratio yields a 1.7% thrust reduction with a 0.5% decrease in specific fuel consumption, while at Mach 0.8, 40,000 feet altitude, cruise power setting, a 6% increase in pressure ratio gives a 0.2% decrease in thrust with a 1.3% reduction in specific fuel consumption.

Table III is the performance tabulation for the 12 PR turbojet which contrasts a degraded operating line against the base operating line from the compressor map. It shows that, for the same Mach number and altitude operational regime as for the 9 PR turbojet, the 12 PR turbojet has the following performance changes. At sea level static, military power, a 9% decrease in operating pressure ratio gives a 1.2% increase in thrust with a 2.9% increase in specific fuel consumption, while at Mach 0.8, 40,000 feet, cruise power setting, an 11.4% decrease in operating pressure ratio gives a 1.4% loss in thrust and a 3.5% increase in specific fuel consumption.

Tables II and III show the influence of shifting the operating line on given compressor maps, since the changes of the compressor efficiency are also incorporated in the tabulated performance. The analysis bears out the previous discussion pertaining to minimizing the specific fuel consumption with a simultaneous reduction in specific power output. The analysis is also in consonance with the statement that, in general, up to a Mach number of 2.5, the turbine inlet temperature capability exceeds the capability to achieve pressure ratio. Of even greater importance, however, the analysis points out the specific fuel consumption advantages that are being lost due to the requirement for matching compressors low on their performance characteristic maps. In both the upgraded and the degraded operating line cases, the specific fuel consumption trade is in favor of matching the compressors higher on their performance characteristics maps.

The above analysis assumed that the changes in compressor performance were the only influences and that the other engine components were unaffected. This, of course, is not the true case. As the compressor match line changes, the 12 PR turbojet operating line degradation influence was imposed on the operation of the combustor and the turbine. Figure 24 shows the variation of the combustor pressure loss versus the variation of the combustor flow function for lines of constant combustor temperature ratio. Figure 25 shows the variation of the combustor efficiency versus the variation of combustor temperature rise. Figure 26 shows the variation of turbine efficiency versus the corrected turbine specific work function for lines of constant turbine corrected speed percentages. Table IV shows the effects of the

TABLE II

UPGRADED COMPRESSOR PERFORMANCE EFFECTS FOR
A PRESSURE RATIO = 9 TURBOJET

Conditions	Cycle Parameter	Base PR Performance	Upgraded PR Performance
Sea level static — military rated power	$\frac{PR}{PR_{DESIGN}}$	1.0	1.04
	η_c	0.855	0.852
	T_4 - °R	2260	2260
	F_n - %	100	98.87
	SFC - %	100	99.00
Mach 1.9, 40,000 ft, military power	$\frac{PR}{PR_{DESIGN}}$	1.00	1.06
	η_c	0.855	0.847
	T_4 - °R	2260	2260
	F_n - %	100	98.3
	SFC - %	100	99.5
Mach 0.8, 40,000 ft, cruise power	$\frac{PR}{PR_{DESIGN}}$	1.00	1.06
	η_c	0.855	0.847
	T_4 - °R	1980	1980
	F_n - %	100	99.8
	SFC - %	100	98.7

Code: PR = pressure ratio
η_c = compressor efficiency
T_4 = turbine inlet temperature
F_n = net thrust

TABLE III

DEGRADED COMPRESSOR PERFORMANCE EFFECTS FOR
A PRESSURE RATIO = 12 TURBOJET

Conditions	Cycle Parameter	Base PR Performance	Degraded PR Performance
Sea level static — military rated power	$\frac{PR}{PR_{DESIGN}}$	1.0	0.91
	η_c	0.81	0.795
	T_4 - °R	2260	2260
	F_n - %	100	101.2
	SFC - %	100	102.9
Mach 1.9, 40,000 ft, military power	$\frac{PR}{PR_{DESIGN}}$	1.0	0.886
	η_c	0.835	0.819
	T_4 - °R	2260	2260
	F_n - %	100	101.2
	SFC - %	100	102.1
Mach 0.8, 40,000 ft, cruise power	$\frac{PR}{PR_{DESIGN}}$	1.0	0.886
	η_c	0.855	0.819
	T_4 - °R	1980	1980
	F_n - %	100	98.6
	SFC - %	100	103.5

Code: PR = pressure ratio
η_c = compressor efficiency
T_4 = turbine inlet temperature
F_n = net thrust

