

NASA Contractor Report 3673

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Literature Search of Publications Concerning the Prediction of Dynamic Inlet Flow Distortion and Related Topics

W. G. Schweikhard and Yen-Sen Chen

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Literature Search of Publications
Concerning the Prediction of
Dynamic Inlet Flow Distortion
and Related Topics

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SUMMARY

Dynamic distortion test and analysis publications prior to March 1981 were surveyed to determine inlet flow dynamic distortion prediction methods and to catalogue experimental and analytical information concerning inlet flow dynamics at the engine-inlet interface of conventional aircraft.

This report presents the results of this survey of the open literature to identify publications concerning the prediction of inlet flow dynamic distortion and related subjects. It focuses primarily on papers which reveal dynamic nature of flow at the inlet/engine interface. An attempt has been made to include all materials, however, subjects such as internal engine induced instability papers and many V/STOL application reports have not been included.

Sixty-five publications were found and are briefly summarized and tabulated according to topic. Also they are cross-referenced according to content and nature of the investigation such as predictive, experimental, analytical and types of tests. Three appendices include lists of references, authors, organizations and agencies conducting the studies. Also selected materials -- summaries, introductions and conclusions -- from the reports are included.

Few reports were found covering prediction methods. The three predictive methods found are those of Melick, Jacox and Motycka. The latter two require extensive high response pressure measurements at the compressor face, while the Melick Technique can function with as few as one or two high response measurements.

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LIST OF ACRONYMS

<u>Acronym</u>	<u>Definition</u>
AAAF	Association Aeronautique et Astronautique de France
AD	Arnold Engineering Development Center
AGARD	Advisory Group for Aerospace Research and Development
AIAA	American Institute of Aeronautics and Astronautics
ASME	American Society of Mechanical Engineers
CR	Contractor's Report
MBB	Messerschmitt-Boelkow-Blohm G.M.b.H. Ottobrunn (West Germany)
NASA	National Aeronautics and Space Administration
NTIS	National Technical Information Service
PSD	Power Spectral Density Function
RMS	Root Mean Square Value
TM	Technical Memorandum
TN	Technical Note
TR	Technical Report
VSD	Vought Systems Division

I. INTRODUCTION

The requirement for relatively uniform nonturbulent flow to the engine has long been appreciated but the definition of distortion as the deviation from uniform flow has only been developed since the early 1960's. Since then, distortion descriptors have been developed to describe the various distortion patterns in terms of the total pressure measured at the compressor face, and to reveal the susceptibility of a particular engine to the magnitude of these descriptors. While the significance of time variant as well as steady state distortion has become better understood the prediction of extreme values of the time variant distortions have remained elusive. This is especially true if the distortions are to be predicted with a minimum of experimental information.

Considerable work has been done in the dynamic distortion field over the past 15 years; however, little has been done to assemble it into a cohesive body of knowledge. This survey of open literature is a step in that direction. Initial computer searches conducted, from July 1 to August 31, 1980, did not produce the desired comprehensive coverage of the subject. A second search was conducted January 1 to March 31, 1981. The search involved a manual search of NTIS, NASA and AIAA publications, a computer search by the NASA Lewis technical library and a search of the personal libraries of key NASA personnel who have been active in the field. In addition, references cited in the literature found in this study were backtracked to produce a few of the more obscure references that were not uncovered by other means. While a concerted effort has been made to include all of the available reports and papers on distortion of the engine inlet interface, a few may have been missed. This work has been accomplished under the sponsorship of NASA Lewis Research Center as a part of a grant to study dynamic distortion prediction and analysis.

The purpose of this report is to document compressor face distortion related literature in detail and to catalog and index the information contained therein so that others working in this area may benefit by knowing where specific types of information may be found.

2. LITERATURE SEARCH PROCEDURE

Three computer searches have been conducted of open literature concerning inlet flow dynamics, dynamic distortion and related subjects. One search was of National Technical Information Service (NTIS) publications from 1964 to May 1980. The second search was of Compendex publications (Engineering Index) from 1970 to May 1980. Only publications prior to 1976 were found in the above searches. The third one was conducted by NASA Lewis library for publications from 1975 to March 1981.

Keywords used to process these searches were:

FLOW DISTORTION
INLET FLOW
INLET PRESSURE
INTAKE SYSTEMS
ENGINE INLETS
INTERNAL COMPRESSION
NOSE INLET
SIDE INLET
SUPERSONIC INLETS

In addition to the above mentioned computer searches, a manual search was conducted using the "Index of NASA Technical Publications" for the period between 1970 and 1979. Only 11 additional papers that relate to inlet flow distortion were found from this source.

A publication was considered pertinent for inclusion in this report if one or more of the following criteria were met:

1. Publications dealt with Prediction of Dynamic Distortion (most preferred).
2. Analytical Theory and/or Experimental Data on Distortion were presented.
3. Publications dealt with Inlet Flow Distortion and/or its effect on compressor stall.
4. Distortion Factors were discussed.
5. Publication related to vortex theory and inlet flow field effects.

Review of the above mentioned NTIS and Compendex Computer listings found that some publications dealt with acoustic effects caused by flow distortion; these were not included in this survey.

3. GENERAL FINDINGS

A total of sixty-five reports and papers have been found that reports and papers have been found that relate the subject of inlet distortion of conventional aircraft. The majority of these deal with high performance military aircraft or supersonic cruise inlets where distortion becomes more of a problem because of extreme aircraft altitudes or speed. V/STOL applications are a subject unto themselves in the extreme low speed region and are highly sensitive to the particular configuration. They are not included in the 65 references listed.

The references listed each cover a variety of subjects. These include theoretical and statistical analyses, steady state and dynamic experimental results, measurement instrumentation and data processing, scale and internal/external flow effects, and vortex theory. As would be expected, each report or paper covers a number of these topics so that the study of any one topic requires cross-referencing between several reports. An attempt has been made to identify the topics covered in each report and to cross-reference them so that the would-be investigator has some idea where to look.

The first grouping shown in table 3.1 reveals the general topics or subtopics that describe the content of the references listed on the right hand side. It can be seen that there are 49 references relating theoretical analysis and 39 references containing experimental results; 26 of them contain both theoretical and experimental discussions. Also note that a particular reference may appear in several topic areas.

TABLE 3.1: INDEX TO TOPICS DEALT WITH IN REFERENCES 1 THOROUGH 65

<u>Topic</u>	<u>References</u>
1. Theoretical Analysis:	1, 2, 3, 4, 5, 6, 7, 11, 13, 14, 15, 16, 17, 18, 19, 20, 21, 22, 23, 24, 25, 26, 27, 28, 29, 31, 32, 33, 34, 35, 36, 37, 42, 43, 44, 46, 48, 49, 50, 51, 52, 53, 55, 56, 59, 61, 62, 63, 64
2. Experimental Results:	1, 2, 3, 4, 5, 7, 8, 9, 10, 12, 13, 14, 15, 16, 17, 19, 20, 21, 23, 24, 26, 30, 32, 35, 36, 38, 39, 41, 45, 47, 53, 55, 57, 58, 61, 62, 63, 64, 65
3. Contains both 1 & 2:	1, 2, 3, 4, 5, 7, 13, 14, 15, 16, 17, 19, 20, 21, 23, 24, 26, 32, 35, 36, 53, 55, 61, 62, 63, 64

4. Statistical Analysis:	1, 2, 3, 4, 5, 13, 26, 29, 30, 34, 35, 37, 42, 65
5. Dynamic (& Steady State) Distortion:	1, 2, 3, 4, 5, 6, 7, 8, 10, 11, 12, 13, 15, 16, 21, 22, 24, 25, 26, 27, 28, 29, 30, 32, 34, 35, 36, 37, 38, 39, 40, 41, 42, 43, 44, 45, 46, 47, 48, 50, 51, 53, 54, 55, 56, 57, 58, 59, 60, 61, 62, 63, 64, 65
6. Steady State Distortion Only:	9, 17, 20, 23, 33, 49, 52
7. External and Internal Flow:	8, 14, 18, 19, 31
8. Vortex Theory:	1, 2, 3, 4, 5, 18, 29, 31, 33, 42, 54
9. Scale Effect:	1, 2, 3, 4, 5, 12, 13, 15, 29, 42
10. Probe Sensitivity:	1, 3, 4, 5, 9, 58

A second breakdown of this information is shown in Table 3.2 where an attempt has been made to cross-reference the topics and subtopics according to whether the material is analytical or experimental. Between the information contained in Tables 3.1 and 3.2 it should be possible for one to identify reports of particular interest. Once this has been done Table 3.3 will be useful to identify the report number, first author, title, and content of that report. If more detailed information is required Appendix C contains copies of the summaries, conclusions and in some cases introductions of each of the reports.

Appendix A contains complete reference information for all of the references in the conventional reference list format. Appendix B contains an index of authors and organizations who are or have been active in dynamic inlet distortion investigations. From these it is observed that few authors have more than one publication on distortion and that the majority of the effort has been conducted or sponsored by NASA or the Air Force.

NO.	REFERENCE	TITLE AND REMARKS
1	NASA TM-X-73145 By: Melick, et al.	<u>ESTIMATING MAXIMUM INSTANTANEOUS DISTORTION FROM INLET TOTAL PRESSURE RMS & PSD MEASUREMENTS.</u> Estimates inlet instantaneous distortion through inlet pressure RMS and PSD measurements using a physical turbulent flow model constructed by a group of random vortices which are random in strength and size. Finds the extreme value of statistical variables. Includes inlet configuration and scale effects.
2	NASA CR-114577 By: Melick et al.	<u>ANALYSIS OF INLET FLOW DISTORTION AND TURBULENCE EFFECTS ON COMPRESSOR STABILITY.</u> Discusses the effect of steady-state circumferential total pressure distortion on compressor stall characteristics with comparisons between analytical and experimental data. Investigates inlet turbulent flow model.
3	VSD, LTV Aerospace Corp. TR-2-57110/ SR-2309 By: Melick, et al.	<u>ESTIMATING MAXIMUM INSTANTANEOUS DISTORTION FROM INLET TOTAL PRESSURE RMS & PSD MEASUREMENTS.</u> Develop an inexpensive alternative procedure to statistically evaluate the extreme values of instantaneous inlet distortion through inlet pressure RMS and PSD measurements. Includes the effects of number of probes is compared with 40-probe basis.
4	TR-2-57110/ 5R-3210 (Volume I) By: Ybarra, et al.	<u>COMPUTER PROGRAM DOCUMENTATION FOR MELICK'S METHOD (Vol. 1).</u> Illustrates the computer program and test cases following the development of Reference 3.
5	TR-2-57110/ 5R-2310 (Volume II) By: Ybarra, et al.	<u>COMPUTER PROGRAM DOCUMENTATION FOR MELICK'S METHOD (Vol. 2).</u> Illustrates the computer program and test cases following the development of Reference 3.

Table 3.3. SYNOPSIS OF THE SUBJECT MATTER CONTAINED IN REFERENCES 1 THROUGH 65

NO.	REFERENCE	TITLE AND REMARKS
6	NTIS AD/A-004-104 By: Jacocks, et al.	<u>STATISTICAL PREDICTION OF MAXIMUM TIME VARIANT INLET DISTORTION LEVELS.</u> A probabilistic model of distortion is proposed with three parameters being evaluated by fitting the inlet data using the method of maximum likelihood based on Gumbel's extreme value statistics. Sampling rates are discussed in context with Moore's similarity parameter.
7	NASA TM-X-73118 By: Latham	<u>INVESTIGATION OF TWO BIFURCATED-DUCT INLET SYSTEMS FROM MACH 0 TO 2.0 OVER A WIDE RANGE OF ANGLES OF ATTACK.</u> Describes test measurements of engine face total pressure recovery, steady-state and dynamic distortion and surface static pressures on the forebody and inlet surfaces of a 15.354% scale lightweight fighter-type inlet-forebody. Includes test data only.
8	NASA CR-264 By: Motyka, et al.	<u>INLET-TO-INLET SHOCK INTERFERENCE TESTS.</u> Discusses the effect of the influence of shock waves from another inlet on inlet pressure recovery and inlet flow
9	NASA TM-72859 By: Stoll, et al.	<u>EFFECT OF NUMBER OF PROBES AND THEIR ORIENTATION ON THE CALCULATION OF SEVERAL DISTORTION DISCRIPTORS.</u> Discusses the effects of the number and position of pressure probes on the calculation of five steady-state compressor face distortion descriptors and average compressor face total pressure based on the results of 320-probe station derived from a 40-probe rotatable rake.
10	NASA TM-X-3540 By: Costakis	<u>CHARACTER OF RANDOM INLET PRESSURE FLUCTUATIONS DURING FLIGHT OF F-111A AIRPLANE.</u> Compressor face dynamic total pressures from four F-111 flights were analyzed. Statistics of the nonstationary data were investigated by treating the data in a quasi-stationary manner.