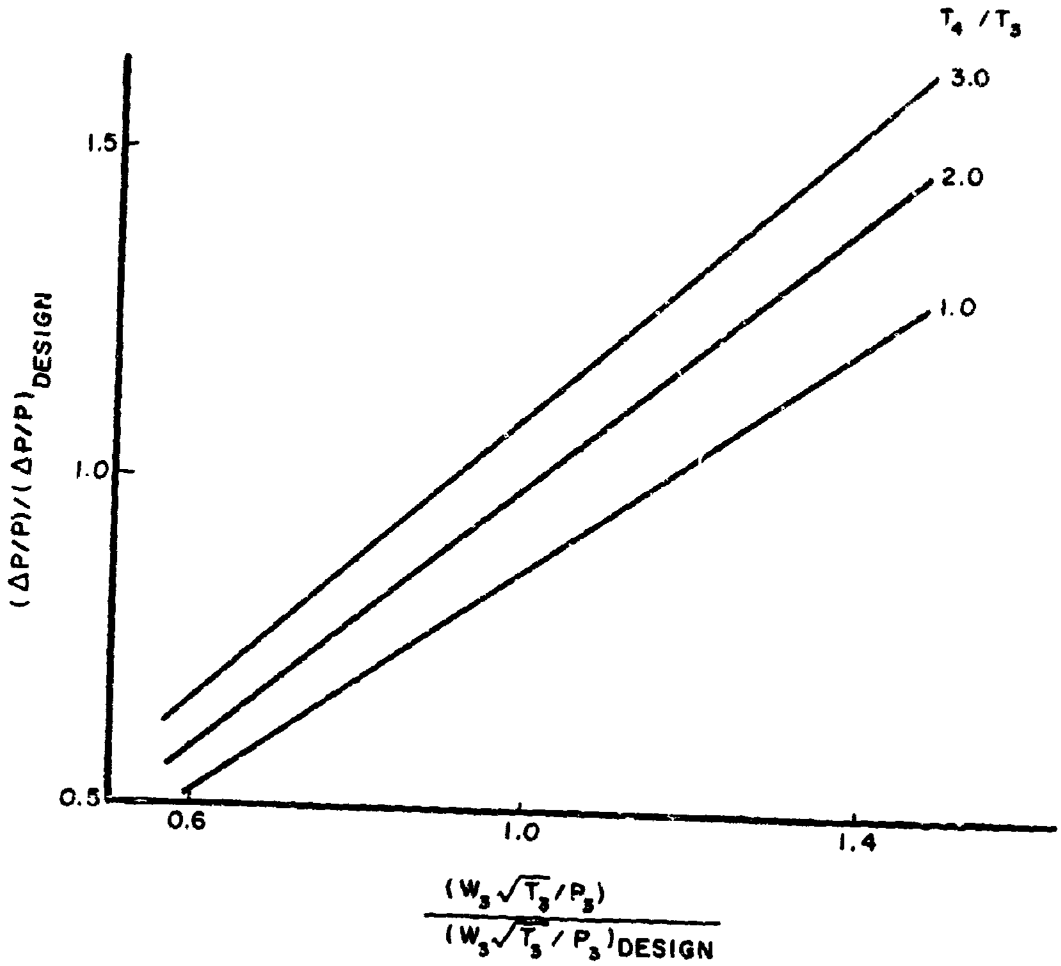


Figure 24. 12 Pressure Ratio Turbojet Combustor Characteristics

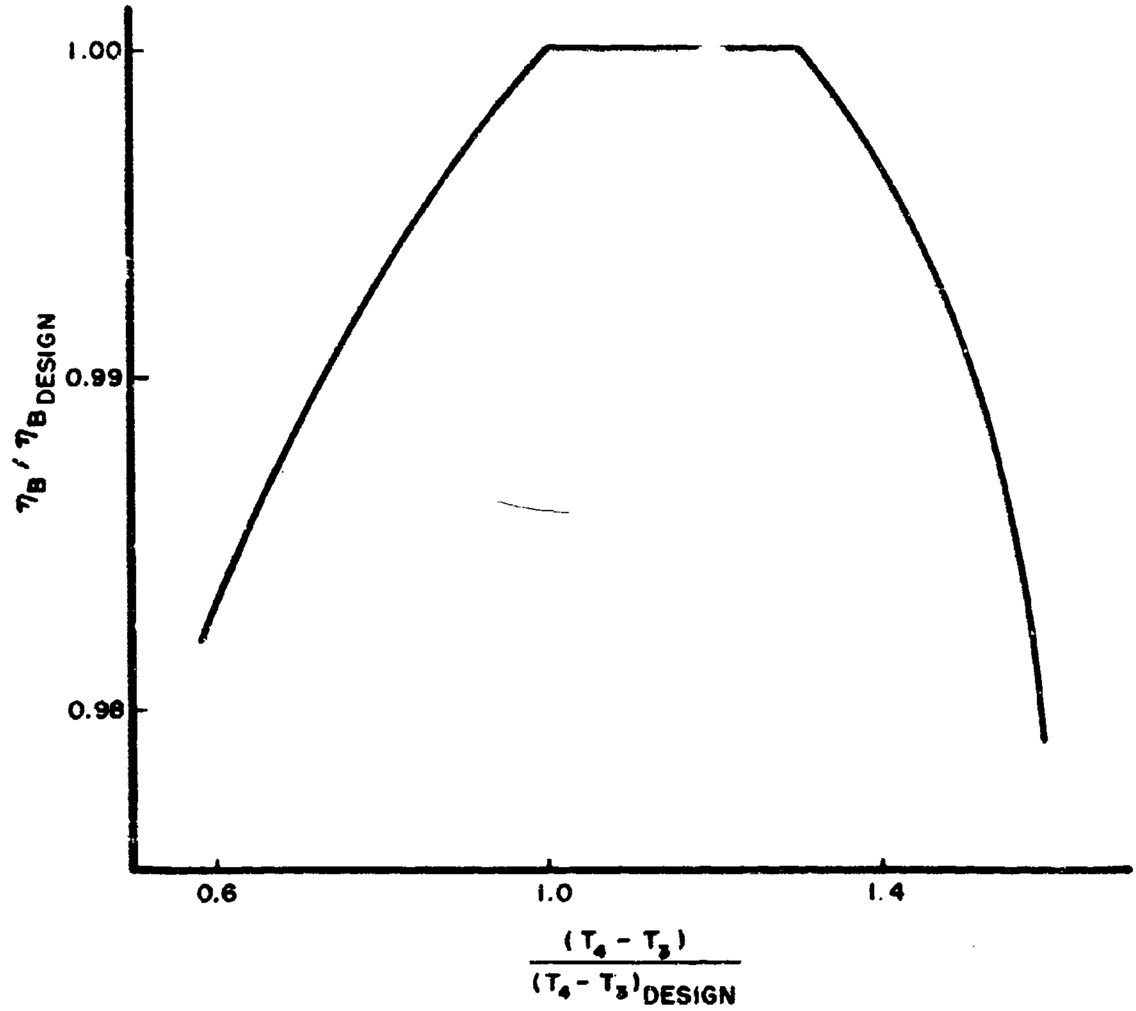


Figure 25. 12 Pressure Ratio Turbojet Combustor Characteristics

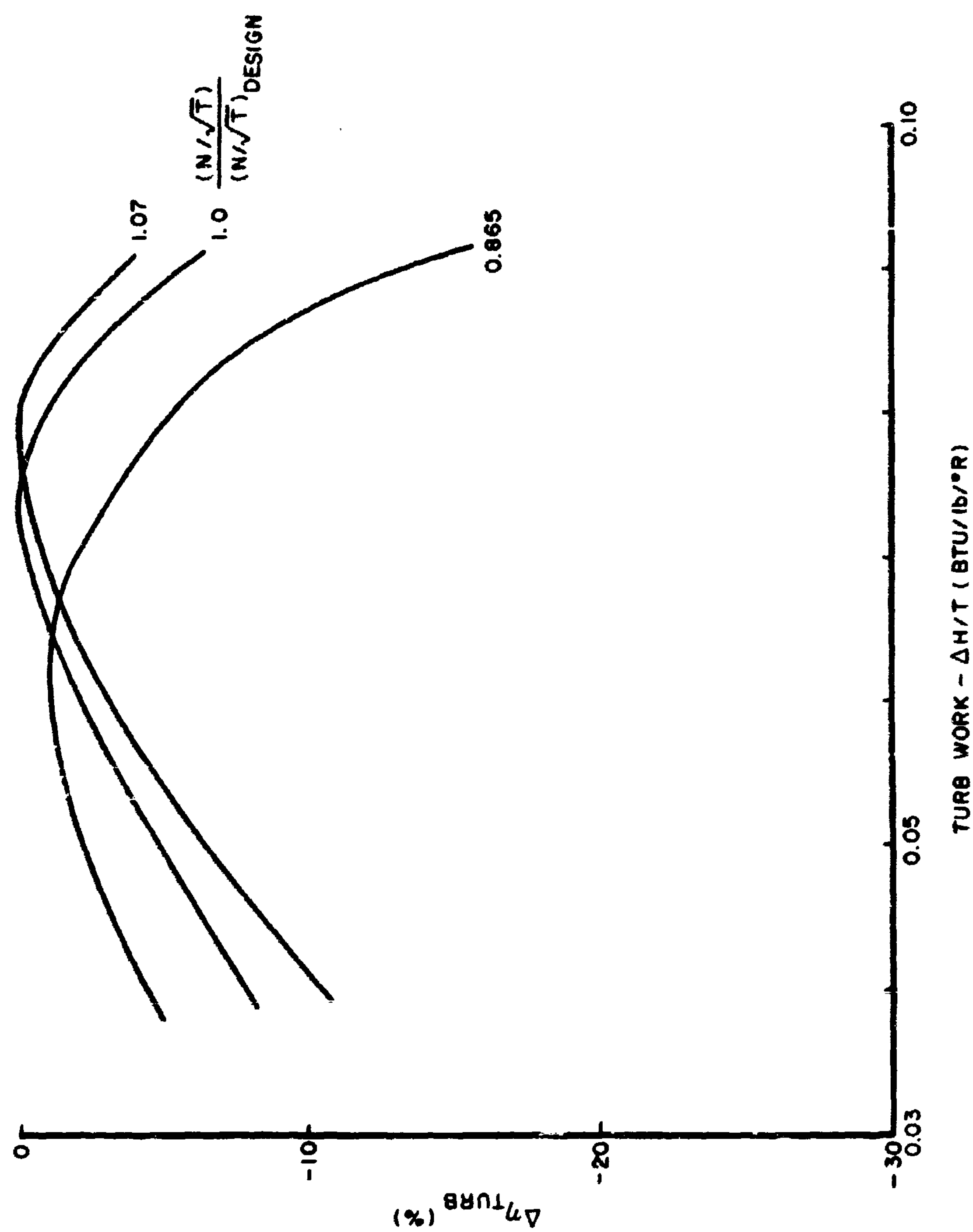


Figure 26. 12 Pressure Ratio Turbojet Turbine Characteristics

TABLE IV

INTEGRATED EFFECTS OF COMPRESSION PERFORMANCE DEGRADATION ON A PRESSURE RATIO = 12 TURBOJET

Conditions	Cycle Parameter	Base PR Performance	Degraded PR Performance
Sea level static — military rated power	$\frac{PR}{PR_{DESIGN}}$	1.0	0.91
	η_c	0.81	0.795
	T_4 - °R	2260	2260
	F_n - %	100	98.3
	SFC - %	100	103
Mach 1.9, 40,000 ft, military power	$\frac{PR}{PR_{DESIGN}}$	1.0	0.886
	η_c	0.835	0.819
	T_4 - °R	2260	2260
	F_n - %	100	100.8
	SFC - %	100	103.1
Mach 0.8, 40,000 ft, cruise power	$\frac{PR}{PR_{DESIGN}}$	1.0	0.886
	η_c	0.835	0.819
	T_4 - °R	1980	1980
	F_n - %	100	97.8
	SFC - %	100	104.2

Code: PR = pressure ratio
η_c = compressor efficiency
T_4 = turbine inlet temperature
F_n = net thrust

compressor degraded performance line when the combustor and turbine performance shifts were included. Other component performance levels, such as that of the inlet and nozzle, were held the same as in other perturbations and, furthermore, were identical. As is illustrated in Table IV, the compounded effect of the compressor degradation and the combustor and turbine performance shifts yields a more severe engine performance change than just the compressor degradation alone, as is to be expected. At sea level static, military power, the 9% decrease in operating pressure ratio gives a 1.7% decrease in thrust with a 3% increase in specific fuel consumption. At Mach 1.9, 40,000 feet, military power, the 11.4% decrease in operating pressure ratio yielded a 0.8% increase in thrust with a 3.1% increase in specific fuel consumption, while at Mach 0.8, 40,000 feet, cruise power, an 11.4% decrease in operating pressure ratio gave a 2.2% decrease in thrust and a 4.2% increase in specific fuel consumption. A graphic comparison of the performance changes due to the degrading of the operating line of the 12 PR turbojet is given in Figure 27. As is shown on the chart, two levels of performance change are presented for each engine operating condition — the effect of the compressor's degraded performance alone and then the effect of the compressor's degraded performance plus the off-design operation of the combustor and the turbine. Two things are indicated by the chart: (1) the off-design operation of the combustor and turbine is not as influential as the degradation of the compressor operating line; and (2) the subsonic cruise condition is the point where the engine is most sensitive to component degradations in general.

The above performance perturbations for the 12 PR turbojet are indicative of the effects of first an engine that had to be de-rated in the compressor section only with a redesign of the combustor and turbine section to minimize the performance penalty, and second, the engine performance penalties that can occur when, due to late timing, the decision has to be made to de-rate with no possibility of redesigning the combustor and turbine sections. As can be observed, engine performance suffers any time a compressor design must be compromised to lower operating bands that are inherently possible within the compressor.

14. DECISIONS LEADING TO PERFORMANCE COMPROMISE

When the history of engine development is surveyed with respect to systems applications, it appears that a goodly portion of the engine compressor derating has been in response to distortion levels more severe than had been anticipated. More often than not, this derating process has been attributed to a maldistribution of flow and/or pressure generated by the inlet. The corollary to this is that the engine distortion tolerance is marginal. The decision to de-rate an engine is not always dependent on the incidence of compressor surge/stall. Sometimes an engine's compressor reacts to an input distortion in a neutral or even an amplifying manner. When a compressor is neutral or amplifying to the distortion, the decision may have to be in favor of a reduction in turbine inlet temperature. The necessity for the reduction in turbine inlet temperature is generally the result of the compressor not having damped out (attenuated) the input distortion and having passed it into the combustor. This often increases the severity of the combustor's exit temperature pattern factor which can result in either a reduction in hot section life or, in severe cases, in rapid hot section failure. With cooled turbine components, the life reduction of failure can sometimes be offset by an increase in turbine cooling air quantity and placement, but this increase in coolant flow results in a performance penalty. In any case, the requirement to de-rate an engine, whether it be in response to compressor surge/stall or a worsening of the combustor exit temperature (pattern factor), will degrade engine performance and it is incumbent on all involved to insure that a derating decision is not required at any time.

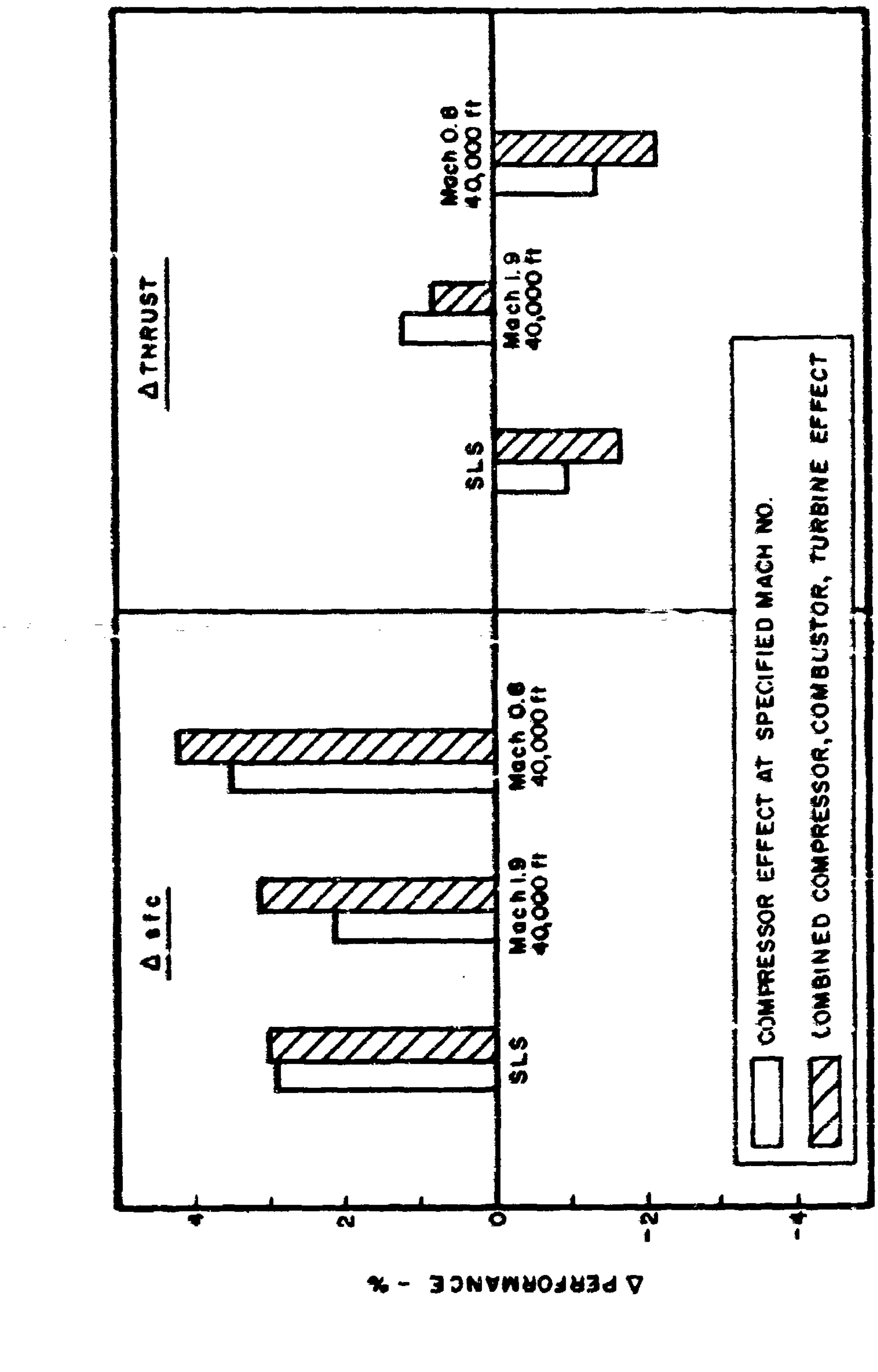


Figure 27. 12 Pressure Ratio Turbojet Performance Changes

15. PROCEDURES TO PRECLUDE DERATING

What can be accomplished to insure against a requirement to de-rate an engine? To answer this question specifically requires a detailed examination of the complete engine design. However, some general observations and historical trends can be discerned. We again consider the compressor section, since this is generally the most difficult and time-consuming component to develop for an engine. The following performance development pattern has emerged.

The time span required to develop an advanced compressor to full design performance levels is generally about 24 months. This time span is applicable to advanced development compressors and, although considered advanced, the designer has attempted to increase only one to two performance parameters at a time. Case histories are multiple wherein a simultaneous upgrading of stage loading, stage matching, and weight reduction has been attempted with results varying from marginal success to dismal failure. Unfortunately, the performance-weight improvement attempts have not always been restricted to the advanced development programs but have sometimes been attempted in an engineering development program.

The engineering development phase should be devoted to attaining durability with the performance capability having been solidly derived previously in the advanced development phase. When upgraded performance levels are attempted in an engineering development program, the impact on the system development can be heavy. The impacts are generally escalated systems cost and program slippage, with marginal weapon system capability for the end product if not full system cancellation. Figure 28 is a plot of the general compressor performance derivation with respect to time for advanced development compressors. Indications are that at least one redesign of the compressor will be required at about the 10 to 12 month time period, with the design trade being exercised in the direction of increased weight with performance targets held constant. At least three configurations are generally required — the original design, the redesign, and a minor performance adjustment design based on the results of the first two configurations.

Figure 28 is a general history of advanced compressor development that pertains only to aero/thermodynamic performance attainment and does not have induction system distortion influences factored into it. It is estimated that if input distortion testing is made an integral part of the advanced development area, as it should be, the net effect will be an extra 6 to 10 months time extension and one more design configuration. Considering the impact in terms of time and dollar expenditure if put off until the engineering development phase, the imposition of distortion testing in the advanced development area, or even back into the exploratory development area, is the only logical and prudent course to pursue.

Figure 29 illustrates the breakdown of the types of compressor-performance testing that should be accomplished in each of the three development phases currently recognized in the Air Force. The majority of distortion testing should be carried out in the exploratory and advanced development programs, prior to engineering development (system acquisition). Two key points are recognized. First, multistage performance testing is in an overlapping position between exploratory and advanced development. Depending on the magnitude of the advancement in the state-of-the-art attempted, the multistage testing could be a feasibility demonstration and hence an exploratory development program, or it may be purely an advanced development program that is a slight advance in the state-of-the-art whose feasibility has been clearly established previously. In either case, distortion testing of the multistage unit is both valuable and necessary. In the case of the advanced development area, distortion testing of the multistage unit is mandatory for the derivation of the compressor-engine performance-distortion trades to insure overall system stability with minimum performance degradation across the system's operational band. The second key point is that the engineering development program should be devoted primarily to durability testing with distortion testing devoted

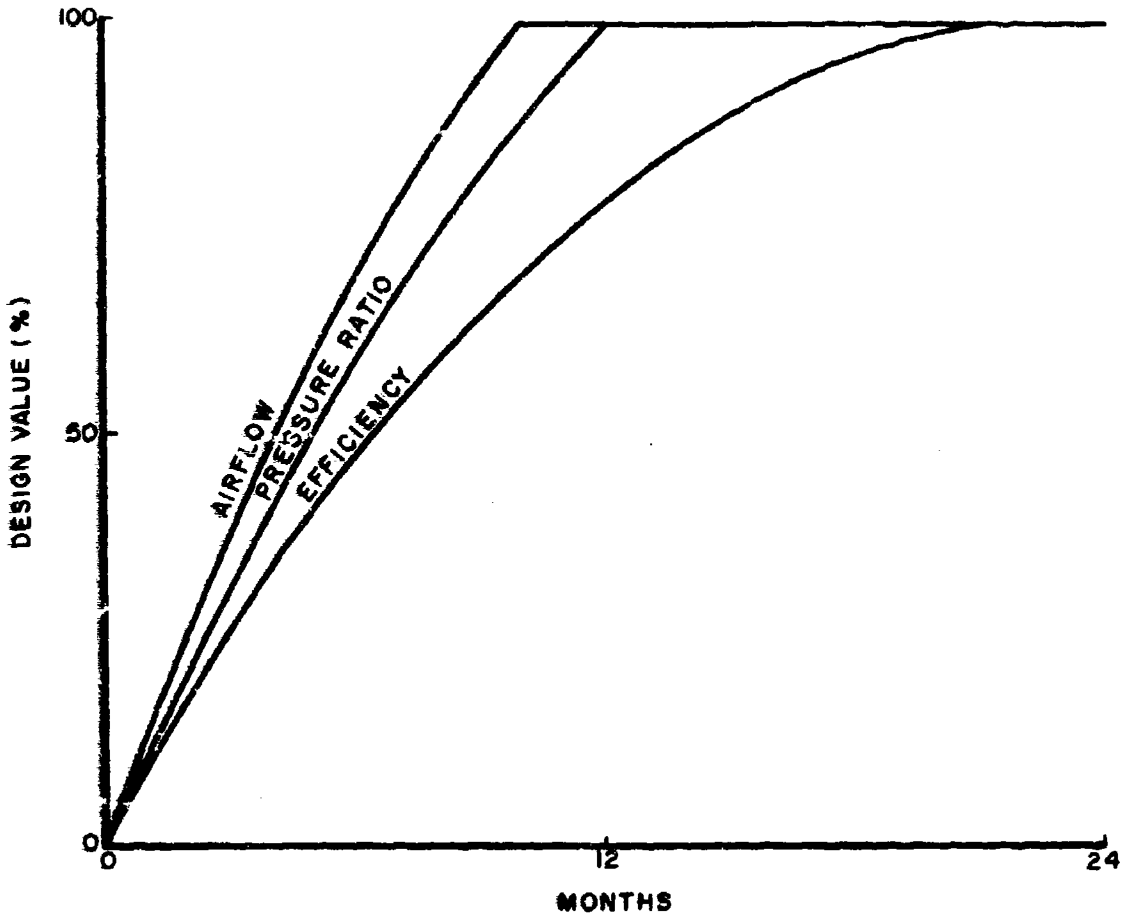


Figure 28. Advanced Compressor Development General Performance History

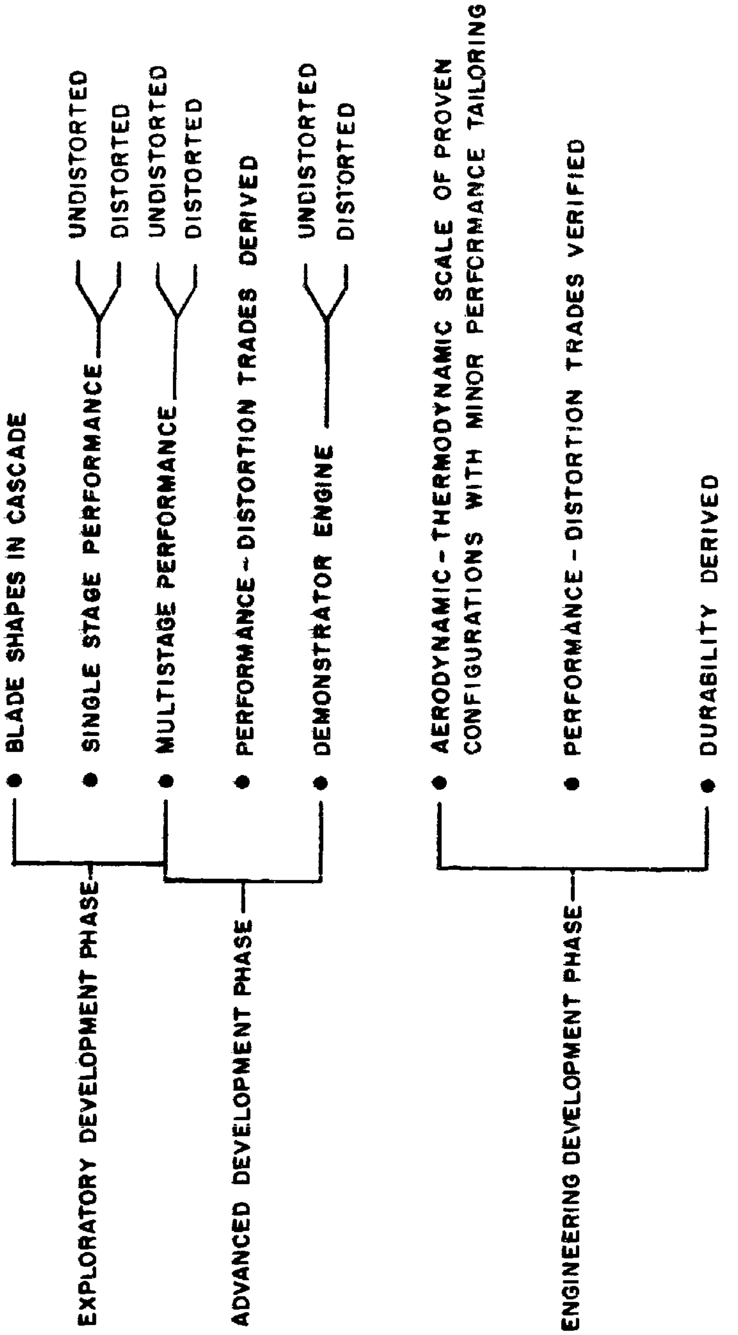


Figure 29. Compressor Testing

to verifying and exercising the performance-stability-distortion trades identified previously in the advanced development program. Due to the increased complexity and cost of systems, in terms of money, time, and manpower, deviations from logical sequencing of the time-phased testing and demonstration of performance-stability capability will exact an ever increasing toll if allowed to continue unchecked. Late identification of the performance-stability characteristics of a compressor-engine configuration has been tantamount to systems acquisition slippage and increased cost, with reduced system capability as the end product.

The performance perturbations performed and discussed within this section have been restricted to the simple turbojet case. As can be appreciated, compressor degradations would also shift the performance characteristics of an augmentor operating behind the engine in the same manner as the combustor and turbine were shifted in performance level. The absolute sensitivity of the afterburner, with respect to input changes of pressure, temperature, and flow function, has not been derived for this study. From the previous discussion on high augmentation ratio and pressure drop, it is apparent that an augmentor's performance shifts will be larger than either the combustor or the turbine. The simple turbojet illustration can also be thought of as the gas generator of a turbofan engine. Hence, the turbofan engine, for subsonic applications, is basically an augmented turbojet with excess tail-pipe energy translated through a shaft to a fan-pressurized duct; the compressor (gas generator) performance degradation will influence the low-pressure turbine (fan drive) in much the same manner as the high-pressure turbine. Since the low-pressure turbine is designed to extract energy from a lower energy level gas than the high-pressure turbine behind which it operates, the low-pressure turbine will exhibit a higher sensitivity to off-design input conditions for the same consideration outlined in the performance section for the simple turbojet. The effect will appear in the form of a net increase of performance sensitivity to compressor degradations or turbine inlet temperature degradations. When all of the above performance influence factors are applied to an augmented turbofan, the absolute determination of performance-stability trading becomes complex, to say the least, but can be completed.

SECTION V

INLET SYSTEM TEST

1. INTRODUCTION

Previous sections have outlined some of the various operational and engineering factors that have specific impacts upon inlet-engine inter-performance and the implied performance sensitivity of the overall weapon system. The discussion of the effects of the internal adjustments of the engine and resulting performance losses to accommodate a mismatched installation underlines the weapon system performance risk associated with maldefined installed performance. The discussion also emphasizes the importance of early, categorized subcomponent, component, and engine testing to minimize this risk.

Early definition of aero-thermodynamic and mechanical reactions downstream of the compressor face to various flow disturbances at the compressor face, although vital, represents only half the problem of insuring adequate definition of installed performance before the fact. The validity of programming early tests to define the reaction of the inlet exit-plane flow conditions to inlet design, position, and system operating envelope is recognized throughout industry. This recognition is tempered with various degrees of a rationale that limits the depth of early performance definition attainable due to a surmised inability to devise adequate early test programs and/or interpret test data to adequately generalize performance predictions. Recent experience with inauguration of weapon systems into the operational inventory shows an increasing penalty in performance and/or availability timing which can be directly associated with overdependance on this rationale.

The trend toward overdependence upon the weapon system flight-test phase to define and overcome installation performance shortages becomes more evident as operational requirements become more stringent, and if the trend is permitted to continue, could prevent the Air Force from achieving future system performance requirements. It becomes important, then, to discuss some of the qualitative aspects of an inlet test program that would logically sequence data acquisition throughout the development phases of specific weapon systems.

2. CURRENT LIMITATIONS

A survey has been made of recent aircraft test-flight histories to determine the extent to which installation performance and stability problems were significant factors in late engineering changes to the systems. The majority of the stability and interface performance shortages that flight test programs revealed could have been discerned or predicted via ground testing earlier in the development cycle. In general, failure to discern and correct interface stability and performance deficiencies prior to the test flight phase of development programs can be attributed to one or more of the following:

The ground test program was insufficient in scope.

There was a failure to recognize indicators of the deficiency revealed by data generated during a ground test program.

Ground test program timing did not permit revealed deficiencies to be corrected within the overall development schedule.

Two aspects are often cited, during after-the-fact analyses, as major causes of insufficient inlet test programs. One is the high cost of generating data that will accurately predict the performance levels and inter-effects of the end item across a sufficiently wide operating band. The other aspect concerns a surmized severe limitation in obtaining useful inlet performance characteristics prior to final definition of the vehicle, inlet, and propulsion system configurations.

3. SCALE EFFECTS

The test program cost involves the use of subscale test equipment and the definition of data validity as a function of test article size. Subscale models for steady-state data acquisition have been used throughout the industry; results have been directly proportional to the engineering talent applied to the various programs. Sizes down to the range of 1/20 linear scale have yielded valid and useful data. Available information concerning past programs indicates that scale sizes to less than 10% are effective for steady-state stability definition. Boundary layer thermodynamic and displacement simulation requires increasing emphasis as the Mach number range of inlet designs is expanded, and adequate simulation to scale sizes of 10% will require additional test facility requirements, particularly in the areas of Mach number and total pressure range. An indication of the degree of success in steady-state stability simulation is evident in Figure 30. The test program from which this data was extracted was recently completed for a high-performance subsonic system. The contractor recognized the potential jeopardy to system performance and availability timing that could occur if early stability definitions were not obtained; therefore both 0.2 scale static, and 0.1 scale wind tunnel programs were accomplished. These programs defined the performance inter-effects of throat area, lip geometry, contraction ratio, recovery, and distortion level for various airspeeds, angle of yaw, and angle of attack. Changes to the final inlet design were made as a routine part of the development program, using the test program results. Subsequent flight test data has yielded data which falls within the envelope of the subscale results. Similar examples are evident for steady-state performance prediction for supersonic applications. In these cases, emphasis upon both engineering techniques and program timing permitted performance detriments to be identified and design changes to occur as a normal and planned part of the system development program. Examples are also available of weapon systems with major installation performance losses becoming evident in the test flight phase, and weak, ill-defined inlet scale model test programs evident in the early development phases.

The requirement for sound inlet test programs, which identify time-phased exit plane flow characteristics as a function of transients applied to or generated by the inlet, has been given significantly increased emphasis recently. Here again, test-facility limitations and program costs dictate the use of subscale testing procedures. The advent of the requirement for precise definition of inlet system performance and transient response characteristics over the entire flight regime has underlined the importance of defining the effects of boundary layer in achieving conditions of similarity. The problem of defining and achieving similarity with test facility limitations increases in complexity as the scale is decreased as also happens with steady-state testing. The transient test function overlays the boundary layer simulation and viscous effects problem with the requirement for frequency simulation, which is a major cause of the significantly higher level of engineering emphasis required for a pertinent subscale transient test program. Correlation of flight with large scale ground data has recently revealed that engine behavior with cyclic distortion is a direct function of the maximum pressure amplitude up to an impressed frequency at least equivalent to the engine RPM. This being the case, inlet exit plane flow behavior with externally or internally impressed transients must be defined to the frequency range of 150 CPS for engines of the 300 to 400 lb/sec nominal airflow class. If we assume geometric similarity and that Mach number and free stream velocity are maintained on the test model, the required test frequency for frequency similarity must be inversely proportional to the test model scale to define the response characteristics of the full-size inlet. Adequate definition of cyclic distortion patterns

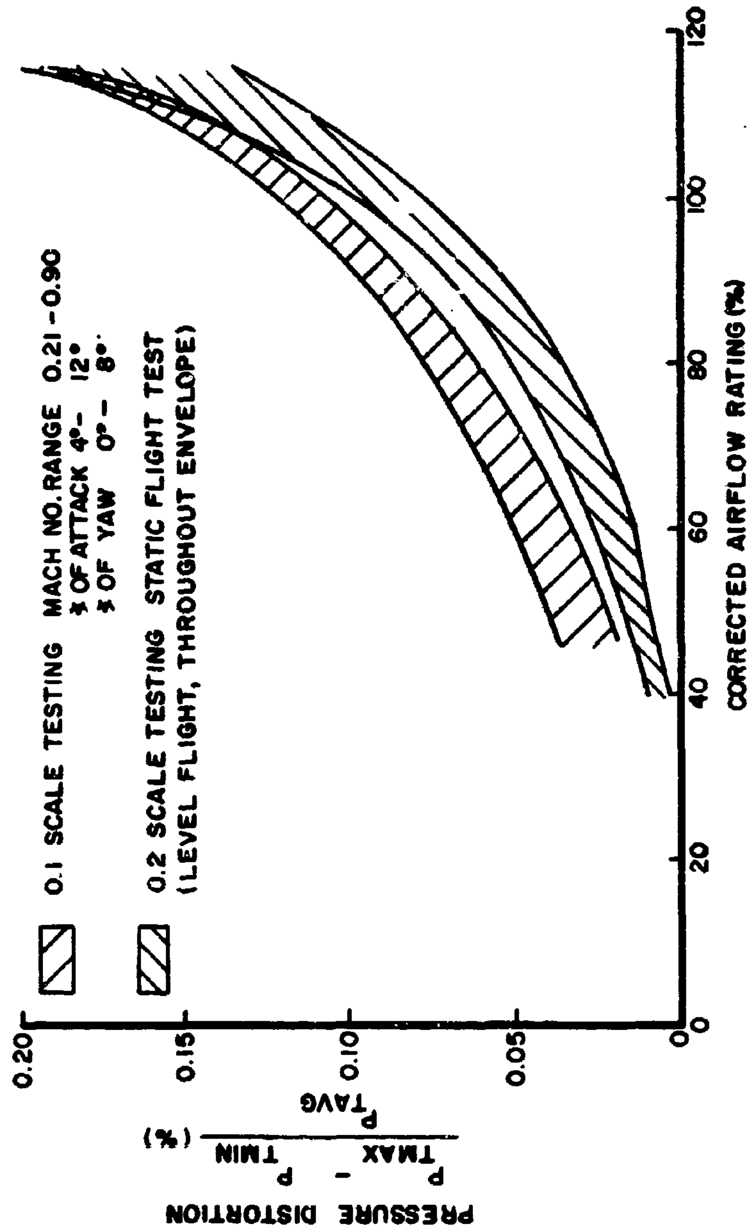


Figure 30. Inlet Stability Testing Model and Full Scale Correlation

whose frequency may be a function of inlet dimensions also will require instrumentation capable of recording data to a frequency range inversely proportional to the scale size. For example, maintenance of frequency similarity on a 1/10 scale inlet model can require test frequencies 10 times greater than the basic level. Current and future requirements for definition of response and cyclic distortion-generating characteristics, therefore, require multiple mapline and readout instrumentation accurate to at least 1500 CPS. Test model sizing is limited, therefore, by two major phenomena — the physical size limits for effective boundary layer simulation and the trade between scale size and instrumentation response rate. These limits are ill-defined and require much additional engineering emphasis to reach a technically justifiable compromise between data acquisition and program cost.