Table 3.3 CONTINUED

NO.	REFERENCE	TITLE AND REMARKS
11	NASA TM-X-71505 By: Graber, et al.	<u>SUMMARY OF RECENT INVESTIGATIONS OF INLET FLOW DISTORTION EFFECT ON ENGINE STABILITY.</u> Provides a review of recent experimental results, analytical procedures and test techniques employed to evaluate the effects of inlet flow distortion on engine stability characteristics.
12	McDonnell Aircraft Company (1974) By: Farr, et al.	<u>SYSTEM FOR EVALUATION OF F-15 INLET DYNAMIC DISTORTION.</u> An instrumentation and data acquisition system for evaluating inlet dynamic distortion has been developed for use in the F-15 full-scale wind-tunnel and flight-test programs.
13	NASA TM-X-67495 By: Burcham	<u>A FLIGHT INVESTIGATION OF STEADY STATE AND DYNAMIC PRESSURE PHENOMENA IN THE AIR INLETS OF SUPERSONIC AIRCRAFT.</u> Estimates inlet steady-state and dynamic performance due to normal shock oscillation using statistical parameters. Shows the results of flight test of F-111A at Mach 2.0. Include scale effect through wind tunnel testing.
14	NASA TM-X-66885 By: King, et al.	<u>STUDIES OF AIRCRAFT FLOW FIELDS AT INLET LOCATIONS.</u> Investigates flow of inlet through wind tunnel test for several configurations and a wide range of Mach number and angles of attack. Predicts the results by using numerical method of Walittretal. Discusses sensitivity of main variables on inlet flow field.
15	NASA TN-D-6679 By: Hughes	<u>FLIGHT-DETERMINED CHARACTERISTICS OF AN AIR INTAKE SYSTEM ON AN F-111A AIRPLANE.</u> Discusses the quasi-steady-state flow phenomena of a variable geometry inlet system of F-111A. Contains a broad range of Mach number, altitude and angle of attack. Compares 1/6 and full-scale wind tunnel model data.

Table 3.3 CONTINUED

NO.	REFERENCE	TITLE AND REMARKS
16	NASA TN-D-7328 By: Burcham, et al.	<u>STEADY-STATE AND DYNAMIC PRESSURE PHENOMENA IN THE PROPULSION SYSTEM OF AN F-111A AIRPLANE.</u> Investigates the effect of inlet steady-state and dynamic pressure fluctuations on inlet pressure recovery, distortion and turbulence factor data as function of Mach number, inlet flow, inlet geometry and angles of attack and sideslip by using a random turbulence generator ground facilities.
17	NASA TM-X-71574 By: Evans, et al.	<u>SOME COMPARISONS OF THE FLOW CHARACTERISTICS OF A TURBONFAN COMPRESSOR SYSTEM WITH AND WITHOUT INLET PRESSURE DISTORTION.</u> Contains measured effects of a circumferential distortion in inlet total pressure on the fan, low, and high compressor of an after burning turbofan engine.
18	NASA CR-2414 By: Marshall	<u>SEPARATED FLOW OVER BODIES OF REVOLUTION USING AN UNSTEADY DISCRETE-VORTICITY CROSS WAKE.</u> Simulates the unsteady cross flow by steady separated flow. A wake description of discrete point vortices arising from a separation of shear layers at the surface as they convect and diffuse downstream. Includes comparisons with experimental data.
19	NASA TN-D-7839 By: Cole	<u>ANALYSIS OF THE DYNAMIC PRESSURE OF A SUPERSONIC INLET TO FLOW PERTURBATIONS UPSTREAM TO THE NORMAL SHOCK.</u> Investigates the effect of normal shock position on supersonic inlet dynamics by linearized mathematical analysis. Shows the comparisons on a frequency response basis with a method-of-characteristics solution. Results are compared with experimental data.
20	NASA TM-X-3264 By: DeBodgan	<u>EFFECT OF A 180-DEG-EXTENT INLET PRESSURE DISTORTION ON THE INTERNAL FLOW CONDITIONS OF A TF30-P-3 ENGINE.</u> Discusses the measured effects of inlet pressure distortion on the internal flow temperature and pressure of TF30-P-3 afterburning turbofan engine. Steady-state flow distortion factor is given.

Table 3.3 CONTINUED

NO.	REFERENCE	TITLE AND REMARKS
21	NASA CR-2686 By: King	<u>ANALYSIS OF DISTORTION DATA FROM TF30-P-3 MIXED COMPRESSION INLET TEST (Final Report).</u> Steady-state and dynamic distortion of a mixed compression inlet with a TF30-P-3 engine at Mach 2.5. Shows the effect on compressor stall and estimates peak instantaneous distortion. Contains experimental data.
22	NASA TM-X-73438 By: Burstadt	<u>A METHOD TO ACCOUNT FOR VARIATION OF AVERAGE COMPRESSOR INLET PRESSURE DURING INSTANTANEOUS DISTORTION ANALYSIS.</u> Instantaneous distortion analysis. Surge margin for inlet-engine compatibility. Contains analytical results only.
23	NASA CR-134996 By: Mazzawy	<u>MODELING AND ANALYSIS OF THE TF30-P-3 COMPRESSOR SYSTEM WITH INLET PRESSURE DISTORTION.</u> Investigates the effect of generated circumferential inlet flow distortion on a TF30-P-3 engine stall with comparisons between model calculations and test data. Contains steady-state effects only.
24	ASME Paper 81-GT-211 By: Ferrand, et al.	<u>THEORETICAL STUDY OF FLOW INSTABILITIES AND INLET DISTORTIONS IN AXIAL COMPRESSORS.</u> Describes a method for evaluating single and multistage compressors and their responses to steady and unsteady inlet distortions. Uses Laplace's transformation for time dependent variables. Includes mach number effects.
25	Journal of Fluid Mechanics By: Goldstein, et al.	<u>THE EFFECT OF FINITE TURBULENCE BY A CONTRACTING STREAM.</u> The turbulence downstream of a rapid contraction is calculated for the case when the turbulence scale can have the same magnitude as the mean-flow spatial scale. Discusses boundary layer separation and turbulent flow.

Table 3.3 CONTINUED

NO.	REFERENCE	TITLE AND REMARKS
26	AIAA Paper 80-1109 By: Stevens, et al.	<u>EVALUATION OF A STATISTICAL METHOD FOR DETERMINING PEAK INLET FLOW DISTORTION USING F-15 AND F-18 DATA.</u> Methods have been developed for significantly reducing the cost of determining inlet peak dynamic distortion values for advanced design purposes. Provides test data of F-15 aircraft and F-18 aircraft. Uses statistical analysis.
27	AIAA Paper 80-1108 By: Marous et al.	<u>AN ANALOG EDITING SYSTEM FOR INLET DYNAMIC FLOW DISTORTION, DYNADEC -- PAST, PRESENT AND FUTURE.</u> An analog/digital (hybrid) editing system DYNADEC (Dynamic Data Editing and 80-Computing) used to screen inlet dynamic pressure distortion data is described.
28	Journal of Mechanical Engr. Science Vol. 22, Feb. 1980 By: Greitzer, et al.	<u>GENERATION OF STREAMWISE VORTICITY IN AN ASYMETRIC SWIRLING FLOW.</u> The behavior of a circumferentially nonuniform swirling flow, which is of interest in connection with the problem of the response of axial compressors to inlet flow distortion, is examined. Uses the model of oscillating 3-D flow.
29	AIAA Paper 80-1110 By: Sanders	<u>AN EVALUATION OF STATISTICAL METHODS FOR THE PREDICTION OF MAXIMUM TIME-VARIANT INLET TOTAL PRESSURE DISTORTION.</u> The paper presents an evaluation of statistical methods for the prediction of maximum time-variant inlet total pressure distortion. Includes Motyeka's method, Melick's method and Jacock's method.
30	AIAA Paper 80-0386 By: Neumann et al.	<u>AN ANALYTICAL AND EXPERIMENTAL STUDY OF A SHORT S-SHAPED SUBSONIC DIFFUSER OF A SUPERSONIC INLET.</u> An experimental investigation of a subscale HiMat forebody and inlet was conducted over a range of Mach numbers to 1.4.

Table 3.3 CONTINUED

NO.	REFERENCE	TITLE AND REMARKS
31	West Germany Paper By: Elcrat, et al.	<u>A SPATIAL DECAY ESTIMATE FOR THE NAVIER-STOKES EQUATION.</u> An Expression is derived which gives the exponential decay, in the distance from fixed reference plane, for an energy-type functional of solutions using the Navier-Stokes approach. Uses channel flow model.
32	AIAA Paper 79-0104 By: Stevens, et al.	<u>REYNOLDS NUMBER, SCALE AND FREQUENCY CONTENT EFFECTS ON F-15 INLET INSTANTANEOUS DISTORTION.</u> An inlet instantaneous distortion study program sponsored by NASA was recently completed using an F-15 fighter aircraft. Uses bandpass filters. Includes Reynolds number effects.
33	ASME Paper By: Inoue, et al.	<u>SMALL PERTURBATION ANALYSIS OF NONUNIFORM ROTATING DISTURBANCES IN A VANELESS DIFFUSER.</u> The behavior of the distorted flow discharged from a centrifugal impeller within a vaneless diffuser is examined theoretically by assuming small disturbances to a main flow.
34	AIAA Paper 78-970 By: Melick, et al.	<u>ESTIMATING MAXIMUM INSTANTANEOUS DISTORTION FROM INLET TOTAL PRESSURE RMS MEASUREMENTS.</u> A new mathematical model of inlet turbulence is developed by application of basic fluid dynamics and statistical concepts. Contains turbulent flow pressure measurements.
35	AIAA Paper 77-876 By: Brilliant et al.	<u>COMPARISON OF ESTIMATED WITH MEASURED MAXIMUM INSTANTANEOUS DISTORTION USING FLIGHT DATA FROM AN AXISYMMETRIC MIXED COMPRESSION INLET -- FOR YF-12C AIRCRAFT.</u> YF-12C flight-measured inlet dynamic distortion data are compared with predictions made on the basis of the method reported by Melick et al. (1976).

Table 3.3 CONTINUED

NO.	REFERENCE	TITLE AND REMARKS
36	West Germany Paper By: Ogorodnikov, et al.	<u>INVESTIGATION OF THE FLOW PATTERN AT THE ENGINE FACE AND METHODS OF THE FLOW PATTERN SIMULATION AT SUPERSONIC FLIGHT SPEED.</u> Steady-state distortions and fluctuations of a nonuniform time-dependent fluctuating flow field at an aircraft engine face at supersonic flight speed significantly affect the engine operational stability.
37	AIAA Paper 76-705 By: Motycka	<u>DETERMINATION OF MAXIMUM EXPECTED INSTANTANEOUS DISTORTION PATTERNS FROM STATISTICAL PROPERTIES OF INLET PRESSURE DATA.</u> An inexpensive and time-saving procedure is proposed which uses random numbers to synthesize instantaneous inlet distortion in turbine engines from statistical properties of inlet pressure data.
38	AIAA Paper 75-59 By: Ting, et al.	<u>DESIGN AND TESTING OF NEW CENTER INLET AND S-DUCT FOR B-727 AIRPLANE WITH REFANNED JT8D ENGINES.</u> The work described in this paper was part of the NASA refan program. Contains subjects of aircraft engines, Boeing 727 aircraft, ducted flow, engine design and engine & inlet tests.
39	AIAA Paper 74-1197 By: Clark	<u>STATISTICAL AVERAGES OF SUBSONIC INLET DISTORTION.</u> Results of an experimental investigation of dynamic distortion in a typical subsonic aircraft inlets are discussed. Includes the effects of aerodynamic interference, flow distortion and pressure gradient.
40	AFFDL-TR- 70-20 By: Kutschen- reuter et al.	<u>LARGE SCALE INLET DISTORTION INVESTIGATION.</u> Describes dynamic characteristics of engine inlets. Inlet pressure, pressure recovery and flow distortion are discussed. Includes performance tests and steady-state effect.