4. TIMING

In addition to the cost aspects of inlet test programs, which have in the past limited the extent of early definition of installed propulsion system stability and performance, the aspect of test program time-phasing has also limited definition. Two basic objectives need to be achieved by inlet performance test programs. First, the various and inter-related effects of the many inlet performance variables mentioned earlier need to be reexamined with the objective of more closely defining their distortion-producing aspects. Data must become available that will permit each of the inlet design variables to be evaluated for its potential effect on producing various levels of distortion during operation of the weapon system. These potential distortion-producing effects must then become a part of the primary design criteria from the outset of a particular inlet system design. Second, the inlet test program must yield the capability to discern, during contract definition, inlet system configurations that can meet specified performance levels, including predicted maximum levels of distortion, both steady-state and cyclic, throughout the operating range. A vital step in minimizing weapon system delays and performance degradation is adequate technical assessment, by the Government during the preacquisition phase activities, of the potential distortion-producing tendencies of the planned inlet system.

The timing associated with achieving inlet system performance and stability definitions through initial configuration-oriented tests prior to the start of the System Acquisition Phase is considered to be about 3 years for a new system design. To the extent that an emerging weapon system design can be based upon test data acquired under previous programs will be the extent to which this lead time can be decreased. In general, the required lead time prior to the start of the acquisition phase will vary from 1 year for minor redesigns for known systems to 3 years for system designs employing new and revised concepts. Failure to maintain this time phasing of test data acquisition will result in scrambling weight, thrust, specific fuel consumption, and stability objectives during the acquisition program with attendant system performance degradations and lag in availability.

As has been mentioned frequently, a very important part of this inlet system test program is early definition of the transient effects on exit-plane flow due to presence, position, and actuation of various upstream components and due to changes in yaw and angle of attack. Companies are finding it necessary to determine instantaneous inlet exit plane velocity and pressure distortion at rates up to 2,000 cuts per second. Instrumentation that is sensitive and responsive enough to meet such data requirements tends to be sensitive and responsive enough to pick up every vibration and stray electric field. Nearly simultaneous variation of numerous parameters obscure that which is cause and that which is effect. These types of data-interpretation difficulties make the use of an electronic dynamic simulation system in conjunction with the test program imperative. Furthermore, extrapolating spot-point test data to defined and continuous performance throughout specified envelopes implies the coexistence of a mathematical dynamic model. Effective data interpretation, then, requires that transient testing and mathematical dynamic model buildup proceed in parallel from the outset.

The major emphasis during the early (preconfiguration) phases of an inlet test program should lie in defining the steady-state and dynamic performance intereffects of the parameters associated with establishing the basic configuration. Inlet placement criteria should be established, with such items as wing-body interference effects delineated to insure that unforecast performance detriments are kept to a minimum and to minimize unknown performance changes due to inlet-vehicle placement alterations that might occur later in the program. These early indications of steady-state and cyclic distortion patterns that the final configuration will present to the compressor face are required by the engine companies to define compressor stage matching characteristics that will offer the greatest performance potential for a defined stability margin. For instance, the loadings to which the first compressor stages are matched should be a function of the expected distortion levels; lack of adequately interpreted "pre-configuration freeze" inlet data has prevented this from occurring in the past.

Later in the inlet test program, as the end of Contract Definition Phase approaches with its attendant system configuration freeze, the performance definitions and trades of the many variables, such as the presence of a boundary layer bleed system, bypass system, and control system transients, should become more firmly established. Model testing to determine the effect of geometry changes, alternate bleed schedules, control system dynamics, angle of attack, etc., on the general distortion levels will produce data required for preliminary integrated system stability definition. The final propulsion system component stack up should be dependent upon the characteristics of the compressor face flow having been defined by the inlet test program in conjunction with the dynamic model data that is being generated. It is only through such data availability during the contract definition phase that the engine stability margin as installed can be predicted with an acceptable degree of confidence.

With the basic engine, vehicle, and installation configuration established at contract definition, the concurrent buildup of propulsion and inlet system test data and dynamic simulation procedures has established system performance and stability criteria that are defined within the various documents necessary to the start of System Acquisition. The major programs in this phase will be large-scale inlet-engine compatibility testing using engine and inlet hardware in conjunction with flight-worthy control system components to refine the overall performance levels and engineer the minor design adjustments that will become necessary. Refinement of the mathematical dynamic model with test-defined control system, propulsion system, and inlet system performance transients, along with defects defined and corrected by the integrated installation test-dynamic model program will yield a test-flight configuration that should become the production vehicle with only minor modifications required.

5. SUMMARY

An inlet test program organized to begin the definition of necessary stability data should be established approximately 3 years in advance of weapon system configuration definition. This is vital to the inclusion of high-performance weapon systems, with the USAF forecasted mission capability requirements, in the inventory at the projected availability date. Lack of an adequate test program from the outset will mean a failure to accurately predict weapon system performance and availability, and there will be an attendant risk of a decision not to proceed with system acquisition.

SECTION VI

INSTRUMENTATION

1. INTRODUCTION

The derivation of propulsion system performance and stability requires an intensive effort in both data acquisition and data management. Generally, any engineering product passes through the following phases: (1) the conceptual phase; (2) the prototype or model phase; (3) the development phase; and (4) the production phase. Air Force system acquisition procedure, with respect to propulsion systems, treats these phases as (1) the experimental development phase, (2) the advanced development phase, (3) the engineering development phase, and (4) the production of phasing-into-the-operational-inventory phase. The data requirements for these separate phases of the acquisition procedure are different and, as will be discussed, can conflict with other requirements. This section will concern itself primarily with the propulsion system, which is considered to be the inlet, the engine, and the nozzle. For easy discussion, the exploratory and advanced development phases will be consolidated with respect to data and instrumentation requirements. There is little question as to the parameters that require recording for the determination of performance and stability. The ability to have constructed inlets, engines, nozzles, and aircraft in the past and to have been able to numerically define the systems in the first place, was based on the compilation of parameters. The basic questions then are when to record various parameters, with what accuracy should they be recorded, and what is the data generator configuration.

2. EXPLORATORY AND ADVANCED DEVELOPMENT PHASE

As pointed out in the previous section on inlet testing (Section V), the utilization of scale models requires instrumentation with escalated frequency response when contrasted to the full-size inlet instrumentation. This instrumentation frequency multiplication factor was necessary to assure just a similarity correlation from size to size. The instrumentation frequency requirement for the generation of stability-performance data under the influence of cyclic input could impose another multiplication factor. These multiplication factors imply that the test instrumentation frequency required is

$$f_{required} = f_{basic} \ (K_{scale}) \ (K_{dynamic})$$

where f_{basic} is the frequency associated with the full-size inlet, K_{scale} is the influence factor associated with sizing for similarity, and $K_{dynamic}$ is the influence factor associated with the imposition of nonsteady-state input to the system.

Section II discussed the basic frequency of instrumentation associated with compressor testing and showed this to be functionally related to the dwell time of blading within a particular defect area. This is implied also in Section V and is functionally related to the fundamental blade passing frequency. This fundamental frequency band for steady-state instrumentation requirements for compressor-engine testing will have the same type of multiplication factors as those for the inlet when the compressor-engine is not of the exact size or configuration as the anticipated final product. The instrumentation requirements for scaled or demonstrator articles then are seen to be functionally related to and, in most instances, more stringent than the instrumentation of the final product. These test articles, which are tested and utilized as part of the exploratory and advanced development phases, are necessary, and it is not possible to bypass them in the classical sense. The data derived in these phases cannot be replaced by engineering development data with any assurance that the best balance between performance, stability, weight, and cost has been struck.

3. ENGINEERING DEVELOPMENT

The instrumentation requirements for the engineering development phase generally will have been relieved of the multiplication factor associated with scale model testing, at least as far as the basic engine is concerned. This phase, as brought out in Section IV, is generally devoted to durability achievement with performance-stability goals having been essentially completed in the advanced development phase. The engineering development phase has another very important constraint imposed near its conclusion, however. This constraint is the flight test program. The flight test program is conducted to verify performance, stability, and durability of the weapon system as a whole, not to develop them. Due to the many parameters that must be recorded for this verification, across the flight spectrum of the complete weapon system, the development of stability and performance during the flight test phase is all but impossible. The limitations of volume, weight, and placement of the data recording system are prohibitive to the simultaneous recording of a complete set of both aircraft and engine parameters, therefore, if stability and performance have to be developed in the engineering development flight test phase, the program will have to be concentrated on one or the other. For instance, inlet-engine distortion sensitivity testing will have to be pursued independently of say, maximum range demonstration or minimum time to climb, etc.

4. SUMMARY

The frequency response of instrumentation applicable to steady-state stability testing of inlet models is anticipated to be in the neighborhood of at least 2000 CPS. This frequency is further anticipated to require a frequency multiplication factor of about 2 to 3 to assure fidelity of results under cyclic input to these scale model inlets. This cyclic input multiplication factor would then require instrumentation response capability to 6000 CPS. This instrumentation response rate capability should be sufficient to cover the range of stability testing, including dynamic input, for turbine engines anticipated for current and near-future Air Force applications. Instrumentation capable of a 6000-CPS response rate exists within industry. The problem now is to assure that recording sizes and weights do not forestall data availability by leading to assumptions that the data can be derived during a later test phase, for instance, during the flight test program. During this phase, volume and weight are at a premium, and the cost of data acquisition is sometimes prohibitive. Furthermore, certain data acquisition can become virtually impossible to obtain within this phase.

SECTION VII

CONTROL SYSTEM DETERMINATIONS

1. INTRODUCTION

The adaptation of an engine in an air vehicle designed to achieve specific operational objectives involves the detailed task of establishing all of the performance penalties that may result due to compromises and trades. One of the more difficult problems, from the standpoint of lead time scheduling, is the detailed definition of the compressor pumping characteristics along with inlet flow stability characteristics that will be required for the air vehicle to meet its performance objectives. Because of the importance of inlet-engine stability margin to overall system performance, distortion-acceptance capabilities of the engine must be constantly improved, while the distortion-generating characteristics of the inlet must be minimized. Unfortunately the most direct engineering solution to both of these problems generally results in reduced propulsion system performance.

2. CURRENT CONTROLLING PROCEDURE

The reasoning leading to this conclusion can be illustrated best by examination of typical system development procedures. Initial schedules are calculated for a particular engine design and are subsequently readjusted after component performance has been verified and during engine development test. Engine steady-state operation is controlled by a closed looped control of RPM, while transient performance is controlled by refinements of the appropriate schedules and compensation to meet the required engine response. For rapid engine response, the traditional approach has been to set a large enough margin between the operating line and the steady-state stall limit line. This method wastes engine performance potential, since a large surge margin applied to an axial flow machine means that the engine is matched to a lower-than-available pressure ratio and sometimes a lower-than-available compressor efficiency.

If, during flight test development, it becomes apparent that a significant level of inlet flow distortion exists, then depending upon the magnitude of the distortion, several possibilities exist. The immediate, short-term fixes to allow continuation of the development program will include confining or restricting certain flight conditions where distortion becomes most critical, rematching the engine to provide a larger nominal surge margin, or readjusting schedules. The long-term solution, which can be time-consuming and expensive, is sought by modifying the inlet duct to improve its flow characteristics (which may be at the expense of average pressure recovery) and attempting to improve the distortion tolerance of the engine itself. The short-term fixes, though unsatisfactory, do provide a method of minimizing the delay in achieving some operational capability of a new weapons system. The long-term fix, if not completely successful, will result in some penalty to the overall performance of the weapons system and this penalty will have to be accepted throughout the system's life.

Translating the preceding experience to the advanced systems now under consideration for development leads to a further bleak conclusion. Environmental flight conditions, inability of the inlet to provide uniform flow characteristics, manufacturing tolerances, downstream component operation, etc. already require a turbine engine compressor to be matched for operation at other than the maximum attainable pressure ratio. Present practice is to establish an empirically predetermined steady-state stall margin based on the estimated inlet pressure, temperature, and velocity distortion conditions that might be encountered due to flight maneuvers during rotation at takeoff and moderate angles of pitch and yaw. In the past, these practices have been, at best, intermittently effective in the establishment of a stable area of operation for the propulsion system. This stable area does not generally cover the complete region of the flight envelope desired for the weapon system. As a result, the propulsion system may exhibit a high incidence of stalls or compressor surges when operating during particular maneuvers in specific sections of the flight envelope. Currently, excessive propulsion instabilities per flight while in these special operating zones have created problems in flight test scheduling along with delays in system development, while long lead time improvements in engine loading and/or inlet geometry are pursued. For advanced weapon systems, where the emphasis has been placed on multimission and/or maneuver capability, the number of engine stalls per flight maneuver must be eliminated. The most direct approach to improve the engine-stall to flight-maneuver ratio is to lower the compressor steady-state operating line. This increases the nominal compressor stall margin and generally allows an engine to accept increased levels of nonuniform inlet flow, however, thrust and/or SFC penalties result.

The reduction of the engine-stall to flight-maneuver ratio is not unique to planned advanced weapon systems. This ratio reduction has been a consideration throughout the design, development, and flight test of every turbine engine powered weapon system. The methods presently used to reduce the engine-stall to flight-maneuver ratio are not expected to be adequate for the planned advanced weapon systems. Current practice is to establish a compromised steady-state operating line for the engine throughout the flight envelope. This compromised operating line may or may not be adequate to achieve an acceptable engine-stall to flight-maneuver ratio, and severely reduced performance in terms of thrust and/or specific fuel consumption is likely. Since the attainment of any mission is directly dependent on thrust and fuel consumption, additional alternative approaches to inlet distortion-engine compatibility problems are required.

3. FUTURE CONTROL CONSIDERATIONS

The choice of alternatives involves many systematic evaluations of the current capability in several fields of endeavor in the propulsion area. An assessment of these fields of effort is essential to the choice of a particular approach to the problems of propulsion stability.

Whatever the particular problem area and the corrective measures during development, the conventional hydromechanical fuel control system is invariably developed so that its characteristics will provide the required degree of engine response and speed-holding accuracy. Because of control component tolerances and the limitations associated with the dynamics of certain components, an appreciable allowance in excess compressor pressure ratio is debited to the control system. Further allowances are made for engine-to-engine variations and individual engine degradation with time.

In terms of how the potential of propulsion system performance is compromised to provide satisfactory weapon system operational capability and flexibility, two major factors are seen to predominate. These are control system capability and inlet-engine compatibility. Regarding engine control system capability, some further discussion is warranted. In other instances, where the particular control function requirement can be identified, it is not always possible to isolate the variables. An example of this is the difficulty associated with identifying a precise signature of incipient compressor stall. Various possibilities have been explored, including time decay of pressure and acoustical phenomena, but to date, none have proved to be true indicators. The current fundamental research into a three-dimensional stall and separation theory (conducted by Cornell Aeronautical Laboratory) is expected to ultimately identify some parameter as a precise signature of imminent stall and/or surge of an axial flow compressor. After a signature of incipient instability has been identified, the next phase will be to sense this signature with sufficient accuracy and response. If such sensing is subsequently proven feasible, and here the burden is borne jointly by the engine and the control manufacturing industry, then the concept of applying closed loop control to maintain compressor flow stability under all operating conditions becomes especially attractive.

This capability would significantly increase potential performance, and if ultimately successful, will allow improved operation to be realized, even under distorted inlet conditions. It is also desired that integration of the inlet and engine controls becomes possible to avoid adverse interactions of the aerodynamic loop which exists between the engine and the inlet.

While the achievement of these ideal situations cannot be projected in terms of either total development time or cost, it is identified as the desired requirement, and, as such, defines the general lines of approach to an orderly and logical development program.

In the meantime, other techniques for engine stability control appear to offer earlier promise. These techniques involve in-flight propulsion system rematching to particular environmental conditions and could be made applicable in the interim time period until complete closed loop control is demonstrated as feasible. Many current propulsion systems feature variable geometry in the compressor and exhaust nozzle. Current projected systems further extend the variable geometry feature to the turbine and the bypass duct. In such future engines therefore, it is feasible to shift the operating line at the same time that fuel supply is increased in response to an acceleration demand. By matching the acceleration fuel schedules and corresponding downstream flow area variation, transient conditions can be accommodated while operating with a reduced nominal steady-state surge margin. To some extent, this technique is an extension of present practice in which compressor variable stators and/or interstage bleeds are actuated to perform essentially similar functions. Of course it is necessary that the effects of inlet distortion on compressor and engine operation have been predetermined and are scheduled into the control. Where distortion effects change with flow pattern over the flight spectrum, the magnitude of the effect can hardly be fully predetermined for each and every flight condition, even by extensive wind tunnel testing. However, the relative effect may not be widely different and prescheduling for a distorted stall limit line could provide suitable engine operation. In extreme cases, for example, where engine inlet maldistribution is magnified by flight at high angles of attack, some sensed signal to trim the engine would be necessary to maintain overall stability. Again, in this particular example, an acceptable compromise solution might be found in transmitting angle-of-attack signal to automatically trim propulsion system operation to some stable condition corresponding to the magnitude of the input signal. Such techniques are feasible and are worthy of further investigation.

4. DESCRIPTION OF AN ADVANCED CONTROLLING PARAMETER

The evaluation which follows is concerned with conditions causing an increase in inlet flow distortion and the maximum utilization of available engine performance. Consideration of the stall/surge phenomena in turbine engines and the internal flow dynamics of inlets is of utmost importance. Stall and surge occur when the incipient unstable flow conditions introduced into an engine are no longer damped out by the operating characteristics of the configuration. The blading is triggered into stall by mechanisms not yet fully understood. The extent, type, and seriousness of the stall triggered by incipient unstable flow conditions determine the degree of performance degradation.

Compressor and fan stall and surge reaction can occur in less than one complete engine revolution, depending upon individual and combined stage performance characteristics. Any device that can sense a condition that will produce surge/stall must respond in less time than the stall or surge reaction time. Present actuation systems for variable geometry require about one half second from sense to initial actuation. An additional 0.1 to 0.25 second is required for the movement of variable geometry to influence the turbine engine cycle. This time lag is too great to be effective in preparing the turbine engine to accept adverse inlet distortion patterns. Although an inlet distortion sensor is desired, the time lag in present circuitry will generally preclude this type of sensor from being utilized, making necessary a development of an inlet distortion anticipator.

A brief discussion of the implications of various timing factors in defining a distortion protective subsystem can amplify the requirement for careful consideration of the elements to be sensed. Consider the following simplified transfer function for a very large supersonic inlet

$$\frac{\text{Shock Position (in.)}}{\text{Bypass Area (in.}^2\text{)}} = \frac{K \cdot e^{-t_d s}}{1 + ts}$$

The preceding transfer function describes the response of the shock position to changes in the bypass area. The response consists of a simple lag and a first order time delay. In this case, typical values for the time constants of the time delay (t_d) and the first order lag terms are approximately 0.03 second. Similarly, the description of the time response of a compressor to a flow anomaly is important. This time response is complex, however; it is known that a compressor can experience its own flow instability within 0.01 to 0.05 second after the development of undesirable flow irregularities at the compressor face. If the above transfer function is at all indicative of the dynamics of a flow anomaly, then it is evident that little time is left to correct the situation before compressor instabilities result. Also, when the dynamic response of actuators for bleed doors, valves, and IGV's is considered, it is evident that a sensing system capable of sensing an actual flow anomaly must be extremely fast to allow time for the engine to be prepared to receive the undesirable flow condition without experiencing an instability of its own. This response requirement is relaxed if a sensing system can discern conditions which exist immediately prior to undesirable inlet flow conditions. Further relaxation of dynamic requirements is obtained by use of a system that anticipates inlet anomalies by sensing gross aircraft flight parameters which are indicative of conditions known to be conducive to inlet problems. The extreme relaxation is the impractical case in which a separate pilot command would prepare the propulsion system for a critical flight condition. Thus, the closer the sensing system comes to sensing the heart of the problem, the greater are the demands for dynamic response of the individual component. Overall system effectiveness will depend upon aircraft and inlet aerodynamics, the dynamics of sensing, transmission, actuation, and the dynamic response of the engine itself. A comprehensive effort is necessary to ensure careful consideration of the dynamic coupling and tradeoffs regarding the sophistication of the sensors, response, and actuation of engine variable geometry.

Exploratory investigations, accomplished by engine aircraft and control manufacturers, complemented by independent research studies, indicate that successful inlet distortion protection subsystems, associated control circuitry, and actuation mechanisms can be constructed and flight tested. Such subsystems will allow for an additional margin of engine stability during flight maneuvers that induce a higher than normal inlet distortion. An engineering approach to an adaptive flow distortion protective subsystem involves the determination of an effective numerical ranking and consequence of incipient engine stall during various portions of the flight envelope. Defining such a series of "engine-stall to flight-maneuver ratios" for several current mission envelopes can reveal maneuvers that are critical from the standpoint of stable engine performance, and can, in turn, define a distortion protection subsystem applicable over a significant range of critical maneuvers.

SECTION VIII

DYNAMIC MODELING

The other sections of this report have dealt with many, if not all, of the causes and effects of engine-airframe stability problems. All of the components affecting, or affected by, transient parameters have been treated as entities and to some extent some of the near-vicinity cross-component influence effects have been discussed. The general area left, then, is what methods are available for examining propulsion system response. No attempt will be made to assess a total system here except for purposes of illustration; rather, the methods available, and to some extent being employed, will be examined.

For 10 to 15 years, turbine engine steady-state performance has been calculated by mechanized methods. Many techniques are available for this type of calculation, but most of them revolve around a trial-and-error method of guessing front end conditions, looking up actual component characteristics, and solving the classical aero/thermodynamic equations until a point is reached where an input and an output are known which can be used to develop an error. This error is then used to reset an initial guess, and so it goes until all balances are reached with acceptable error. Some attempts have been made to use this type of model for calculation of the transient performance of the engine, but with only minimal success. This approach is limited to initial estimates of acceleration time, etc. and has no real capability for examining time variant component interactions.

Along with the steady-state digital cycle calculations, analog methods have long been used for examining the transient characteristics of engines. This technique appears to have stemmed from the controls groups who have used analog computational techniques for some time in checkout and development of control systems. In reality, this method is probably the most accurate and least time-consuming way of evaluating a given system; however, it is not without its drawbacks. Unless one has available an extremely large analog computer, many approximations are required to simulate the entire engine plus control to say nothing of the inlet and secondary flow systems. Additionally, once a model of this sort is set up and wired, it behooves the user to go on line for long periods of time and investigate as many aspects of the model as possible before the model is torn down, and the computer is given to the next user. In this respect, the normal analog computer is a dedicated machine, and the translation of one model to the next is cumbersome and time-consuming. Finally, the factor of most significance is that the computer model under investigation cannot be translated from one contractor facility to another or to government facilities without almost insurmountable problems. This one factor almost eliminates the pure analog approach from consideration when one observes that, for a normal weapons system, as many as four or more contractors and the Government may be influenced by the transient performance of an element developed by some source other than themselves. As has been pointed out in other sections of this report, the airframe vendor, the engine vendor, the engine control vendor, and the inlet control vendor are all influenced to some extent by the inputs of all the others. It therefore becomes mandatory that some method of translating dynamic models be developed much like the methods used for static performance models.

The preceding comments lead to the subject of digital engine transient simulation models. The primary reason for the Air Force emphasis on digital models is the ease of translation from one computer center to another. Previous sections of this report have emphasized the absolute necessity for all vendors involved in a weapons system development to communicate with each other and with the Government. Since the primary subject of this report is stability and since stability is a dynamic problem, it behooves each party involved to communicate by the best means possible to define the absolute stability characteristics of the overall weapons system concerned. It is in the area of cross-contractor communication that the pure digital model becomes extremely useful, even at the expense of running time. There is no question that digital models sacrifice running time when compared to analog simulations, and also they divorce the user from the simulation. In the analog case, the user can actually intervene in the simulation and can respond to model outputs; this is very seldom the case in the pure digital mode. However, even with these limitations, the digital model has a very definite place in the overall system stability definition.

Accuracy has not been discussed, but it appears, that the digital model can be just as accurate as the analog model and in some areas more accurate. Figure 31 shows a problem which was run initially on an analog system, then on a digital. The response of the analog run is shown as a solid line with the circled dots being the digital response. In the time domain between 3 and 4, the analog plot became very difficult to read since the problem was run with several amplification factors, and the difference between the analog and digital points is attributed to curve reading accuracy, not to computing accuracy. The analog computer ran this problem in real time and the digital simulation was three times real time. Obviously, the time relationship is not good, but considering setup time, accuracy, and translatability of the model, this appears to be a very small penalty to pay.

Having now established the case for a digital model which can be translated, the next point to be discussed is the time-phasing of the models into the overall system and the general requirements of each model as it is introduced into the system structure. For purposes of discussion, the overall system program is herein considered to include not only acquisition, but also exploratory and advanced development phases. This then implies that dynamic modeling does not start with contract definition and acquisition, but rather must be built up over a time span preceding this phase of the overall system program.

Assuming then that each system must go through exploratory and advanced development, contract definition, and system acquisition phases, the problem becomes one of responsibility within each phase. Throughout the remainder of this section a turbine engine will be used for illustrative purposes, however, the general philosophical approach can be applied to any subcomponent, component, or component group of the airframe-engine system.