Table 3.3 CONTINUED

NO.	REFERENCE	TITLE AND REMARKS
41	AAAF-NT-80-18 By: Farcy, et al.	<u>HIGHLY SEPARATED TRANSONIC FLOW -- AND FLOW VISUALIZATION TECHNIQUES.</u> A highly separated transonic flow was studied under laboratory conditions in order to show the usefulness of combining various flow visualization and analysis techniques in defining jet engine intake flow characteristics.
42	AD-A089817 By: Sanders, et al.	<u>AN EVALUATION OF STATISTICAL METHODS FOR THE PREDICTION OF MAXIMUM TIME-VARIANT INLET TOTAL PRESSURE DISTORTION.</u> An analysis was conducted to determine the accuracies and limitations of three statistical methods used to predict engine-face maximum time-variant total pressure distortion. Includes Mach no. effect. Contains four different inlet models.
43	Von Karman Inst. of Fluid Dyn. Unsteady Flow in Turbomachines By: Peacock	<u>UNSTEADY PRESSURE DISTORTION -- IN COMPRESSOR FLOW.</u> A model is proposed for the solution of unsteady flows in a compressor embedded in ductwork and subject to repetitive or nonrepetitive pulses.
44	Ph.D. Thesis North Carolina State University By: Hardin	<u>AN EXPERIMENTAL STUDY OF THE RESPONSE OF A TURBOMACHINE ROTOR TO A LOW FREQUENCY INLET DISTORTION.</u> An experiment was conducted to measure the response of an isolated turbomachine rotor to a distortion in inlet axial velocity.
45	MBB-UFE-1359-0 By: Lotter	<u>AERODYNAMIC PROBLEMS IN ENGINE AIRFRAME INTEGRATION ON FIGHTER AIRPLANES.</u> Different types of intake are discussed together with engine mass flow/air intake matching problems. Discusses aerodynamic interference. Uses the case of fighter aircraft.

Table 3.3 CONTINUED

NO.	REFERENCE	TITLE AND REMARKS
46	AD-A064776 By: Hardi	<u>AN EXPERIMENTAL STUDY OF THE RESPONSE OF A TURBO-MACHINE ROTOR TO A LOW FREQUENCY INLET DISTORTION.</u> An experiment was conducted to measure the response of an isolated turbomachine rotor to a distortion in inlet axial velocity. Uses a once-per-revolution sinusoidal axial velocity variation.
47	AD-A059437 By: Sajben, et al.	This is part of a continuing investigation of unsteady transonic diffuser flows. Discusses nonuniform flow characteristics in vaneless diffusers.
48	NASA-CR-2766 Koch, et al.	<u>ANALYSIS OF PRESSURE DISTORTION TESTING.</u> The development of a distortion methodology, method D, was documented, and its application to steady state and unsteady data was demonstrated.
49	AGARD Unsteady Phenomena in Turbomachinery By: Mokolko	<u>THE EFFECT OF TURBULENT MIXING ON THE DECAY OF SINUSOIDAL INLET DISTORTIONS IN AXIAL FLOW COMPRESSORS.</u> A small perturbation actuator disc theory is presented for the prediction of the decay of sinusoidal flow distortions in high hub tip ratio axial compressors with steady, circumferential inlet maldistribution.
50	AGARD Unsteady Phenomena in Turbomachinery By: Freeman	<u>THE RELATIONSHIP BETWEEN STEADY AND UNSTEADY SPECIAL DISTORTION -- IN TURBOCOMPRESSOR INTAKE FLOW.</u> Simple theories of turbulence are used to develop a model that relates the fluctuating spacial distortion to the time average spacial distortion.
51	Paper from Pennsylvania State University By: Bruce	<u>AXIAL FLOW ROTOR UNSTEADY RESPONSE TO CIRCUMFERENTIAL INFLOW DISTORTIONS.</u> The unsteady response of an axial flow fan rotor to steady, circumferential inflow velocity and stagnation pressure distortions is assessed by two different methods.

Table 3.3 CONTINUED

NO.	REFERENCE	TITLE AND REMARKS
52	AGARD Unsteady Phenomena in Turbomachinery By: Hanson	<u>APPLICATION OF ROTOR MOUNTED PRESSURE TRANSDUCERS TO ANALYSIS OF INLET TURBULENCE -- FLOW DISTORTION IN TURBOFAN ENGINE INLET.</u> Miniature pressure transducers installed near the leading edge of a fan blade were used to diagnose the non-uniform flow entering a subsonic tip speed turbofan on a static test stand.
53	AGARD-CP-177	<u>UNSTEADY PHENOMENA IN TURBOMACHINERY.</u> Turbomachinery unsteady aerodynamics are reviewed with emphasis on flow distortion phenomena inside subsonic, transonic and supersonic axial flow compressor stages.
54	NASA-TM-X-3246 E-8213 By: Mitchell	<u>EFFECT OF INLET INGESTION OF A WING TIP VORTEX ON COMPRESSOR FACE FLOW AND TURBOJET STALL MARGIN.</u> A two-dimensional inlet was alternately mated to a coldpipe plug assembly and a J85-GE-13 turbojet engine, and placed in a Mach 0.4 stream so as to ingest the tip vortex of a forward mounted wing.
55	NASA-TM-X-3169 By: Costakis	<u>EXPERIMENTAL INVESTIGATION OF A SIMPLE DISTORTION INDEX UTILIZING STEADY-STATE AND DYNAMIC DISTORTIONS IN A MACH 2.5 MIXED-COMPRESSION INLET AND TURBOFAN ENGINE.</u> A wind tunnel investigation was conducted to determine the amplitude and spatial distribution of steady-state and dynamic distortion produced in an inlet. Formulates a simple index that combines steady-state and dynamic distortions.
56	In AIAA Colloquium By: Kimzey	<u>AN ANALYSIS OF THE INFLUENCE OF UNSTEADY CASCADED AIRFOIL BEHAVIOR ON AXIAL FLOW COMPRESSORS WITH UNSTEADY AND DISTORTED INFLOW.</u> The influence of unsteady and circumferentially distorted flow on the stall of axial flow compressors. Reassesses the quasi-steady-state assumption. Compare to experimental results.

Table 3.3 CONTINUED

NO.	REFERENCE	TITLE AND REMARKS
57	NASA-TM-X-68189 By: Calogeras	<u>INSTANTANEOUS DISTORTION INVESTIGATION.</u> Review some of the results obtained in an inlet-engine compatibility test run in the 10 x 10 supersonic wind tunnel of the NASA-Lewis Research Center.
58	AIAA Paper 72-1099 By: Ellis, et al.	<u>A PROCEDURE FOR ESTIMATING MAXIMUM TIME-VARIANT DISTORTION LEVELS WITH LIMITED INSTRUMENTATION.</u> Inlet data measured with complete high-response instrumentation have been used to establish a new procedure for assessing propulsion system flow stability from tests with limited instrumentation.
59	NASA-TM-X-2081 By: McAulay	<u>EFFECT OF DYNAMIC VARIATIONS IN ENGINE-INLET PRESSURE ON THE COMPRESSOR SYSTEM OF A TWIN-SPOOL TURBOFAN ENGINE.</u> An investigation was conducted to determine the effects of dynamic inlet pressure variations on the compressor system of a turbofan engine.
60	NASA-TM-X-1946 By: Meyer	<u>TECHNIQUE FOR INDUCING CONTROLLED STEADY-STATE AND DYNAMIC INLET PRESSURE DISTURBANCES FOR JET ENGINE TESTS.</u> An investigation was conducted to evaluate a technique wherein secondary air was injected through an array of small nozzles uniformly distributed in an engine inlet duct. Creates steady-state and dynamic distortion thru the control of the secondary-air distribution and flow rate.
61	NASA-TM-X-1928 By: Wenzel	<u>EXPERIMENTAL INVESTIGATION OF THE EFFECTS OF PULSE DISTORTIONS IMPOSED ON THE INLET OF A TURBOFAN ENGINE.</u> A YFT-30-P-1 turbofan was operated in an altitude chamber. Uses distortion device for the test of engine stall sensitivity as a function of pulse duration. Distortion sector angle was mapped for reference. Transient recordings during stall are presented.

Table 3.3 CONTINUED

NO.	REFERENCE	TITLE AND REMARKS
62	NASA-TM-X-2993 By: Costakis	<u>ANALOG COMPUTER IMPLEMENTATION OF FOUR INSTANTANEOUS DISTORTION INDICES.</u> Test of the compatibility of a J85-GE-13 engine and an axisymmetric mixed-compression inlet is presented in this report.
63	NASA-TM-X-52595 By: Coltrin, et al.	<u>SUPERSONIC WIND TUNNEL INVESTIGATION OF INLET-ENGINE COMPATIBILITY.</u> Results are presented from an experimental investigation in the Lewis 10- by 10-Foot Supersonic Wind Tunnel of inlet-produced dynamic distortion and its effects on the stall margin of a J-85 turbojet engine.
64	NASA-TM-X-1842 By: Calogeras	<u>EXPERIMENTAL INVESTIGATION OF DYNAMIC DISTORTION IN A MACH 2.50 INLET WITH 60 PERCENT INTERNAL CONTRACTION AND ITS EFFECT ON TURBOJET STALL MARGIN.</u> Wind tunnel investigation to determine the amplitude and spatial distribution of dynamic distortion produced in an inlet with 60 percent of the overall supersonic area contraction occurring internally. Discusses compressor stall margin.
65	AIAA Paper 68-649 By: Kostin, et al.	<u>APPLICATION OF STATISTICAL PARAMETERS IN DEFINING INLET AIRFLOW DYNAMICS.</u> Presents wind tunnel and flight test data taken during tests of several inlet duct designs. Uses auto-correlation and cross-correlation, variance, and power spectral density for statistical analysis.

Table 3.3 CONTINUED

4. CONCLUDING REMARKS

Considerable research has been conducted in the past fifteen years on inlet dynamic distortion primarily because of problems or potential problems that were encountered on aircraft being developed or studied in that time period. Less attention is being given to this subject today so that a new generation of investigators will be stepping into breach when it again becomes of greater concern at some time in the future. It is important that a cohesive body of knowledge exist so that there will be a ready reference to past work and experience. This literature survey is a first step in that direction.

This report presents the results of a literature survey on the subject of inlet dynamic distortion on conventional high performance aircraft, its measurement, analysis and prediction. Sixty-five documents were found on both experimental and theoretical subjects. No doubt some unavoidably have been missed. Most of those found deal with experimental results and their analysis and comparisons. Few deal with the prediction of peak distortion levels and none deal with the a priori prediction of dynamic distortion, that is, all maximum distortion prediction techniques utilize experimental data as the basis for extreme value prediction. The works of Melick, Jacocks and Motycka were found to be most relevant to the prediction of dynamic distortion.

An attempt has been made to present the material in such a way that an investigator could readily determine the reports of most interest before ordering them from the library or NTIS. The various topics have been referenced, cross-referenced, and synopsisized. Finally, summaries and conclusions from each report are contained in Appendix C.

It is hoped that this will facilitate and expedite the work of current and future investigators of dynamic distortion and related problems.

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APPENDIX C. LIST OF REPORT SUMMARIES AND CONCLUSIONS

This section presents copies of summaries and conclusions taken from References 1 through 65. It is hoped that this information will assist the user in determining the content and its application to a particular problem (i.e. the influence of a particular variable on the inlet dynamic distortion or compressor stability). In a few instances, more extensive excerpts are included to give a more comprehensive picture of the content of the report.

All copies of the summaries and conclusions presented here are cross-referenced in Appendix B.

ESTIMATING MAXIMUM INSTANTANEOUS DISTORTION FROM
INLET TOTAL PRESSURE RMS AND PSD MEASUREMENTS

H. Clyde Melick, Andres H. Ybarra,
Daniel P. Bencze[†]

Abstract

The purpose of this study is to develop an inexpensive procedure to determine the extreme values of instantaneous inlet distortion on a statistical basis and to provide insight into the basic mechanics of unsteady inlet flow. This analysis is based on fundamental fluid dynamics and statistical methods and quantitatively relates the measured root mean square (rms) level and power spectral density (PSD) function of the time-dependent total pressure fluctuations to the strength and size of the instantaneous low-pressure regions at the inlet/engine interface. The extreme values of these instantaneous low-pressure regions are analyzed in combination with the steady-state distortion to determine the most probable maximum instantaneous distortion. Results of the analysis show that this maximum instantaneous distortion is dependent upon the steady-state distortion, the rms level and PSD function of the inlet turbulence, and the length of record analyzed. Predictions based on this analytical technique are presented and compared with the maximum instantaneous distortion computed from the simultaneous readings obtained from a 40-probe rake employing high-response total pressure instrumentation. These comparisons show exceptionally good agreement between the measured and predicted results, substantially verifying this approach.

SUMMARY AND CONCLUSIONS

An analytical technique has been developed in this study to predict the maximum instantaneous inlet distortion from the easily measured rms level and PSD function of the inlet total pressure fluctuations. The method is based on a mathematical model of turbulent flow, described by a random distribution of discrete vortices being convected downstream by the mean flow. Fundamental fluid dynamics were used to define the flow properties of the discrete vortices, and statistical methods were applied to the random properties describing the vortices to relate them to the rms level and PSD function of the total pressure fluctuations. The momentary low-pressure region created by the passage of each discrete vortex was described in terms of a representative distortion factor; the statistical characteristics of this distortion factor were formulated and related to the rms and PSD total pressure measurements. The extreme values of the statistical characteristics were analyzed to establish the most probable maximum distortion level.

This most probable maximum distortion level is defined to be the maximum instantaneous distortion expected under the particular operating conditions and was shown to be dependent upon the rms level and PSD function of the total pressure fluctuations, the steady-state distortion level, the length of time at the operating conditions, and the frequency response characteristics of the engine. Comparisons between results of this analysis and conventionally determined values of maximum instantaneous distortion showed excellent agreement. The comparisons were made using data from two independent inlet tests encompassing three inlet designs and a range of Mach numbers, mass-flow ratios, and simulated engine response characteristics. In addition, by computing the results using different numbers of probes, it was determined that only a limited number of probes are required to obtain results comparable to the average of a typical 40-probe rake. The average of 20 probes yielded essentially the same results as the average of 40 probes, while errors of less than 10 percent were experienced when the results were based on only 4 probes.

The excellent agreement between the measured and predicted results provides substantial verification of this approach, although further verification over a broader range of operating conditions is required and recommended. However, even at this level of verification, this approach has considerable potential in the analysis of all scale model inlet testing where size and/or expense precludes the use and detailed analysis of a large quantity of high-response pressure data to identify the maximum instantaneous distortion.

This statistical approach also appears directly applicable to the evaluation of all dynamic data, including full-scale inlet/engine testing, for the following reasons. First, the identification of the maximum instantaneous distortion must be in terms of both the length of time of expected operation under the condition examined and the required level of probable occurrence. These can easily be taken into account using this approach. Second, based on the results of the data analysis comparisons, the accuracy of this method is comparable to that of the conventional method of defining the maximum instantaneous distortion. Third, the necessary equipment for the acquisition and processing of the data required for this statistical approach is readily available and inexpensive to operate, whereas the cost to acquire and process data following the conventional procedure may exceed the cost for the entire design, fabrication, and testing of the inlet model to satisfy the steady-state objectives.

NASA CR-114577

ANALYSIS OF INLET FLOW
DISTORTION AND TURBULENCE EFFECTS
ON COMPRESSOR STABILITY

By

H. C. MELICK

31 March 1973

Technical Report No. 2-57110/3R-3071

NASA CR-114577

SUMMARY

The effect of steady state circumferential total pressure distortion on the loss in compressor stall pressure ratio has been established by analytical techniques. Full scale engine and compressor/fan component test data were used to provide direct evaluation of the analysis. Favorable results of the comparison are considered verification of the fundamental hypothesis of this study. Specifically, since a circumferential total pressure distortion in an inlet system will result in unsteady flow in the coordinate system of the rotor blades, an analysis of this type distortion must be performed from an unsteady aerodynamic point of view. By application of the fundamental aerothermodynamic laws to the inlet/compressor system, parameters important in the design of such a system for compatible operation have been identified. A time constant, directly related to the compressor rotor chord, was found to be significant, indicating compressor sensitivity to circumferential distortion is directly dependent on the rotor chord.

As an initial step in the investigation of the effects of time dependent total pressure distortion on the compressor stability characteristics, an analytical model of turbulent flow typical of that found in aircraft inlets has also been developed. Due to the non-deterministic (random) nature of this type of flow distortion, the flow analysis requires use of statistical methods. These methods were combined with basic fluid dynamic concepts to provide a usable analysis technique. With this model, the power spectral density function and root mean square level of the time dependent total pressure take on considerable significance as indicators of the strength and extent of low pressure regions that are important in the compressor reaction to inlet flow disturbances. Spectra obtained from the model were compared with those obtained in tests of a Mach 3 mixed compression inlet to illustrate the technique of determining the mean size and strength of instantaneous low pressure regions by statistical techniques and to verify the turbulent flow model. Excellent agreement was obtained in the comparison verifying this fundamental approach.

Both the steady state distortion/compressor analysis and the turbulent flow model are considered developed to the point necessary to initiate the development program to achieve the long term program objective of combining these results to establish a fundamental relationship between both inlet steady state circumferential distortion and turbulence and loss in compressor stall margin.

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CONCLUSIONS AND RECOMMENDATIONS

The analytical developments reported herein provide a fundamental approach to the problem of inlet/engine compatibility. With further development, this approach will provide a method of evaluating inlet tests and engine designs early in the propulsion system development cycle. Ultimately, it shows promise as a method for predicting and evaluating the effects of distortion and turbulence on engine stall characteristics, prior to system test.

The following are the more significant conclusions arising from the work to date and suggestions for continued activity to achieve the basic program goal: establishing the fundamental relationship between inlet distortion and turbulence and loss in compressor stall margin.

Conclusions

(1) The effect of circumferential total pressure distortion on the loss in compressor stall margin has been established analytically. The analysis has shown that the stall margin loss is directly a function of the distortion pattern, the distortion level $((\bar{P}_t - P_{t_{min}})/\bar{P}_t)$, and of the compressor rotor reduced frequency, $k = \omega c/2U$.

(2) The rotor chord is the principal design variable in the reduced frequency, k , and therefore emerges as a significant engine parameter in design for compatibility.

(3) Favorable comparison of distortion and engine stall data with analysis results is considered verification of the fundamental hypothesis of the analysis. Specifically, a circumferential total pressure distortion will result in an unsteady flow over the rotor blades requiring these unsteady aerodynamic effects to be included in the stage characteristics.

(4) The accuracy of the stall prediction technique is sufficient to justify the simplified approach which considers an overall compressor work balance rather than a detailed stage-by-stage development.

(5) A phenomenological model of turbulent flow typical of that found in aircraft inlets has been developed by combining statistical techniques with the basic laws governing fluid dynamics. The power spectral density function and root mean square level of the fluctuating total pressure take on considerable significance as a consequence of the model resulting in a means of determining the strength and extent of time variant low pressure regions.

(6) Favorable comparison of spectra obtained from the analytical model with test data of a Mach 3 mixed compression inlet verify the Turbulent Flow Model.

(7) The agreement with test data for both the Compressor Analysis and Turbulent Flow Model strongly suggest that compatibility problems, heretofore only attacked by empirical methods, are amenable to analysis.

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Recommendations

(1) Both the Compressor Analysis and Turbulent Flow Model are considered developed to the point necessary to initiate the program to achieve the long term goal of establishing a fundamental relationship between both inlet distortion and turbulence and compressor stall margin loss.

(2) Further comparisons of the compressor analysis with test data are recommended for refining the method. The data used should provide the detailed distortion patterns, compressor geometry and compressor operating conditions.

(3) Additional analysis of turbulence data from a well documented inlet test program should be conducted to demonstrate the use of the turbulent flow model in isolating the source of turbulence and establishing turbulence decay characteristics.

(4) Finally, the developed relationships between distortion, turbulence and loss in compressor stall margin should be compared with data from an aircraft flight test program to verify the analysis.

TR No. 2-5711/3R-3209

ESTIMATING MAXIMUM INSTANTANEOUS
DISTORTION FROM INLET TOTAL
PRESSURE RMS & PSD
MEASUREMENTS

By

H. C. Melick

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March 1975

TR No. 2-5711/3R-3209

SUMMARY

Present methods of estimating the maximum instantaneous inlet distortion are highly complex, costly, and fail to provide insight into the actions governing the flow. These methods also attempt to characterize a random phenomena by use of deterministic descriptions. To date, their accuracy has not been verified. The purpose of this study is to develop an inexpensive alternative procedure to statistically evaluate the extreme values of instantaneous inlet distortion and provide insight into the basic mechanics of unsteady inlet flow and engine reaction.

This development is based on fundamental fluid dynamics to provide an understanding of the turbulent inlet flow and to quantitatively relate the root mean square (RMS) level and power spectral density (PSD) function to the strength and size of the low pressure regions. The most probable extreme value of the instantaneous low pressure region is then synthesized from this information and from the steady state distortion data to obtain the maximum instantaneous distortion level. Results of the analysis show the extreme values to be dependent upon the steady state distortion, the turbulence RMS level and PSD function, the time on point, and the selected confidence level. Analytical projections are presented and compared with data obtained by a highly correlated 40 probe instantaneous pressure measurement system. Results of this comparison show exceptionally good agreement, without recourse to any empirical adjustment. Consequently, the developed analysis provides both an understanding and quantitative description of the flow. The math model is considered a new contribution to the state-of-the-art and in practical application leads to data acquisition and analysis requirements that are an order of magnitude lower in cost and complexity than the present approach to measuring the extreme values of instantaneous inlet distortion. With this development, it is now practical to account for unsteady distortion in all scale model inlet test programs since the method requires only a limited amount of inexpensively obtained RMS and PSD data and a simple analysis procedure.

TR No. 2-5711/3R-3209

CONCLUDING REMARKS

The developed technique to predict the maximum instantaneous distortion factors from the RMS and PSD measurements of the inlet total pressure fluctuations is based on basic principles of both fluid dynamics and statistical mathematics and is without any empirical adjustment or curve fitting. The exceptionally close agreement found in the data/analysis comparisons provides considerable verification.

The success of the developed procedure to date, appears to warrant its use in all scale model testing where size and expense precludes use of a full complement of 40 high response, highly correlated pressure probes and the requisite data acquisition/reduction systems.

It is believed that a strong case can also be made for use of the statistical procedure in the evaluation of all dynamic data, including full scale inlet/engine testing. This case rests on (a) the necessity of statistical treatment of random phenomena (b) accuracy and (c) cost. Specifically: (a) Turbulent inlet flow is random. Consequently, identification of the maximum instantaneous distortion must be in terms of both the time on point and in terms of required confidence level. This can be done with the developed technique. The instantaneous method, on the other hand, treats the distortion as an extension of steady state evaluations and ignores both time and confidence levels. This, in itself, causes misinterpretation and, as established in this study, errors on the order of 10 to 40%. (b) As a result of the data/analysis comparisons conducted in this study the accuracy using the developed method appears on a par with the highly sophisticated instantaneous methods. (c) The equipment for acquisition and reduction (excluding transducers) of data required for the statistical procedures is readily available; the cost to acquire and reduce data via the instantaneous method may exceed the costs for the entire design, fabrication and test of the inlet model required to meet steady state test objectives.

Realistic assessment of the accuracy requirements of inlet dynamic distortion measurements is necessary. This assessment needs to be made in perspective with such items as the accuracy of the quoted engine

stall pressure ratio sensitivity, the available stall pressure ratio, and the engine frequency response to unsteady flow. Therefore, it is recommended that verification of this statistical technique be continued; that an assessment of accuracy of the instantaneous method be established; and last, the accuracy requirements and risk involved with dynamic distortion be evaluated to obtain a realistic perspective on this aspect of inlet/engine compatibility.

SUMMARY OF RESULTS

The purpose of this study was to develop an inexpensive analytical technique to evaluate the maximum instantaneous inlet distortion factor from the inexpensive, easily secured root mean square (RMS) level and power spectral density (PSD) function of the inlet total pressure fluctuations. The method used is to mathematically model the random vortices creating the turbulent flow by solution of the Navier-Stokes Equations of Motion. Statistical methods were then applied to the random properties composing the vortices to relate the total pressure RMS and PSD measurements to the momentary low pressure regions created by the vortices. The distortion factors were then computed for these low pressure regions and in turn related to the RMS and PSD levels. Results of the development are summarized below.

1. A method is developed to inexpensively evaluate the most likely maximum instantaneous level of a distortion factor from inlet Root Mean Square (RMS) level and Power Spectral Density (PSD) functions.
2. Variations about this most probable level of unsteady distortion can be established for specified levels of confidence.
3. The technique can be used for a variety of distortion factors including the loss in stall pressure ratio as estimated from the compressor analysis developed in Reference 2.
4. Results are shown to be dependent on the measured RMS and PSD, the steady state distortion factor, the time on point, the engine response frequency, and the desired/required confidence level.

5. Exceptionally good results of the data/analysis comparisons have verified the development technique. These included tests of three different inlets at a variety of flight conditions and mass flows.

6. As established from analysis of a limited number of points, errors of less than 10% are accrued using a complement of only 8 high response probes.