During the exploratory phases of any program, the general components, compressor, combustor, turbine, etc. are tested and evaluated as entities or even subcomponent entities. They have been assessed historically and tested purely from a performance point of view, with very little emphasis on stability as influenced by adjacent components. Test conditions and component design characteristics are normally only assessed from the general aero/thermo point of view. Specifically, the inputs and outputs of the component are only established such that they are generally in the range of the outputs of the upstream element and the inputs of the downstream element. Fundamentally, this is the only approach which makes good sense, since firm designs of upstream and downstream components are not available; however, in designing and testing of this nature, it should be remembered that not only must the aero/thermo inputs and outputs be at the correct general level, but so must the dynamic characteristics. This then implies that even during exploratory component development, emphasis must be placed upon the dynamic characteristics of the component as well as upon its aero/thermo performance. By considering and emphasizing the dynamic aspects of all exploratory efforts, the later efforts of integrating components into systems are greatly reduced. At the exploratory level, it is desirable to extract sufficient information from the component tests to enable parallel development of a mathematical model describing component performance both statically and dynamically. This model then forms a base along with other models like it for simulating component interactions and allows extensive analytical assessment of the system stability at a very early date. At this level, however, it is assumed that the only cross-component investigations conducted would be those over which an individual contractor had control and would probably not include specific representations of vendor-supplied items. Gross approximations of control functions can be applied at this level of development without

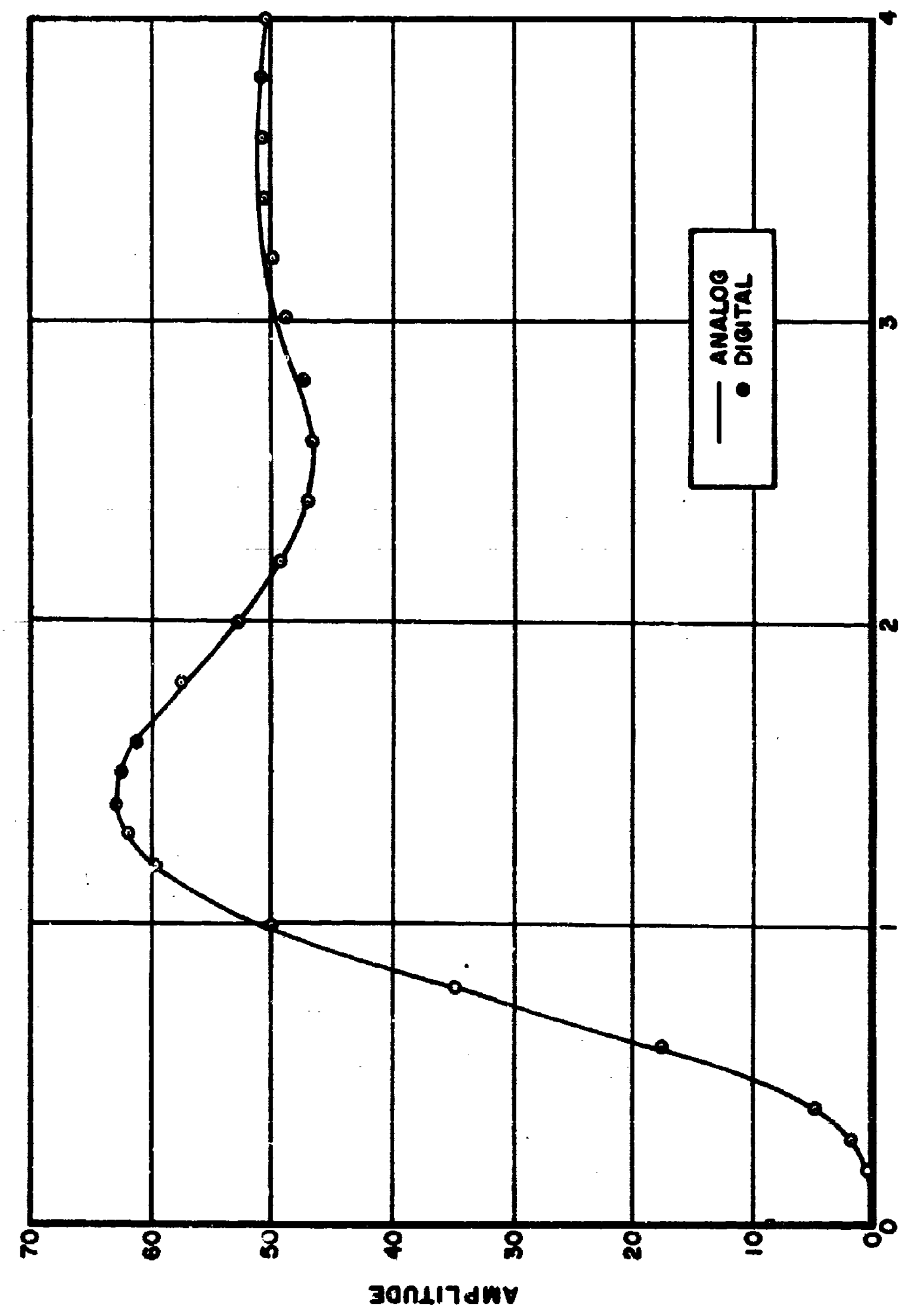


Figure 31. Analog Versus Digital Simulation

jeopardizing the general validity of the model. Also, it is considered quite acceptable to incorporate, in the early phases of development, component models based upon historical information or upon slight modifications to existing hardware.

The foregoing then is a description of a type of exploratory work not normally considered. In reality, it is no more than an extension of parametric or prediction efforts, supported by test, which is currently done. The major difference is that the primary concern is with dynamic interactions rather than with steady-state interactions. This in no way is meant to imply that the steady-state work is not valuable, but rather that, in an age of multiple-design-point philosophy, the steady-state work needs to be augmented at the earliest level of component development. The extension of the dynamic concept into advanced development is quite a simple matter if the exploratory work has been well done. During the advanced development phase, much more information is known about the application, the mission, the size, etc., and one is faced with tailoring of the general work already done.

As the components of the selected propulsion system are accurately defined, more extensive data will become available and should be integrated and correlated into the model. The component models should be assembled into a complete system, and at this point the outside sources of component inputs should be asked for a general description of their particular component. During the latter phases of advanced development, demonstrator testing of the entire system should be accomplished, and it is at this time that the first real component interactions can be tested and correlated to the model. If the task of component model building has been done conscientiously, the model should require only trimming to bring it into agreement with the hardware. If more than minor trimming is required for agreement, the individual component representations should be examined thoroughly and the rationale developed to explain why good correlation did not occur. In this way, future modeling efforts benefit from the omissions and inconsistencies of previous programs.

In concluding the advanced development phase, a model will be available which represents real hardware and there is test backup for the model at both the component and system level. The position is then reached wherein the step from demonstrated capability to required system performance can be validated by tested components and component buildups. Furthermore, since the demonstrated system is only rarely identical to that required by the overall system, the background of component models lends credence to geometric component scaling and minor performance-level modification required to specifically size the hardware. At this point, we have a well justified model of the system under the control of the individual contractor. It is now imperative that this model be utilized to the fullest extent possible to investigate the influence of components not directly controlled by the contractor. This investigation cannot be a one-way street, but should be conducted individually by all vendors concerned and then in combination with each other. Specifically, if the basic model is an engine, the airframe vendor will need to exercise the model in his installation, inlet, inlet control, secondary flow, etc. and the engine contractor will need to investigate the same influences in reverse. By so doing, some of the interference problems can be isolated long before they become cast in concrete hardware problems. If the responsible parties accomplish this phase of the interface well, and if the background work at the component level has been thorough, the resultant overall system model will provide an excellent base for finding interface problems and assessing the possibility of solving them.

SECTION IX

DETERMINATION OF STABILITY MARGIN

1. EVALUATION OF STABILITY MARGIN

Previous sections of this report have covered in varying detail the general elements upon which stability and performance depend. Many of these discussions indicate that sufficient knowledge to form an accurate estimate of the allowable operating range of many components of a propulsion system does not exist. In other areas, generally developed data does exist and derivations can be made. Examples of this data, integrated to form a stability margin evaluation, highlight the fact that a logical path through the details of such an evaluation does exist and that further work areas to improve the statistical quality of existing data can be identified.

A determination of stability margin is dependent on specifying the parameters of several aircraft and engines. For the stability margin illustration to be discussed in this section, the following are known:

Flight speed	Mach 1.9
Altitude	40,000 feet
Engine throttle setting	Maximum A/B
Engine corrected airflow	72%
Throttle transients	None
A/C maneuver load	1 g

It is further assumed that test data for the inlet is defined for the flight condition shown in Figure 32. The magnitude of the distortion ($\Delta P_T/P_T)_2$ is based on the maximum total pressure across the inlet minus the minimum total pressure across the inlet, divided by the area-weighted average pressure, again based on the complete face of the engine. Furthermore, the test data on the integrated compressor aerodynamic characteristics are presumed to have been developed.

Test data generally available indicates that, for an equal measure of distortion, the surge line shift due to radial distortion will be less than will be the surge line shift due to circumferential distortion. This trend is illustrated in Figure 33. As shown on this chart, tip radial distortion is more influential than hub radial distortion at the high-speed portion of the compressor map, while at the low-speed end of the map, just the opposite is true. Figure 33 also indicates that circumferential distortion shifts the surge line further than either tip or hub radial distortion. Figure 34 illustrates the general trend of the surge line shift with increasing level of input circumferential distortion $(\Delta P_T/P_T)_2$, which, as the available data indicates, is the most influential type of distortion. Figure 33 and 34 consider only the magnitude of the distortion. The extent of the distortion, when properly treated, will yield the working chart necessary for analysis of the stability margin of this case.

Since circumferential distortion is the controlling type of distortion, the compressor aerodynamic configuration will have been previously tested to have generated the data for Figure 35. This chart shows the surge line shift as a function of the circumferential extent (θ) of the inlet depression $(\Delta P_T/P_T)_2$, which is shown for levels of 5, 10, 15, and 20%. The trend shown in Figure 35 is based on test data derived from compressor rig tests and indicates that when the circumferential extent of a distortion progresses much beyond $\theta \sim 45°$, the surge pressure ratio depression is nearly complete, and furthermore, from about $\theta \sim 75°$-$180°$, the surge pressure ratio depression is very nearly constant for a given level of