7. A procedure is provided for application of the technique to inlet data.

TR No. 2-57110/5R-3210

COMPUTER PROGRAM DOCUMENTATION-----

ESTIMATING MAXIMUM INSTANTANEOUS

DISTORTION FROM INLET TOTAL

PRESSURE RMS & PSD

MEASUREMENTS (Volume I & II)

By

A. H. Ybarra

H. C. Melick, Jr.

March 1975

I. ABSTRACT

1. Program No. (ABS) 2-57110/5R-3210
2. Date 31 March 1975
3. By Andres H. Ybarra
Phone (214) 266-2254
4. Routine Name and S and P Classification (U) Definition of Mean Size and Strength of the Vortices Creating Inlet Turbulent Flow
5. Routine Number: LTV Designation D1805 Other Designation _____
6. Originator: Name Andres H. Ybarra Extension 2254
Unit Number 2-53392 Other H. C. Melick
7. Programming Language: FORTRAN IV
8. Machine and Configuration: Make California Computer Company (CDC),
Model 6600 Special Feature CalComp Plotter
9. Special Requirements: Operational None Special File _____
10. Running Time: Typical 2-4 Sec per case
11. Limitations: _____
12. Broad Functional Description: The power spectral density function of the unsteady total pressure fluctuations is used as the primary input to the program (ΔP_T^2 versus frequency). Several different analytical PSD functions are generated by the program utilizing the normalized spectrum from the turbulent flow model and several different estimates of the mean vortex core size (\bar{a}), the normalizing parameter. The resulting spectra are then compared with the input spectrum until an acceptable match is found. The vortex core size producing the best match between the two spectra is then considered to be the mean size of the vortices composing the turbulent flow. Both spectra are then plotted so visual inspection of the match can be made. The program

TR No. 2-57110/5R-3210

also integrates the PSD function to obtain the root mean square level of the turbulence. Consequently the two primary results of the program are the mean vortex core size (\bar{a}) and strength, (ΔP_{TRMS}) of the vortices which compose the unsteady, turbulent inlet flow.

SAMPLE

PROBE NO. 1

ETA= 0.600

M= 2.5

U₀= 200.2 M/S

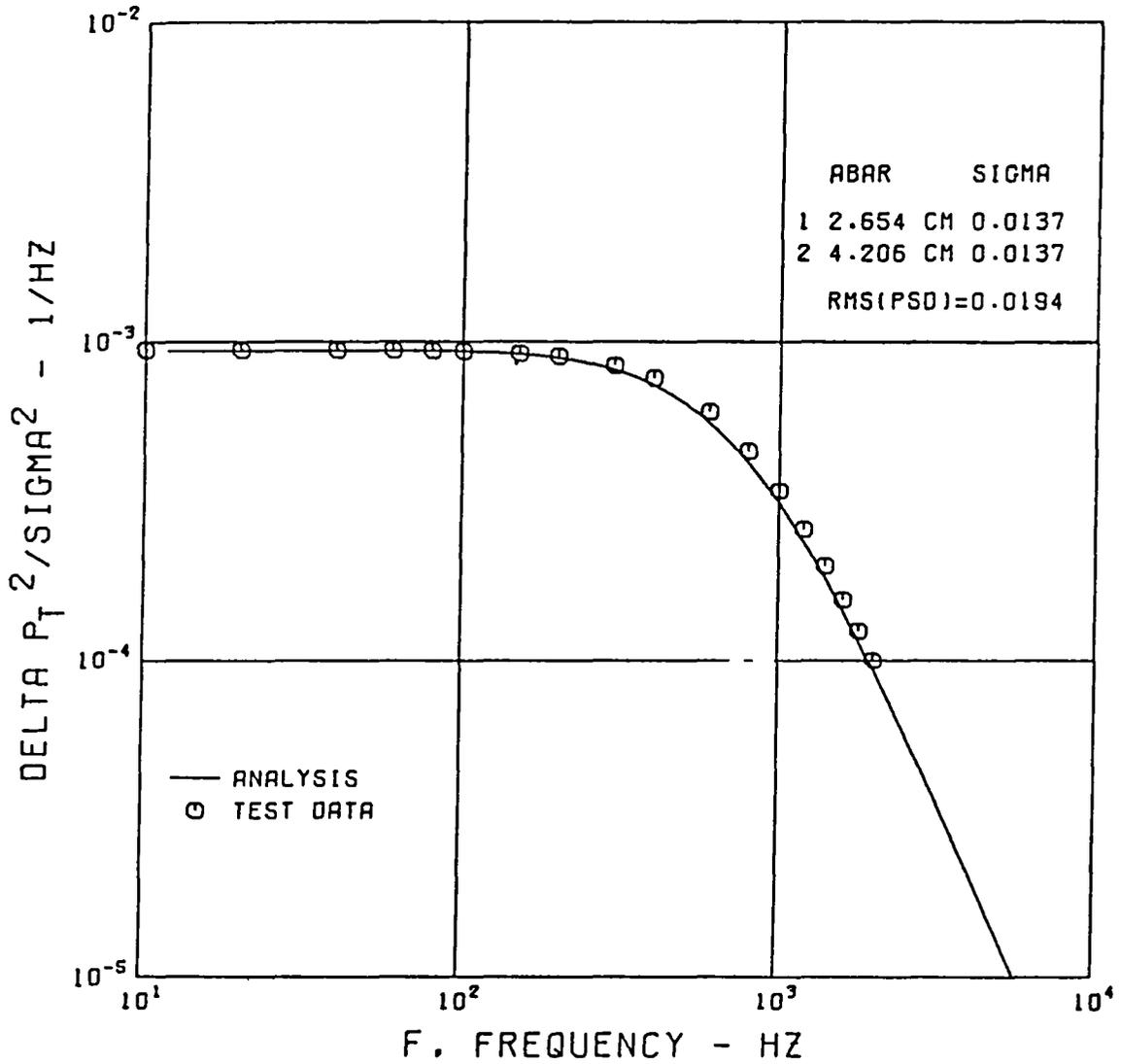


Figure 9. Typical CalComp Plotter PSD Graph.

TR No. 2-57110/5R-3210

I. ABSTRACT

1. Program No. (ABS) 2-57110/5R-3210 13. Rev _____
2. Date 31 March 1975 14. Rev. Date _____
3. By A. H. Ybarra 15. Grp. Aprvl _____ Date _____
Ext. (214) 266-2254 16. Status _____
4. Routine Name and S and P Classification - Estimating the Maximum
Instantaneous Distortion Program (U)
5. Routine Number: LTV Designation D 1792 Other _____
6. Originator: Name A. H. Ybarra Extension 2254
Unit Number 2-53392 Other H. C. Melick
7. Programming Language: FORTRAN IV
8. Machine and Configuration: Make CDC Model 6600
Other None Special Features None
9. Special Requirements None
10. Running Time 2 - 3 seconds
11. Limitations _____
12. Broad Functional Description: This program computes the most probable maximum instantaneous inlet distortion factors from the mean vortex core size (\bar{a}) and strength (ΔP_{TRMS}) as determined from high response inlet instrumentation in conjunction with the computer program described in Volume I of this document. The program computes the instantaneous distortion factor for each inlet operating condition and as a function of the time period of interest and assumed engine frequency characteristics. The distortion factors now computed are K_{D2} , K_Q , IDC, IDR, K_{RA} , K_{A2} and ΔSPR . Any one or all these can be processed at the same time.

TR No. 2-57110/5R-3210

REFERENCES

1. Ybarra, A. H., "Computer Program Documentation, Estimating Maximum Instantaneous Distortion from Inlet Total Pressure RMS and PSD Measurements, Volume I. Definition of the Mean Size and Strength of the Vortices Creating Inlet Turbulent Flow" (Preliminary), VSD Technical Report No. 2-57110/5R-3210, 31 March 1975.
2. Melick, H. C. "Analysis of Inlet Flow Distortion and Turbulence Effects on Compressor Stability", NASA CR 114577, 31 March 1973.
3. Melick, H. C.; Ybarra, A. H.; "Estimating Maximum Instantaneous Distortion Factors from Inlet RMS and PSD Measurements: Vought Systems Division of LTV Aerospace Corporation Technical Report No. 2-57110/5R-3209, 31 March 1975.

AD/A-004104

AD/A-004 104

STATISTICAL PREDICTION OF MAXIMUM TIME-VARIANT INLET DISTORTION LEVELS

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ABSTRACT

Time-variant inlet pressure data from several aircraft designs are analyzed utilizing tools based on Gumbel's extreme-value statistics with the results illustrating the basic randomness of distortion factors. A probabilistic model of distortion is proposed with three parameters to be evaluated by fitting to inlet data by the method of maximum likelihood. The effects of data acquisition time, frequency bandwidth, and sampling rate are discussed in context with Moore's similarity parameter to indicate scalability of the dynamic inlet distortion data. The end result of these analyses is a recommended procedure that enables the use of a short time segment of distortion data to statistically predict the expected maximum distortion level corresponding to any time period of inlet operation.

6.0 CONCLUDING REMARKS

Application of Gumbel's extreme-value statistics analyses to time-variant inlet data from four aircraft designs has led to the following results and recommendations:

1. The peak instantaneous distortion as observed within a finite data acquisition time period is random and not repeatable as is the engine face pressure pattern for that instant.
2. A short time segment of distortion data can be used to statistically predict the expected maximum distortion level corresponding to any time period of inlet operation including estimates of the prediction tolerance.
3. Engine qualification testing should use screens based on the steady state rather than a peak instantaneous distortion pressure pattern with the distortion being intensified to the expected maximum level corresponding to the aircraft operation time at specific test conditions.
4. Data acquisition time periods during inlet development wind tunnel testing can be reduced from the current 30 seconds to nominally 2 seconds for stationary test conditions provided that analysis of the resulting time-variant distortion is based on extreme-value statistics.

NASA TM X-73118

INVESTIGATION OF TWO BIFUCATED-DUCT INLET SYSTEMS
FROM MACH 0 TO 2.0 OVER A WIDE RANGE OF ANGLES OF ATTACK

Eldon A. Latham

Ames Research Center

SUMMARY

A 15.354-percent-scale lightweight fighter-type inlet-forebody was tested in the Ames Unitary Plan Wind Tunnels over a Mach number range of 0 to 2.0. Model configurations consisted of side-mounted normal shock and fixed overhead ramp-type inlets. Each configuration consisted of two inlets ducted (bifurcated) to supply a single engine face. The normal shock inlet variables included a boundary layer splitter bleed system, alternate boundary-layer splitter plates, alternate upper and lower cowl lip shapes, and a blow-in door (auxiliary inlet) in one lower lip. The only variable of the fixed overhead ramp inlet was the boundary layer bleed flow. Reynolds numbers ranged from 7.6×10^6 to 19.5×10^6 /m (2.5×10^6 to 6.4×10^6 /ft). Angle of attack ranged from -10° to 35° and angle of sideslip from -8° to 8° . Test measurements included engine face total pressure recovery steady-state distortion, dynamic distortion, and surface static pressures on the forebody and inlet surfaces. This report includes only representative data of some of the important parameters.

INTRODUCTION

The purpose of this investigation was to obtain inlet performance and dynamic distortion characteristics over an extensive maneuver envelope for a single engine, advanced lightweight fighter aircraft configuration with two types of side-mounted inlets. Normal shock and overhead ramp inlet configurations were tested. Several devices (bleed systems, cowl lip shapes, and a lower lip blow-in door) to minimize the normal shock inlet distortion at high angles of attack were also evaluated.

The test program, which was a cooperative effort of NASA, McDonnell Douglas Corporation, and the Navy was conducted in the Ames 11 by 11 Foot and 9- by 7- Foot Wind Tunnels (ref. 1) at Mach numbers of 0 to 2.0. Angle of attack ranged from -10° to 35° and angle of sideslip from -8° to 8° . Reynolds numbers ranged from 7.6×10^6 to 19.5×10^6 /m (2.5×10^6 to 6.4×10^6 /ft). Test measurements include engine face total-pressure recovery, steady-state distortion, dynamic distortion, and surface static pressures on the forebody and inlet surfaces.

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RESULTS AND DISCUSSION

The run schedule for the present investigation is shown in table 1. A sample of the tabulated steady-state data is shown in the appendix. A complete listing of the tabulated data are not presented in this report because of the large volume required, the data are available in reference 3 or from NASA-Ames Research Center, Moffett Field, California. Selected plots of the data are presented in figures 30 through 33.

Engine-face total pressure recovery, steady-state distortion, and the time-variant Pratt & Whitney fan total distortion parameter for the YF401 engine as functions of inlet mass flow ratio and angle of attack are shown for both the normal shock and overhead ramp inlets at Mach numbers of 0.9 and 1.4. All plots are at $\alpha = 0^\circ$. Plots of the normal shock inlet performance at $M = 0.9$ (fig. 30) show reasonable pressure recovery at $M = 0$ with a rapid drop at 10° . A reduction in pressure recovery is also seen at $\alpha = -10^\circ$. At $M = 1.4$ (fig. 31) a slight increase in pressure recovery is seen with increasing α , and again the decrease at negative angles. The overhead ramp inlet at $M = 0.9$ (fig. 32) shows a drop in pressure recovery at 10° , but not nearly so severe as the normal shock inlet. At negative angles of attack the loss in pressure recovery is much more pronounced at the lower mass flow ratios than with the normal shock inlet. At $M = 1.4$ (fig. 33) the overhead ramp inlet performance is considerably better than the normal shock inlet but shows the same slight increase in performance with increasing angle of attack. Negative angles again show a pronounced loss of performance. A large increase in mass flow ratio over that at $M = 0.9$ can also be seen. In general, improvements in pressure recovery are accompanied by corresponding reductions in inlet distortion for both inlet configurations.

Ames Research Center
National Aeronautics and Space Administration
Moffett Field, California 94035

February 6, 1976

NASA CR-264

INLET-TO-INLET SHOCK INTERFERENCE TESTS

By D. L. Motycka and J. B. Murphy

Prepared under Contract No. NAS 2-2079 by
UNITED AIRCRAFT CORPORATION
Pratt & Whitney Aircraft Division
East Hartford, Conn.

for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

SUMMARY AND CONCLUSIONS

The results of this experimental program can be summarized as follows:

1. It is possible to operate a mixed compression inlet in the region of influence of another inlet which is "buzzing", but it must be operated at a reduced contraction and hence a reduced total pressure recovery.
2. The pressure recovery penalties encountered were large for steady state operation behind the "buzzing" inlet, but the penalty required to maintain stability during the initial transient of the onset of "buzz" was so severe that it cannot be considered practical.
3. The line of demarcation between the regions of stable and unstable operation was found to approximately coincide with the shape of the expelled "buzzing" shock in its maximum forward position.
4. The pressure recovery penalties encountered while operating in the unstable region behind the expelled shock were reduced as the distance between the inlet cowl lip and the source of the expelled shock was made very large. The penalties were also reduced as the Mach number was reduced.
5. The use of a plate as an interference shield was very effective in eliminating the penalty imposed by the expelled shock.
6. The magnitude of the initial pressure pulse at the onset of "buzz" is the same as it is for subsequent cycles.
7. The "buzz" frequency increased as model airflow was reduced (throttled) but the pressure impulse amplitude remained approximately constant.
8. The basic "buzz" frequency was dependent upon the volume in the model between the cowl lip and the choked throttle rather than the volume from the cowl lip to the throat of the inlet.

NASA TM-72859

NASA Technical Memorandum 72859

Effect of Number of Probes
and Their Orientation on the
Calculation of Several Compressor
Face Distortion Descriptors

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*Dryden Flight Research Center
Edwards, California*

NASA
National Aeronautics
and Space Administration

Scientific and Technical
Information Office

1979

Reference 9
C. 23

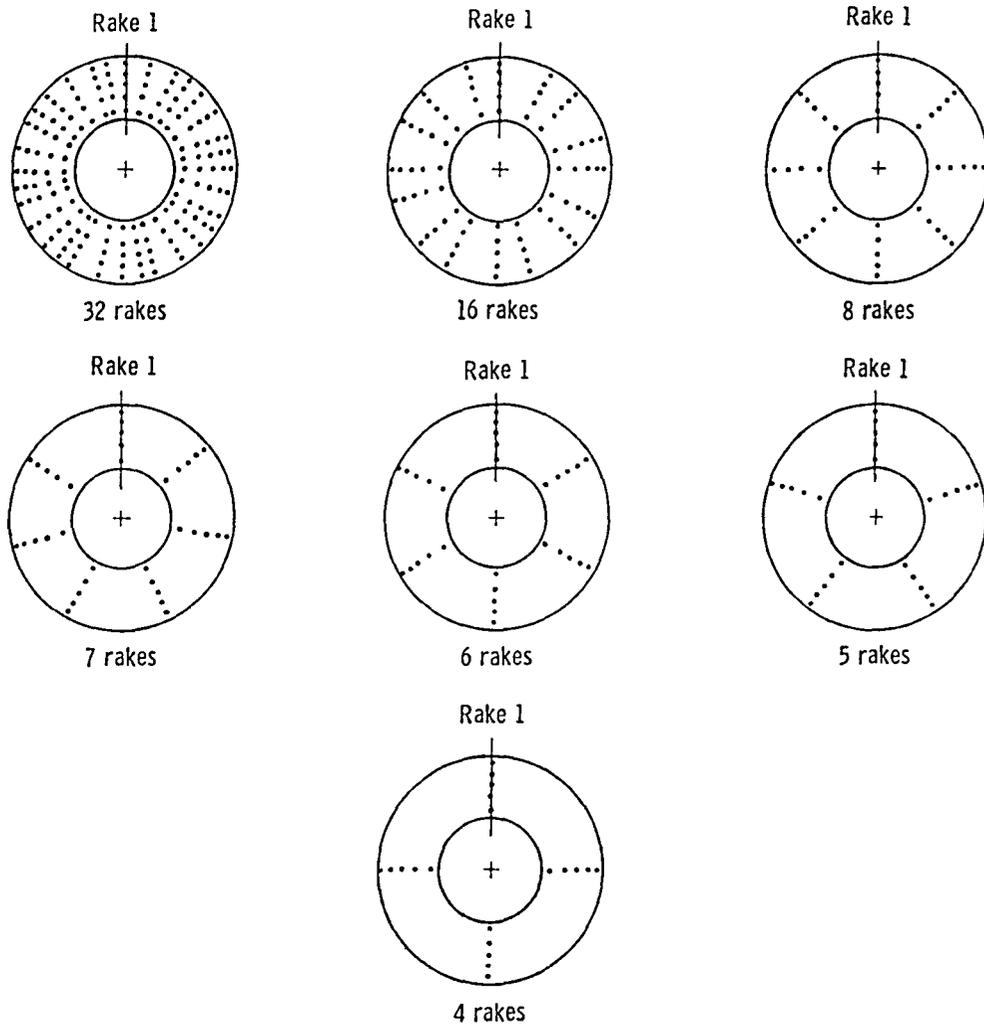


Figure 6. Simulated rake configurations obtained by selecting subsets of 64-rake/320-probe array (fig. 5).

SUMMARY OF RESULTS

A study was performed to determine the effects of the number and position of total pressure probes on the calculation of five steady state compressor face distortion descriptors and average compressor face total pressure. This study used sets of 320 total pressures that were obtained from wind tunnel tests made at three test conditions on a one-third-scale model of a YF-12 airplane mixed-compression inlet. The three test conditions were representative of flight conditions above Mach 2.0. The pressures were measured at the inlet compressor face with a special rotating rake apparatus.

This study showed that large errors can result in the calculation of distortion descriptors even with a number of equal-area-weighted total pressure probes that has been thought to be adequate in the past. For an eight-rake/40-probe configuration, the errors obtained in distortion descriptors for several rake locations were as follows:

1. For one circumferential distortion descriptor (K_{θ}), errors as large as ± 30 percent were obtained, and for another (IDC), maximum errors of 10 to -50 percent were obtained.
2. A radial distortion descriptor (K_r) showed errors of ± 30 percent.
3. A descriptor (K_a) that combined the circumferential distortion descriptor K_{θ} , with the radial descriptor, K_r , showed maximum errors of ± 20 percent.
4. Errors in the maximum-minus-minimum-over-average-pressure distortion descriptor, IDT, showed errors from 0 to -40 percent.

The use of fewer probes (from 20 to 40) resulted in errors for all the descriptors of 40 percent or more.

The calculation of average total pressure was also affected by probe number and location, although not to the same extent as the distortion descriptors. For the eight-rake/40-probe configuration, the errors were less than ± 1 percent. With fewer probes (from 20 to 40), the error increased to ± 2 percent.

Maps of total pressure at the compressor face that were constructed using the probes on eight or seven rakes showed the general pressure pattern, but failed to show the detail apparent in maps created using 64 rakes.

*Dryden Flight Research Center
National Aeronautics and Space Administration
Edwards, Calif., November 17, 1978*

NASA TM X-3540
CHARACTER OF RANDOM INLET PRESSURE FLUCTUATIONS
DURING FLIGHTS OF F-111A AIRPLANE

by William G. Costakis
Lewis Research Center

SUMMARY

Compressor face dynamic total pressures from four F-111 flights were analyzed. Statistics of the nonstationary data were investigated by treating the data in a quasi-stationary manner. This was achieved by analyzing short time segments. During these short time segments the data remained relatively constant. Changes in the character of the dynamic signal are investigated as functions of flight conditions, time in flight, and location at the compressor face. The results are presented in the form of rms values, histograms, and power spectrum plots. These results show that the shape of the power spectra remains relatively flat through the frequency range of the data and that the histograms exhibit an approximate normal distribution

NASA TM X-3540
CONCLUDING REMARKS

Compressor face dynamic total pressures from four F-111 flights were analyzed in a quasi-stationary manner. The following results were obtained:

1. The shape of the power spectra of the dynamic total pressure signals did not change appreciably for different flight conditions, locations at the compressor face, and time in flight. A resonance of approximately 23 hertz appeared in one of the flights.

2. Histograms of the magnitude of the dynamic total pressures indicated a fairly good normal distribution for all locations at the compressor face, flight conditions, and time in flight. Also, a linear relationship with a crest factor (ratio of highest peaks to rms values) of 3 was observed in accordance with theory for normally distributed data.

The results of this study indicate that if the rms values of dynamic pressure signals are calculated over short time segments they can predict the highest peaks expected within the segment when (1) the segment is chosen to be short enough so that the signal appears stationary within the segment, and (2) the character of the signal is consistent as shown by the previous results.

The flat power spectra shape and the normal distribution of the signals are highly desired characteristics. They can be very useful when implementing the distortion pattern synthesis methods, as well as in the formulation of simple distortion indices that can be used as the control signal in schemes designed to avoid stall.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, February 18, 1977,
505-05.

NASA TM X-71505

SUMMARY OF RECENT INVESTIGATIONS OF INLET FLOW DISTORTION EFFECT ON
ENGINE STABILITY

by Edwin J. Graber, Jr. and Willis M. Braithwaite

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ABSTRACT

A review is presented of recent experimental results, analytical procedures and test techniques employed to evaluate the effects of inlet flow distortion on the stability characteristics of representative afterburning turbofan and turbojet compression systems. Circumferential distortions of pressure and temperature, separately and in combination are considered. Resulting engine sensitivity measurements are compared with predictions based on simplified parallel compressor models and with several distortion descriptor parameters.

INTRODUCTION

In the development of an airbreathing propulsion system allowances must be made for the effects of inlet flow non-uniformities on engine stability. These flow distortions are detectable at the engine-inlet interface plane as variations in temperature and/or pressure. There are many factors which can contribute to the formation of such distortions. Typically, pressure distortions result from operation at high angles of attack and/or yaw, from wakes from nearby aircraft, and through inlet boundary layer interactions. Temperature distortions may result from armament firing or the ingestion of hot exhaust gases from thrust reversers, from steam driven catapults used on aircraft carriers, or from the exhaust plumes of nearby aircraft. In addition, temperature distortions or combined pressure and temperature distortions may enter intermediate or aft compressors in the more complex two and three spool compressor systems as the result of inlet pressure distortions. Increasing the design compressor surge margin to accommodate these unknown inlet distortions may result in an arbitrary reduction of engine performance accompanied by an increase in engine weight. Thus, the allocation of surge margin required for inlet distortions must be assessed as closely as possible without compromising engine stability.

During the past several years, NASA Lewis Research Center has been conducting a program to determine the effect of inlet flow distortions (both temperature and pressure) on the stability of several gas turbine engines (e.g., refs. 1 through 13). This paper is an extension of ref. 1 and presents the results of the steady-state circumferential temperature and pressure distortion tests conducted on two representative gas turbine engines; a simple one-spool turbojet (J85-GE-13) and a two-spool turbofan (TF30-P-3). Both the empirical approach of employing a distortion index to establish the distortion sensitivity or the stability limit of an engine and a comparison with the predicted results using parallel compressor theory are presented.

NASA TM X-71505

Summary of Results

1. The J-85 pressure distortion surges can be predicted reasonably well using the parallel compressor theory. As indicated by the theory, defining the loss of compressor surge pressure ratio at a constant corrected speed results in the best correlation with $\Delta P/P$.
2. The J-85 temperature distortion surges are predicted reasonably well for the above 90° extent distortion patterns. There is, however, disagreement for the 90° extent data which may be attributable to the pattern shape.
3. Correlation of the loss in surge pressure ratio with the basis $\Delta T/T$ parameter for the J-85 revealed no advantage in using either a constant corrected speed or constant corrected airflow definition of $\Delta P/P$.
4. Combined pressure and temperature distortion effects on the surge pressure ratio of the J-85 were in reasonable agreement with the parallel compressor model.
5. The stability limit of the TF30 engine could be correlated using either $((\Delta P/P) \cdot \Theta^-)$ or $((\Delta P/Q) \cdot \Theta^-)$ for the pressure distortion induced surge data.
6. The TF30 was slightly more sensitive to pressure than to temperature distortions.
7. Parallel compressor theory was found to be a useful tool in analyzing surge data for both engines.