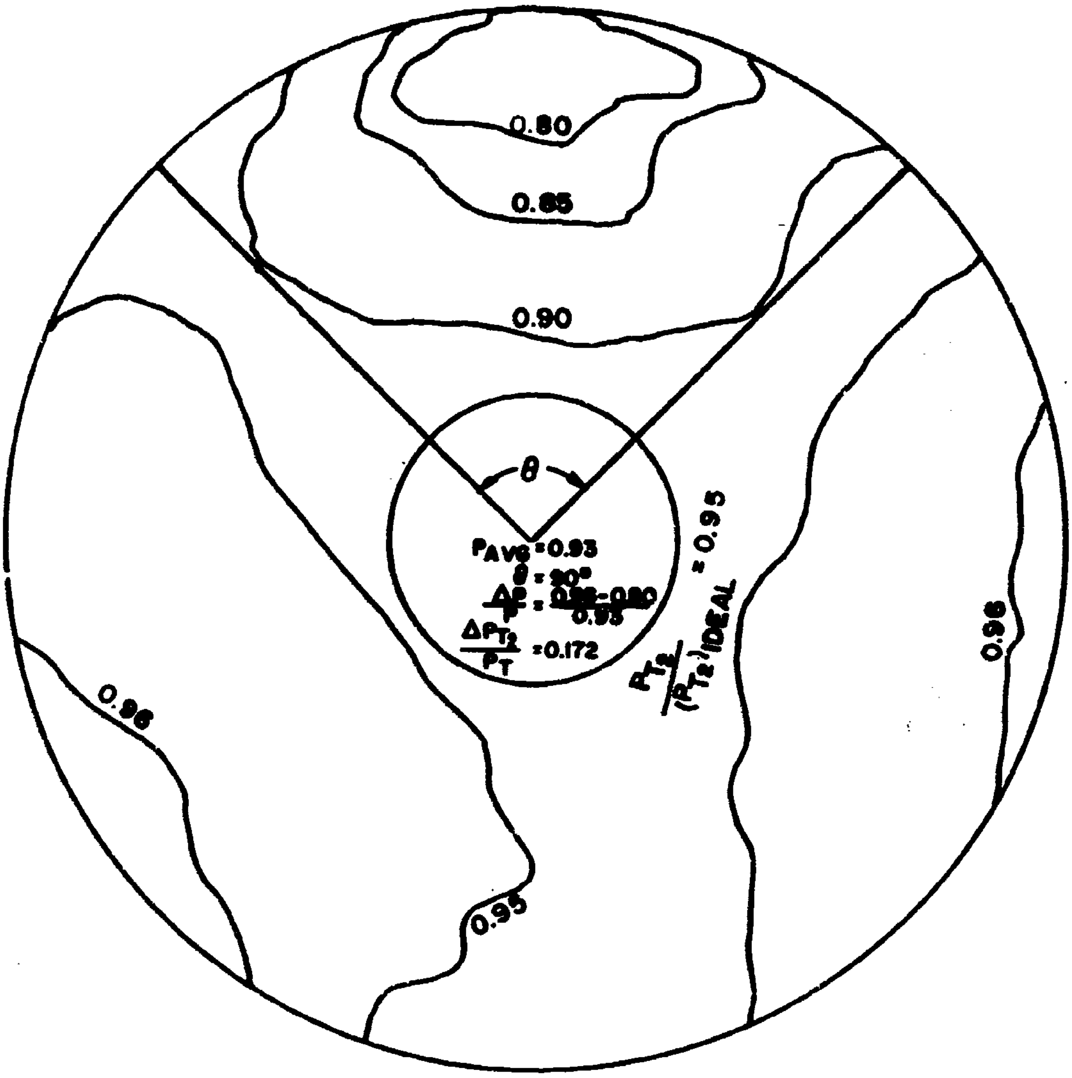


Figure 32. Inlet Map at Mach 1.9, 40,000 Ft Maximum A/B

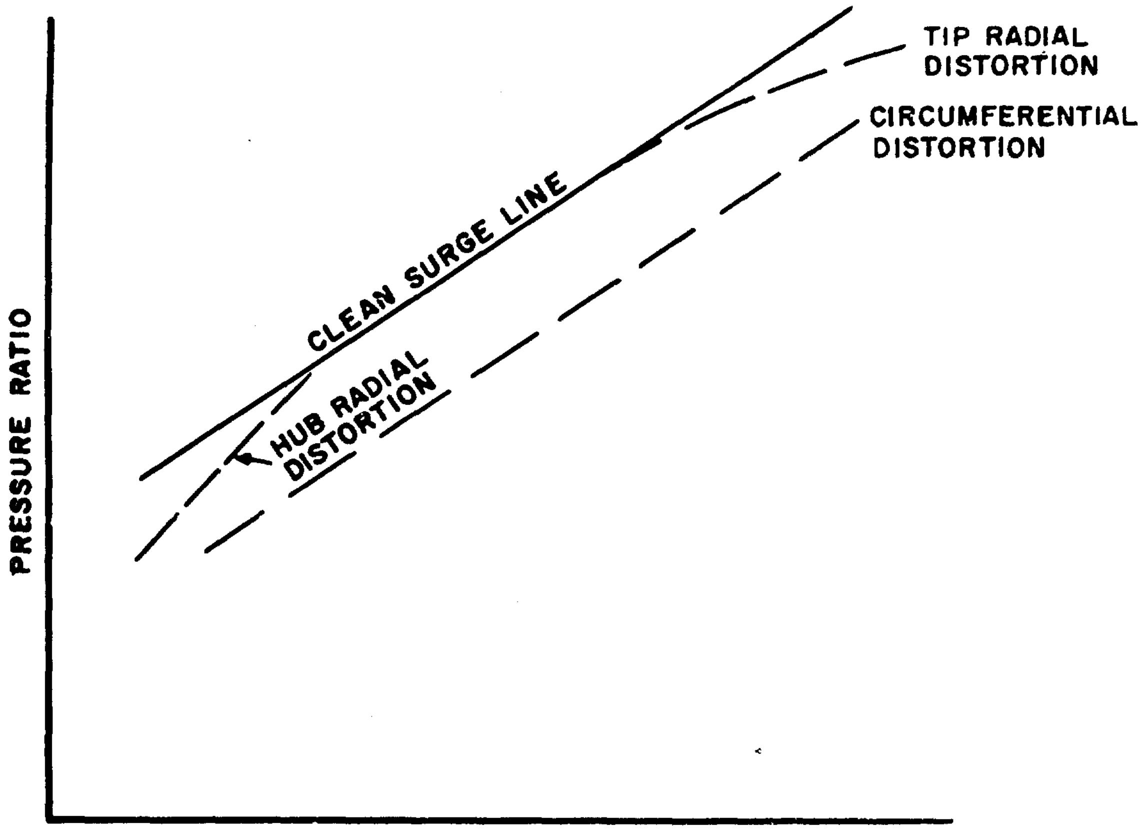


Figure 33. General Compressor Surge Line Shift

input distortion. Figure 35 is representative of the surge pressure ratio depression, either along a given constant corrected speed line or, alternatively, in a given portion of the compressor map. These influence factors will have been generated across the expected operational band for the compressor application, but, for the purposes of this illustration, these curves (Figure 35) are chosen as being representative of all corrected speeds. This assumption is valid since the corrected speed range for the problem about to be discussed does not vary more than 3%.

Figure 36 is a normalized compressor map for the engine-aircraft operational point previously outlined at the beginning of this section. Point A is the steady-state operating point that corresponds to the stated care, i.e., Mach 1.9, 40,000 ft altitude, maximum A/B, engine corrected airflow of 72%, 1.0 g maneuver load, and no throttle transient. The engine under discussion is a single spool, augmented turbojet with a nominal pressure ratio of 12. To develop a stability margin illustration, an augmentor blowout with no throttle movement by the pilot is postulated. The engine control is of the type that governs engine speed against a limiting value. When the augmentor blows out, the exhaust nozzle area is instantaneously too large. This lowers the back pressure on the turbine, increasing the turbine pressure ratio. This condition will appear on the compressor map as point B. The vector from A to B represents only the vectorial path on the compressor map. The real path may be above or below this path, but point B represents the instantaneous position just prior to remedial action having been taken by the engine fuel control. When the A/B blowout occurred, the control was on the required-to-run schedule, but the unbalanced torque due to the higher extraction

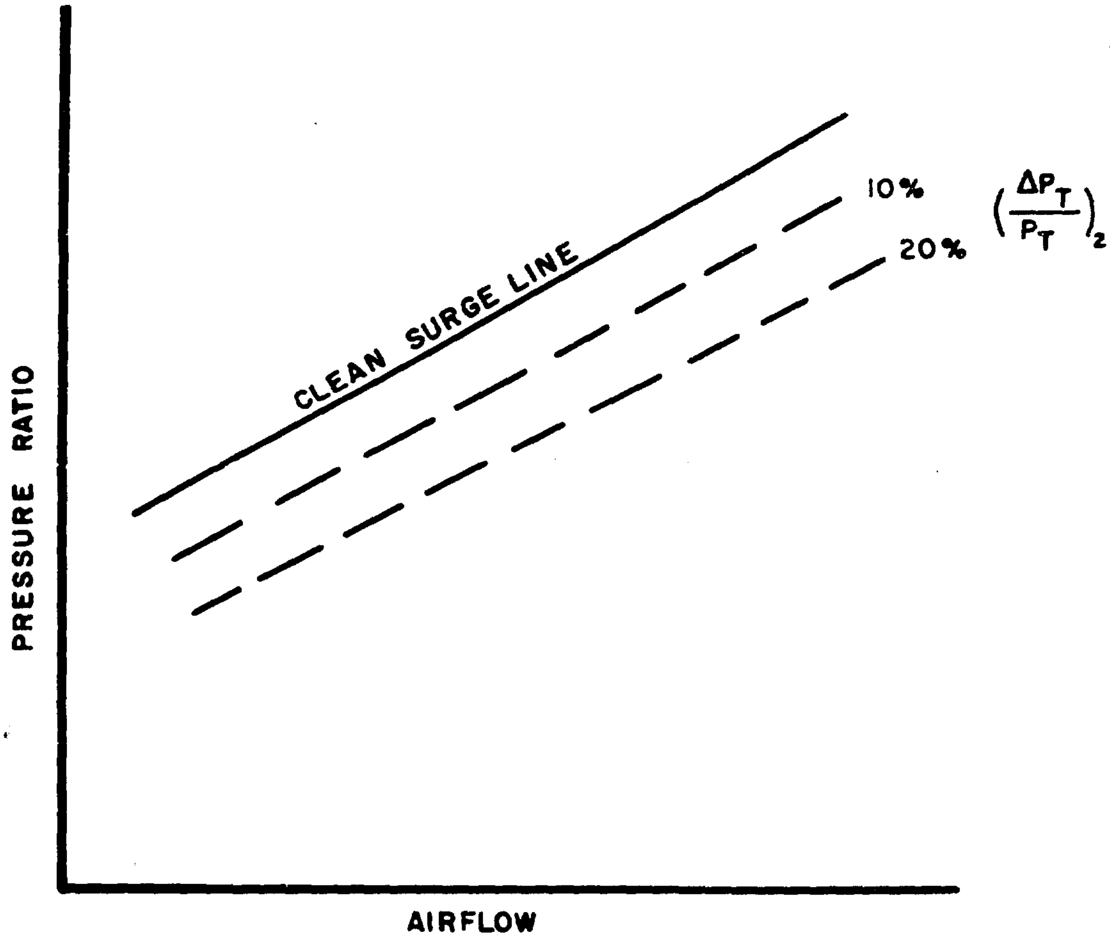


Figure 34. Compressor Surge Line Shift With Varying Levels of Circumferential Distortion

pressure ratio of the turbine initiated an engine speed increase. The fuel control, having sensed an increase in RPM and a too large exhaust nozzle area will have shifted to a deceleration fuel flow setting and will have started a nozzle closure. Point B then represents the maximum excursion in the overspeed direction. From point B to point C, the engine RPM is decelerating along the maximum deceleration cam profile with nozzle closure slew rate lagging. Just prior to point C, the fuel control, sensing an underspeed condition, shifts to the maximum acceleration cam profile, and, at this point, the nozzle closure slew rate has overdriven the exhaust nozzle area such that the path is from C to D. From D to A, the engine control trims the fuel flow, RPM, and exhaust nozzle area to arrive at point A once more. The reader should note that the throttle lever remained in the maximum A/B position. If the augmentor light system, augmentor fuel flow, and nozzle area all were controlled automatically and were functionally interrelated, the augmentor should have relit somewhere between a point just prior to point C and just after point D. If this were the case, point D would occur at some point higher on the compressor map than represented in Figure 36. The trim then would be from the higher point D to point A, balancing main fuel flow, RPM, A/B fuel flow, and exhaust nozzle area. If the cause of the initial A/B blowout has not then been removed, the cycle will repeat.

If, for the A/B blowout points discussed, the distortion levels from Figure 35 are overlaid on the compressor map with the entry point to Figure 35 taken at a circumferential extent, θ, of 90° and if, as previously indicated, the assumption that Figure 35 is representative of