Report of McDonnell Aircraft Co.

SYSTEM FOR EVALUATION OF F-15
INLET DYNAMIC DISTORTION

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Abstract

An instrumentation and data acquisition system for evaluating inlet dynamic distortion has been developed for use in the F-15 full-scale wind-tunnel and flight-test programs. The system was used successfully during the full-scale inlet/engine test in the wind tunnel at AEDC and is currently being used in the flight-test program. The system consists of the following: high and low frequency response pressure transducers mounted in an inlet rake, data acquisition systems for both high and low response measurements, and an analog computer for economical evaluation of dynamic distortion data. The rake incorporates 48 low response and 48 high response total pressure probes, arranged in an 8-leg, 6-ring, configuration. The transducers for the low response probes are located in a temperature controlled compartment within the engine nose dome. The high response transducers are located on the rake legs, adjacent to the low response probes. After filtering, separate data acquisition systems record the low and the high response data. The combined total pressure signal, made up of the low and high response signals, has nearly flat response from 0 to 1000 Hz. Because cost prohibits digital reduction of all of the recorded data, an analog computer is used to monitor the dynamic data and mark the data tape in the regions of peak distortion. System accuracy is satisfactory throughout the aircraft flight envelope. The design has been closely coordinated with the F-15 engine manufacturer and is used for dynamic distortion measurement by both the airframe and the engine contractors.

Report of McDonnell Aircraft Co.

Conclusions

To evaluate F-15 inlet dynamic distortion accurately, a system is required which can measure low and high response pressures during both aircraft steady-state conditions and rapid maneuvering transients. The system developed to meet these requirements provides both steady-state and high frequency response pressure data throughout the aircraft operational temperature and pressure ranges. Accurate low and high response data are measured to establish both the inlet recovery and dynamic distortion levels. In addition, the inlet rake meets all the design criteria for the system, including a minimal interface with the engine and transducer replacement without engine removal. This system was used during a full-scale inlet/engine wind-tunnel test at AEDC and is being used during F-15 flight testing.

Because of the massive amount of data recorded during dynamic distortion testing, a specialized monitoring and editing device is required that selects specific regions in the recorded data for post-test reduction. An analog

EVALUATION OF INLET DISTORTION

computer is used to compute continuously distortion from the dynamic pressure data and to mark the data tapes in the regions of peak distortion. Only data in these regions are required to be reduced digitally.

The accuracy of this measurement system is satisfactory throughout the aircraft flight envelope. The complexity and the potential errors in the distortion measurement system make it desirable to coordinate both the rake design and instrumentation between the engine and the airframe contractors. A potential error source is eliminated if both the engine and the airframe contractors use the same distortion measurement system and the same editing device. Consistent data are then obtained on the distortion tolerance of the engine and on the distortion produced by the inlet.

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NASA TM X-67495

A FLIGHT INVESTIGATION OF STEADY-STATE AND DYNAMIC PRESSURE PHENOMENA IN THE
AIR INLETS OF SUPERSONIC AIRCRAFT

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SUMMARY

The difficulty of achieving adequate inlet performance and stability and avoiding engine compressor stalls at supersonic speeds has led the NASA Flight Research Center to investigate the pressure phenomena in the inlets of several supersonic aircraft. Results of recent tests with the F-111A airplane are presented showing the inlet steady-state and dynamic performance. The inlet total pressure distortion that causes compressor stall is discussed, and the requirement for high response instrumentation is demonstrated. A duct resonance encountered at Mach numbers near 2.0 is analyzed and shown to be due to a normal shock oscillation at the duct fundamental frequency. Various statistical parameters are used in the analysis. Another resonance, in the engine fan duct, is shown to be a possible cause of reduced engine stall margin in afterburning operation. Plans for a comprehensive inlet study of the YF-12 airplane are discussed including flight tests and full-scale, 1/3-scale, and 1/12-scale wind-tunnel tests.

CONCLUDING REMARKS

The steady-state and dynamic pressure phenomena in the inlets of two F-111A airplanes were investigated. A new pressure survey rake incorporating miniature transducers and a slide valve to permit in-flight zero checks was used at the compressor face and was found to yield excellent steady-state and dynamic pressure data. Analysis of the high response pressure data showed the cause on compressor stalls to be high levels of instantaneous distortion lasting for times as short as 3 milliseconds. In many instances, the distortion causing the stall was due to an out-of-phase situation in total pressures on opposite sides of the compressor face. An inlet duct resonance at Mach numbers above 1.8 was analyzed by using time history and statistical parameters and was found to result from a normal shock oscillation at the duct fundamental frequency. Another resonance in the fan duct of the engine during afterburning operation may have affected the compressor stall margin. Plans for studies of the inlets of the YF-12 interceptor include flight tests and comparison with full-scale, 1/3-scale and 1/12-scale wind-tunnel tests.

NASA TM X-66855

STUDY OF AIRCRAFT FLOW FIELDS AT

INLET LOCATIONS

by

L. S. King

January 1971

SUMMARY

Results are presented from a wind-tunnel investigation of the flow fields about fuselage configurations at transonic and supersonic speeds ($0.8 < M_\infty < 2.5$) and at angles of attack up to 24° . A family of seven fuselages with different cross-sectional shapes was tested in conjunction with two nose shapes, two canopies, and two wings of different sweep. Flow-field surveys were performed at two likely inlet locations ahead of and under the wing to assess the effects of forebody geometry throughout the Mach number and angle-of-attack envelope.

Flow field results are presented as contour maps of the local values of Mach number, total pressure, and flow angularities in the two survey planes. Comparisons are made to illustrate the effects of Mach number, angle of attack, and forebody variations. Comparisons are also made of the results with predictions using the numerical method of Walitt, et. al.

The experimental data particularly indicate the strong influence of the canopy, nose droop, and fuselage shape on flow angularities in the forward survey plane. Nose droop and the canopy both tend to reduce sensitivity to positive angles of attack and to reduce the extent of influence of fuselage lower corner geometry. Under the wing, however, the flow field is dominated by the effects of the wing itself.

7. Concluding Remarks

Immediately obvious from the flow surveys at inlet locations is the diversity of flow fields in which the inlet must operate over the range of Mach numbers and angles of attack. None of the fuselage, forebody configurations tested, provided the optimum flow fields a low distortion flow of high total pressure with little change in Mach number and angle of attack. However, some trends with fuselage geometric variables were noted.

Effects of fuselage lower corner geometry are confined primarily to the lower inboard quadrant of the survey area at the forward inlet station. A corner with a small average radius tends to produce large gradients in flow angularity and local Mach number with some loss in total pressure, whereas a larger average radius tends to reduce these gradients and to distribute them over a larger region of the flow field. Concentration of most of the curvature in the turn toward the bottom of the corner also tends to reduce gradients in the survey area.

The influence of the canopy and nose droop is also indicated at the forward inlet station. At the higher Mach numbers, the drooped nose and the canopy combined to cause a downwash and positive sidewash for $\alpha = 0^\circ$ and to reduce the upwash for positive angles of attack. The alternate nose and the alternate canopy generally caused relatively insignificant changes in the flow field. Depending on the relative locations of the canopy and the inlet, however, the flow over the canopy can influence the entire side region and interact with the flow around the fuselage lower corner. For the alternate canopy tested, this interaction improved total pressures near the fuselage corner.

At the aft survey plane, the flow field is dominated by the wing. The wing compresses the flow, resulting in relatively high pressures, local Mach numbers less than free stream, and large variations in downwash and sidewash across the survey region. Flow from around the fuselage lower corner into this high pressure region causes an area of low total pressure at the corner extending into the survey field.

NASA TN D-6679

**FLIGHT-DETERMINED CHARACTERISTICS OF AN AIR INTAKE SYSTEM
ON AN F-111A AIRPLANE**

**Donald L. Hughes, Jon K. Holzman, and Harold J. Johnson
Flight Research Center**

INTRODUCTION

Matching inlet airflow to the propulsion system requirements of today's high supersonic flight vehicles is extremely difficult because of the wide range of Mach numbers, altitudes, and angle-of-attack capabilities required to perform the desired missions. If the air inlet of this type of vehicle is to be compatible with the engine and operate efficiently over a wide range of ambient pressures, some means must be provided for varying the inlet entrance geometry.

The F-111A airplane was of interest to the NASA Flight Research Center as a research vehicle for inlet-engine investigation because of its propulsion system design and its Mach number capability. In addition to three-dimensional external compression inlets with variable geometry, the airplane had a new type of engine, an afterburning turbofan. The F-111A airplane was capable of covering a broad flight envelope that included supersonic flight at sea level as well as supersonic flight at greater than Mach 2.0 at altitude.

This report documents the quasi-steady-state flow phenomena of the air intake system on a preproduction model of the F-111A airplane during 16 flights conducted at the Flight Research Center. Investigated were boundary-layer variations at the leading edge of the splitter plate and at the inlet entrance station as well as the effect of flight variables, such as Mach number, altitude, and angle of attack, on compressor face pressures. The performance of the inlet is shown in terms of pressure recovery, corrected airflow, mass-flow ratio, turbulence factor, distortion factor, and power spectral density of time-variant pressure fluctuations at the compressor face.

Various configurations of the F-111A air inlet were tested in wind tunnels and in flight. The specific inlet configuration flight-tested by the NASA Flight Research Center is not an exact duplicate of any model that had been tested previously in a wind tunnel; however, pressure recovery values at the compressor face obtained in flight and in 1/6-scale and full-scale wind-tunnel tests are compared.

NASA TN D-6679

CONCLUDING REMARKS

Flight-determined characteristics of the pressure and flow phenomena in the air intake system of an F-111A airplane were established by investigating the inlet-forebody and wing glove flow fields, inlet shock system, inlet-engine flow propagation, and compressor face recovery, distortion, and turbulence levels. During evaluation of the flows ahead of the inlet, it was found that the fuselage boundary-layer height exceeded the splitter plate height at the upper splitter plate rake for all Mach numbers investigated. At the inlet lip, the boundary-layer height exceeded the wing glove and sideplate bleed heights at the rakes closest to the translating spike and cone beginning at about Mach 1.6. The height of this boundary-layer air ingested into the inlet increased with increasing Mach number. The trend of the data also showed that the inlet became supercritical at approximately Mach 1.8 to 1.9 with the blunt-lip cowl configuration. Between Mach 1.8 and 2.0, a resonance of 27 hertz to 29 hertz in the inlet duct appeared to originate near the inboard wing glove rake at the inlet lip.

The compressor face pressure recoveries obtained in flight agreed with pressure recoveries obtained on 1/6- and full-scale wind-tunnel models within the measurement accuracy over the Mach number range. The distribution of pressure recovery at the compressor face showed increasing distortion with increasing angle of attack and increasing Mach number. Within the operable limits of the airplane, the increased distortion level was more a function of Mach number than angle of attack. At Mach 2.15 the distortion pattern approached a 180° division of high and low pressures, which approached a stall condition for the engine in the test airplane. The time-averaged distortion factor values, K_D , also approached 1300 at these high supersonic Mach numbers, indicating that the engine was operating near its distortion limit.

An evaluation of power spectral density curves showed that all the probes at the compressor face sensed increased turbulence, which was more uniformly distributed around the compressor face at high supersonic Mach numbers than at low supersonic and subsonic Mach numbers.

Flight Research Center,
National Aeronautics and Space Administration,
Edwards, Calif., October 5, 1971

NASA TN D-7328

STEADY-STATE AND DYNAMIC PRESSURE PHENOMENA IN
THE PROPULSION SYSTEM OF AN F-111A AIRPLANEFrank W Burcham, Jr., Donald L Hughes, and Jon K Holzman
Flight Research Center

INTRODUCTION

The pressure phenomena in the propulsion system of supersonic aircraft have great influence on performance. Low levels of inlet pressure recovery result in reduced performance and range. Nonuniformities in pressure, both steady state and dynamic, may result in reduced performance, and in some cases they cause engine compressor stall. The resulting hammer shock waves and inlet buzz generate high structural loads in the inlet. In some aircraft configurations severe propulsion system airframe interactions may occur. Pressure fluctuations in the engine may also have adverse effects on the aircraft, particularly in afterburning turbofan engines, where afterburner pressure fluctuations may propagate directly to the fan, where they may then affect the inlet flow.