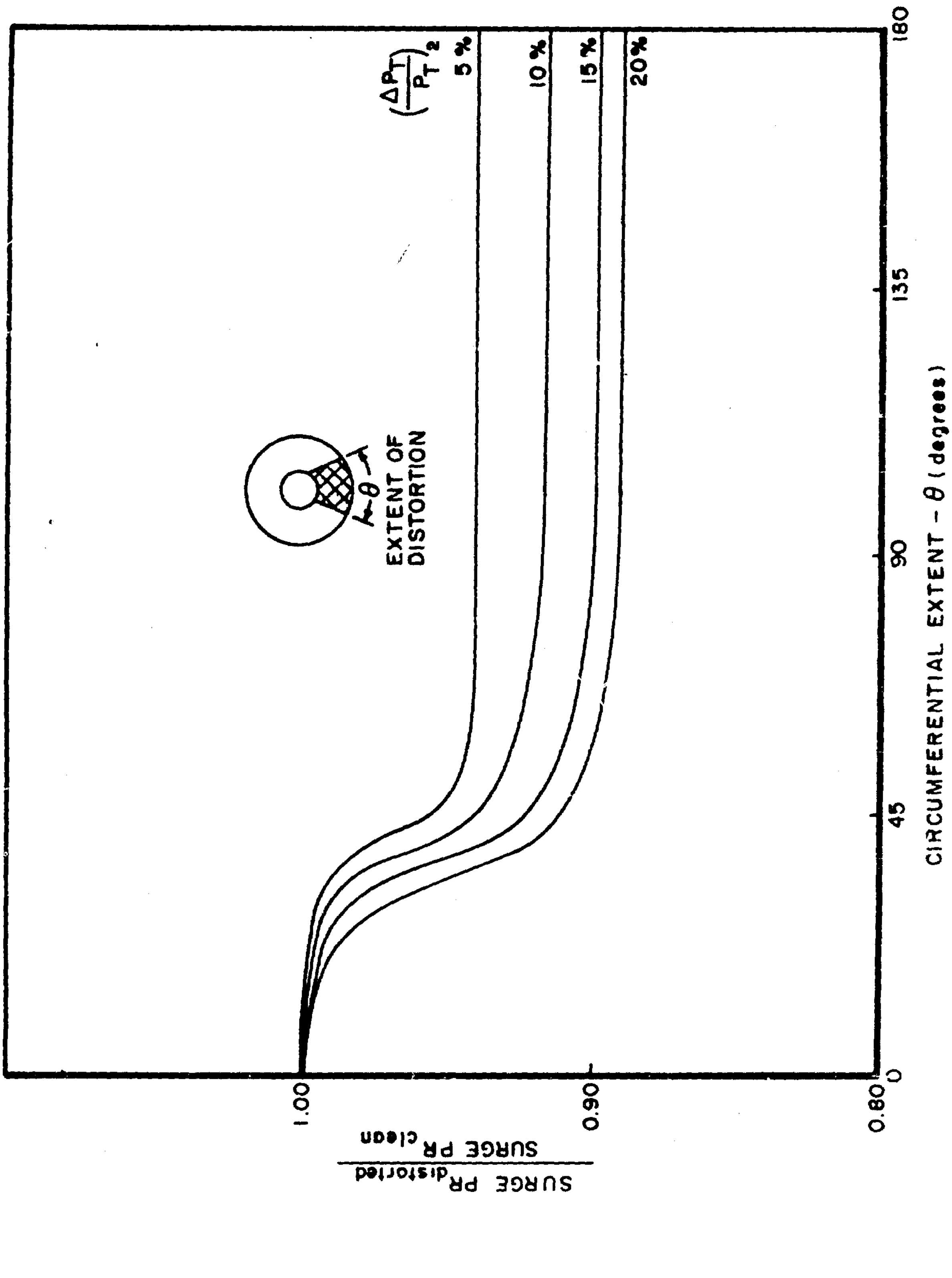


Figure 35. Typical Circumferential Distortion Effects on Axial Compressor

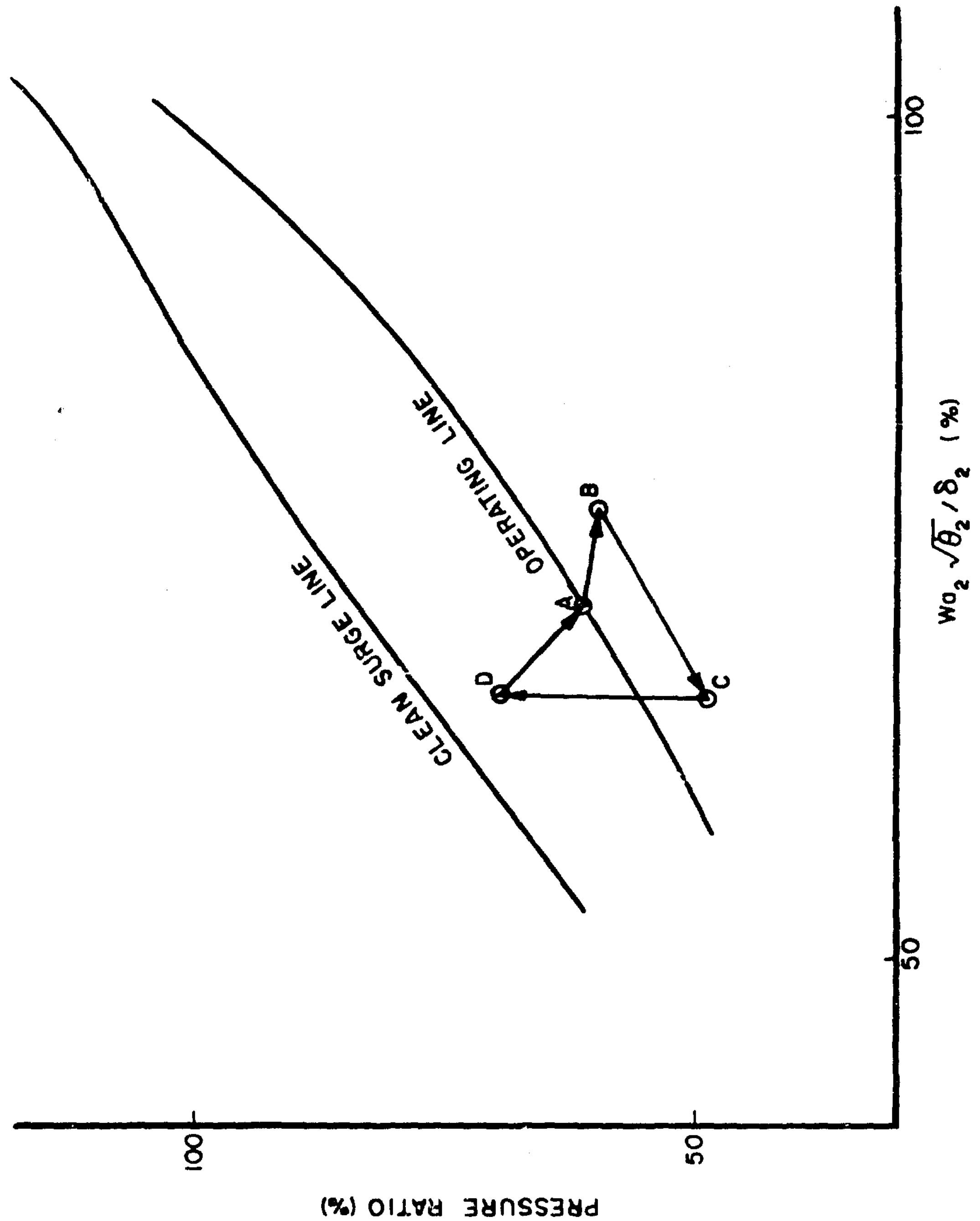


Figure 36. Example of Compressor Surge Line Shift With Distortion

all corrected speeds in the range of interest, Figure 37 will result. Point D, in Figure 37, is seen to lie between the distortion-shifted surge line represented by 10% $(\Delta P_T/P_T)_2$ and 20% $(\Delta P_T/P_T)_2$. In Figure 32, the distortion level is seen to be 17.2% $(\Delta P_T/P_T)_2$ for θ of 90°, which corresponds to a distortion line on Figure 37 below point D; therefore the compressor/engine will have become unstable. As previously described, point D corresponds to a condition wherein the augmentor had not relit. If the augmentor had relit, point D would have been functionally positioned on the map at a higher pressure ratio, thereby forcing this transient operating condition deeper into the unstable region of compressor operation. For the original case, i.e., no relight of the augmentor between points B, C, and D, inspection of the engine compressor's steady-state operational characteristics map reveals that at the corrected flow rate corresponding to point D, the undistorted surge line represents an instantaneous excess pressure ratio capability of 37.7% with respect to the steady-state operating line pressure ratio requirement. The dynamic transient imposed by the afterburner blowout has required the utilization of 25.4% of this excess pressure ratio capability due to the pulsed position of point D. Furthermore, the 17% inlet distortion has imposed a depression of the undistorted surge line to a functional value of 23.2% excess pressure ratio with respect to the steady-state operating line. The summation of these excess pressure ratio capabilities leads to an instantaneous "negative stability margin" of 2.2% for the transient operational point D. Therefore, the engine, having this negative stability margin, under these conditions will be unstable. This illustrative stability margin determination is tabulated below.

Instantaneous Excess Pressure Ratio With Respect to Steady State Operating Line	Condition
+37.7%	Undistorted surge line
-14.5%	Surge line deterioration due to distortion
+23.2%	Functional value of distorted surge line
-25.4%	Functional position of point D
-2.2%	Net stability margin

When negative stability margin has occurred, two remedial paths are immediately available in a weapon system engineering development program. These are control schedule modification or compressor/engine derating. Control schedule modification is basically shifting of the time constants associated with the control functions, i.e., fuel flow schedules for acceleration-deceleration anticipatory functions controlling exhaust nozzle area with respect to conditions and time, inlet area and shock position mechanism actuation schedules, etc. The latitude for solving the problem through control function shifting is limited by the amount of change available with the actual hardware and the requirement to insure that changes incorporated for relief of the instability at the particular operational condition do not excessively deteriorate the stability margin of the propulsion system during other operating modes. Therefore, control changes, being of limited scope within the bounds of hardware limitations and operational constraints, are considered to be insufficient to yield major relief for propulsion system instability and must be considered as trimming functions only. Under the conditions discussed, the occurrence of compressor-engine instability would have to be remedied by derating the engine to point A, shown on Figure 37. This would result in the steady-state operating line labeled "De-rated Operating Line." A condition requiring derate of the engine along this line is the type of situation that would occur if the aircraft-engine delivery schedule were in serious jeopardy. Such a condition often exists when an aircraft weapon system developed is constrained to a tight delivery schedule. If the tight schedule is overlaid with a strong requirement for performance, weight, and cost, the trade between these diverging objectives becomes an impossibility. If timing and cost requirements were such that

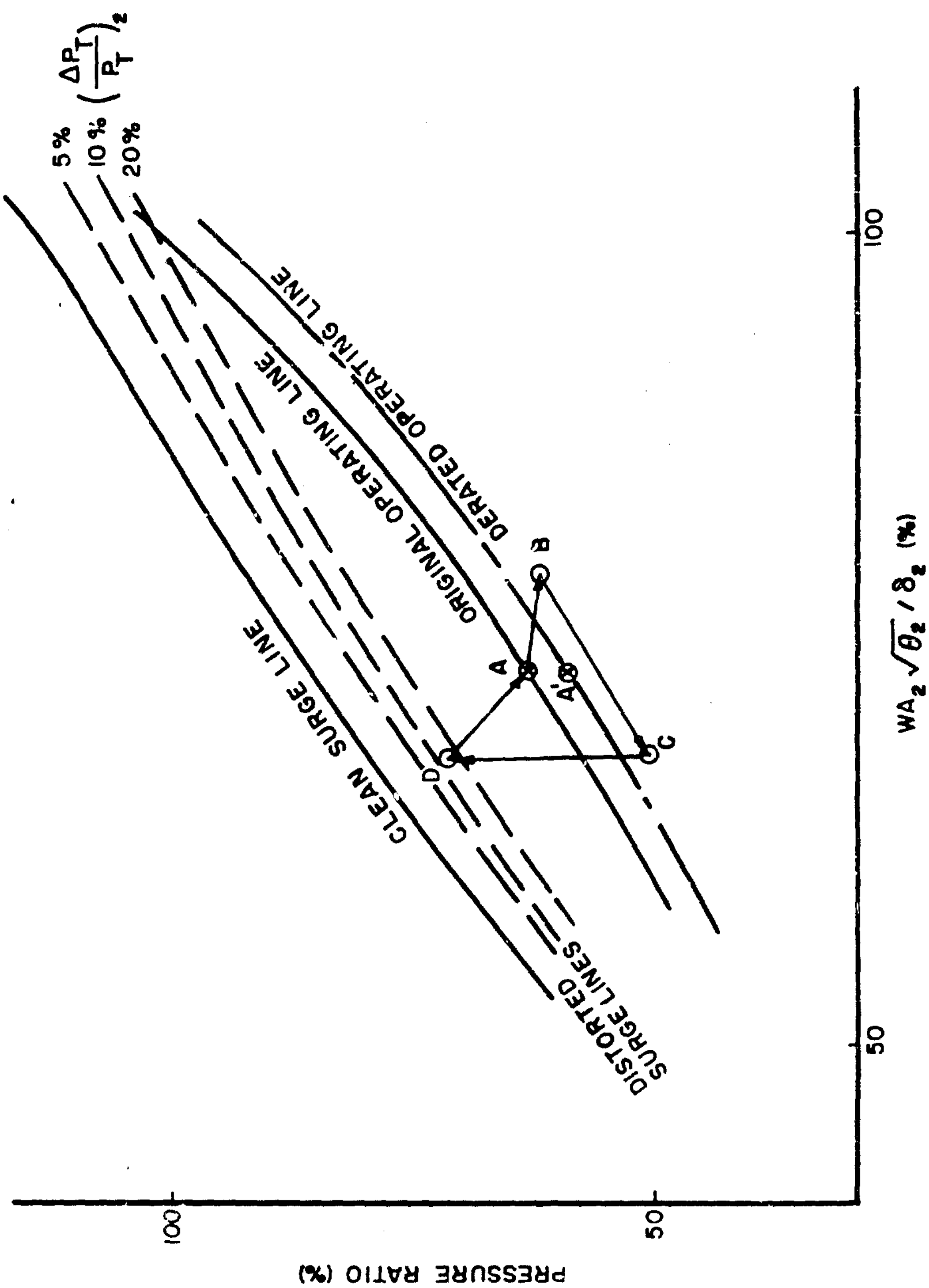


Figure 37. Compressor Derating in Reaction to Distortion

they were overriding, and the engine had to be derated along the line through point A with no time available for redesign of the other components, the engine performance would be reduced to the levels of the degraded 12 Pressure Ratio turbojet of Section IV. These performance degradations pertain only to the compressor, main combustor, and turbine portions of the engine. Additional losses will occur beyond those discussed in Section IV which considered only dry turbojets, whereas this example will have these same losses plus the performance losses associated with off-design operation of the augmentation system. Furthermore, the performance penalties associated with an increase in augmentor dry pressure loss will also be present and have to be accounted for when the engine is operated at a nonaugmented condition, such as subsonic cruise. Section IV and the above discussion consider and isolate only the main engine components' performance influence coefficients. Derating the engine along the line shown in Figure 37 will also influence the performance of the inlet and the nozzle and ejector. As can be appreciated, deteriorations in the performance of inlet, main engine, and the nozzle and ejector system can adversely affect not only the thrust and fuel flow rate of the propulsion system, but can also adversely affect the aircraft performance. This can occur when the pumping characteristics of the main engine have degraded and shifted the levels of bypass drag, spillage drag, and ejector pumping. The latter could change such drag functions as fuselage boundary layer removal, boat-tail drag, etc.

2. SUMMARY

Determining the stability margin and evaluating it have been shown to be both feasible and essential. Propulsion system interface with the aircraft will become more complex as systems' requirements become more demanding. These requirements cannot be relaxed. Propulsion system stability, therefore, with its influence on overall weapon system performance, timing, cost, and weight, will become increasingly emphasized. To avoid belaboring the point, reference is made to Section IV, Paragraph 14, Decisions Leading to Performance Compromise and Paragraph 15, Procedures to Preclude Derating. These two paragraphs treat primarily propulsion system stability as affected by the engine compressor. As brought out in this Section, there are other components in a propulsion system which can exert a negative influence on either stability margin or performance and weight. Timely derivation of both the performance and stability characteristics of each of the subcomponents of the overall propulsion system and the aircraft interface with the propulsion system is mandatory prior to the start of complete weapon system development. Furthermore, the trades between stability, performance, weight, cost, and timing must have been defined within the same time span.

REFERENCES

1. Aerodynamic Design of Axial-Flow Compressors. NASA SP-36. 1965.

2. W. H. Sens, W. S. Bailey, C. B. Smith. Effects of Supersonic Inlet Characteristics on Gas Turbine Performance. Presented at the Institute of Aeronautical Sciences Meeting, 11 March 1955.

3. J. S. Alford. Inlet Flow Distortion Index. Presented at International Days of Aeronautical Sciences. Paris, France. 27-29 May 1957.

4. L. G. Dawson. The Engine Designers' Point of View. AGARD-O-GRAPH 103, Part I. 64 Rue de Varenne, Paris 7E, France. Ocotober 1965.

5. A. C. Loresey. "Gas Turbine Development-Thirteen and a Half Years in Commercial Aircraft." Journal of the Royal Aeronautical Society, Vol. 68. August 1964.

6. A. Ferri and L. M. Nucci. The Origin of Aerodynamic Instability of Supersonic Inlets at Subcritical Conditions. NACA RM L 50K30. January 1951.

7. R. B. Pearce. "Causes and Control of Powerplant Surge." Aviation Week. 1950.

8. C. L. Dailey. "Supersonic Diffuser Instability." Journal of Aeronautical Science. November 1955.

9. W. H. Sterbentz and J. C. Evvard. Criterions for Prediction and Control of Ramjet Flow Pulsations. NACA TN 3506. 1955.

10. W. R. Haageson. Flight Experience with Mach 3 Inlet Control Systems in the XB-70. Propulsion Specialist Conference, AIAA. 1966.

11. A. W. Martin, L. W. Victor. Test Observations of Inlet-Engine Dynamic Interactions. Propulsion Specialist Conference, AIAA. 1966.

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13. ABSTRACT

A major consideration in the current design and operation of air vehicles is the stability of the total propulsion system. The derivation of stability criteria involves many evaluations and determinations at the subcomponent level to ascertain stable operating regions and the performance coefficients that are associated with the elemental margins. Painstaking compilation of the performance and stability margin for the key flight points of operation can thus be derived and a numerical value assessed with respect to stability margin. These data, for the most part, are dependent upon empirical coefficients drawn from previous weapon system developments. As such, a major effort is warranted to define alternative approaches to the problems of stability margin assessment that will provide a likely improvement in performance in the system. Development of the capability for improved performance and defined stability margin is directly dependent on an accurately determined interface between an engine and the associated airframe. Such a determination can be achieved only if there is expert liaison between and complete cooperation by the engine and airframe development companies. The interface between such companies must be strongly developed and is extremely critical in time-phased relationships between the engine and airframe development companies.

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