The effects of steady-state pressure distortion are well known, but the importance of dynamic pressure fluctuations was not fully appreciated until Kimzey's work with the J93 engine (ref 1) which showed that engine stall could occur at very low steady-state distortion levels if the pressure fluctuations were sufficiently severe. At that time the XB-70 airplane, powered by J93 engines, was being flight tested, and a limited number of dynamic pressure measurements were made in the inlet and at the engine compressor face (ref 2). Early flight tests of the F-111A airplane, which is powered by afterburning turbofan engines, showed numerous compressor stalls at steady-state distortion levels well below the limits established from ground tests. Extensive tests of the TF30 engine in ground facilities using random turbulence generators (refs 3 and 4) suggested that dynamic pressure fluctuations were causing the stalls. An airjet system at the NASA Lewis Research Center was used to study the effects of periodic and single pulse pressures on the TF30 engine (refs. 5 to 7).

In order to study these dynamic pressure phenomena in flight, the NASA Flight Research Center instrumented and flight tested an early model of the F-111A airplane (refs. 8 to 11). Miniature transducers were installed inside rakes in the inlet to permit adequate frequency response to study the dynamic pressure fluctuations. After these tests were completed, a new type of compressor face rake (ref 12) was designed which improved the quality of the data. A follow-on program was flown with another early model of the F-111A airplane in which eight of these compressor face rakes were installed, and more accurate and complete data on the inlet were obtained (ref 13).

This report summarizes the data obtained from both F-111A airplanes. It includes inlet pressure recovery, distortion, and turbulence factor data as functions of Mach number, angle of attack and sideslip, inlet airflow, and inlet geometry. Inlet duct resonances are analyzed, and inlet buzz is described. Engine stall tolerance is presented as a function of corrected airflow, power setting, and compressor bleed position using several distortion parameters. The effects of rotating stall and rotor speed harmonics are also shown.

NASA TN D-7328

CONCLUSIONS

Flight tests of two F-111A airplanes to study the effects of steady-state and dynamic pressure phenomena on the propulsion system yielded the following conclusions:

1. Most of the engine compressor stalls were caused by high levels of instantaneous distortion. The distortion was usually the result of a simultaneous increase in pressure in the high pressure recovery area and decrease in pressure in the low pressure recovery area.
2. The instantaneous circumferential distortion parameter K_p exhibited a peak just prior to stall higher than any previous peaks in 73 of 100 stalls analyzed. The distortion factor K_D was the second best parameter for indicating stall while the maximum-minus-minimum distortion parameter D was a poor indicator of stall.
3. Inlet duct resonance occurred in both test airplanes. In F-111A number 6 the resonance was out of phase on opposite sides of the compressor face causing high distortion while in F-111A number 12 the resonance was in phase across the compressor face causing large fluctuations in pressure recovery. Both resonances were believed to be caused by oscillations of the normal shock wave from an internal to an external position.
4. The inlet performance of the two airplanes in terms of pressure recovery, distortion and turbulence was similar. Agreement between flight and wind-tunnel data was good up to a Mach number of approximately 1.8.
5. Power spectral density plots of compressor face pressures generally showed increasing power levels with increasing Mach number. When power levels were low the power spectral density curves tended to be flat, while at higher power levels the curves tended to fall off with increasing frequency.
6. The frequency and amplitude of the inlet duct buzz data obtained in flight compared reasonably well with wind-tunnel and theoretical data.

NASA TM X-71574

SOME COMPARISONS OF THE FLOW CHARACTERISTICS
OF A TURBOFAN COMPRESSOR SYSTEM
WITH AND WITHOUT INLET PRESSURE DISTORTION

by David G. Evans, Claude E. deBogdan,
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ABSTRACT

The measured effects of a circumferential distortion in inlet total pressure on the fan, low, and high compressor of an afterburning turbofan engine are presented and discussed. Extensive inner-stage instrumentation, combined with a unique test technique offered an accurate means of measuring the shifts in flow, performance, and stall mechanisms within the compressor. These effects are compared at one speed to the corresponding effects measured with undistorted inlet flow. The results show the rate at which the distorted flow areas were attenuated and rotated, as well as the change in flow velocities that occurred at various points in the compressor. High response pressure traces indicated the location of stalls including the sequence of dynamic events from the onset and propagation of various stall-recovery events, to compressor surge, to the resulting hammer shock.

INTRODUCTION

The effects of inlet flow distortion on the performance and stability of aircraft gas turbine compressor systems has long been an important consideration in engine development programs. In support of this, an extensive program has been in progress at the Lewis Research Center to evaluate the effects of inlet distortion on various types of engines, including the turbojet and the more complex types of turbofan engines. This paper is directed towards one aspect of the program where a 180 degree circumferential square wave in inlet total pressure distortion was imposed upstream of a TF30-P3 afterburning turbofan engine, shown in figure 1. The effects of the inlet distortion on the flow characteristics within the compressor system were measured and compared to its operation with an undistorted inlet.

The results of the tests are currently being analyzed. However, the data indicate the shifts in the flow conditions and stall mechanisms that occur within the compressor system due to the distorted inlet. Implied in the results are the influence that the design and performance characteristics of the compressor system have on the way in which the flow reacts to the distortion. That is, the way in which the compressor stage characteristics, stage exit volumes, cavities adjacent to the flow path, and the propagation of the flow defect areas effect the compressor inner-stage distortion attenuation, combined distortion profiles, flow velocities, overall compressor performance, and the location and type of dynamic events that occur during surge. These data and results are discussed in the following sections.

NASA CR-2414

SEPARATED FLOW OVER BODIES OF REVOLUTION
USING AN UNSTEADY DISCRETE VORTICITY
CROSS WAKE (Part I)

By

F. J. Marshall

SUMMARY

A method has been developed to determine the flow field of a body of revolution in steady separated flow (e.g. at high angle of attack), including the prediction of the normal force and pitching moment.

The method makes use of a computer to integrate the various solutions and solution properties of the sub-flow fields which make up the entire flow field. Thus, a finite difference solution to the complete Navier-Stokes equations is not employed.

The method utilizes two ideas in this approach to a steady three-dimensional separated flow the unsteady cross flow analogy which reduces the given three-dimensional steady separated flow to a two-dimensional unsteady separated flow and a new solution technique for the latter problem. This technique employs a wake description of discrete point vortices arising from the separation of shear layers at the surface. The point vortices convect and diffuse downstream to form an unsteady wake. Thus the mathematical model follows directly from the physics of the wake evolution with time.

The overall technique is applied, employing a computer, to two test cases an ogive-cylinder and an ellipsoid of revolution at low speeds. Force and moment data are obtained and are found to agree well with experimental data, particularly at high angles of attack where inviscid theory is invalid. Separation regions and wake patterns are found which agree with available experimental findings.

NASA CR-2414

6.0 Conclusions

The method developed herein employing the unsteady cross flow analogy and a discrete vorticity wake for the prediction of local flow conditions and overall forces and moments on bodies at high angles of attack with large regions of separated flow has been shown to be technically feasible. The method is based upon physical understanding of the flow and does not require a complete finite difference solution to the Navier-Stokes equations.

The advantages of such an approach lie in the potential of such solutions to bring about further physical understanding with consequent new analytic solutions (e.g. the wake) and in the area of computer-aided-design of aircraft where the engineer requires the physical understanding. In addition, there is the advantage of reduced computer time and storage although this may be offset somewhat with the advent of larger and faster computers.

The results support the use of the unsteady cross flow analogy for three dimensional steady separated flows. A theoretically derived equivalence between two dimensional unsteady and three dimensional steady flow has been replaced by one empirical factor, σ , with physical bounds, i.e. $0 < \sigma \leq 1$ (cf. the eddy viscosity in turbulent flow). Further testing of the technique, with the use of equation (58) for σ , should shed further light on the role played by this factor.

NASA TN D-7839

ANALYSIS OF THE DYNAMIC RESPONSE OF A SUPERSONIC INLET
TO FLOW-FIELD PERTURBATIONS UPSTREAM
OF THE NORMAL SHOCK

by Gary L. Cole and Ross C. Willoh

Lewis Research Center

SUMMARY

A linearized mathematical analysis of supersonic inlet dynamics is presented. Attention is concentrated on determining the response of normal shock position and subsonic duct pressures to flow-field perturbations upstream of the normal shock in mixed-compression inlets. The analysis is based on a previous NASA report which dealt primarily with perturbations downstream of the normal shock.

The inlet duct cross-sectional area variation is approximated by constant area sections. This approximation, in combination with a linearized analysis, results in one-dimensional wave equations for each duct section. The supersonic and subsonic flow regions are separated by a movable normal shock. A choked exit is assumed for the inlet exit condition. The analysis leads to a closed-form matrix solution for the shock position and pressure transfer functions.

The analysis was compared on a frequency response basis with a method-of-characteristics solution. The agreement in both amplitude ratio and phase was excellent.

Analytical frequency response results were also compared with experimental data. The phase angle results were generally in good agreement. Amplitude ratio response curves, although not in as good agreement as phase data, showed generally good agreement in shape. Some shifts in low-frequency gain were found, however.

NASA TN D-7839

INTRODUCTION

Propulsion system performance of supersonic aircraft depends greatly on the type of inlet system being used. Mixed-compression inlets are best for achieving high propulsion system performance in aircraft that fly at Mach numbers in excess of about 2.0. Because of the nature of mixed-compression inlet design, a normal shock wave usually exists within the inlet during supersonic operation. For best inlet performance the normal shock must be positioned near the throat where the shock is susceptible to being displaced by disturbances arising from such things as atmospheric perturbations, aircraft maneuvers, and changes in engine operation. A downstream displacement of the shock results in a loss of inlet performance. If the shock is displaced in the upstream direction, it is in danger of being expelled from the inlet. This is referred to as inlet unstart. Inlet unstart may result in undesirable consequences such as compressor stall, combustor flameout, reduced propulsion system thrust and increased vehicle drag. To counteract such possibilities, mixed-compression inlets are provided with variable geometry features that can be automatically controlled to keep the shock at the desired position. The design of these shock-position control systems requires knowledge of the shock-position dynamic response to perturbations in the inlet. The purpose of this report is to develop an approximate mathematical analysis for predicting the dynamic response of shock position and pressures, in mixed-compression inlets to perturbations upstream of the normal shock.

In the past a great deal of attention has been given to controlling shock position against airflow perturbations originating downstream of the normal shock (refs. 1 to 6). Normal shock motion due to perturbations downstream of the shock has been examined analytically (e.g., refs. 7 and 8). Reference 9 presents a mathematical analysis that is valid for obtaining shock responses to perturbations downstream or just upstream of the normal shock. However, the upstream terms were dropped and consideration was restricted to the downstream case. Reference 10 does deal with upstream perturbations and presents an analysis that includes both storage-volume and Helmholtz-mass effects. Reference 11 presents transfer functions for shock position to upstream perturbations. The transfer functions vary in complexity from first to fourth order, and results are compared with a method-of-characteristics solution. The fourth-order model is a linearized version of the analysis in reference 10.

In this report a mathematical analysis is presented that combines a set of linearized equations across the normal shock with an exact solution of the linearized one-dimensional wave equation. This analysis is based on the analysis of reference 9. Use of this technique avoids the complexity of the method of characteristics while still predicting the resonances of a distributed parameter system. However, flow-field discontinuities due to oblique shock waves in the supersonic duct are neglected. The analysis presented is more exact than conventional lumped-parameter techniques and for frequency response results no more complicated in application. If transient responses are desired the equations are especially suitable for analog simulation. The analysis is followed by a matrix solution that provides a simple means for obtaining frequency responses on a digital computer. Finally, analytical frequency responses are compared with results from a method-of-characteristics solution found in reference 11 and with experimental data obtained during a wind tunnel program. Dr. Frank Barry of the Hamilton Standard Company supplied us with information regarding the method-of-characteristics analysis in addition to that in reference 11.

NASA TN D-7839

CONCLUDING REMARKS

An analysis was presented for determining the dynamic response of a supersonic inlet to perturbations ahead of the normal shock. The analysis, based primarily on the linearized, one-dimensional, distributed parameter wave equation, is applicable to analysis of mixed-compression inlets. Discontinuities and losses in the supersonic duct due to oblique shock waves are neglected. The equations are suitable for implementation on an analog computer. Also, closed-form expressions for the evaluation of frequency responses were obtained using matrix operations (no inversion required). These expressions are easily programmed on a digital computer and require little computer time to solve.

The major difficulty in applying the analysis arises in choosing the correct value of the A'/A parameter. Because of shock-boundary layer interaction effects, the shock response is not uniquely dependent on the geometric value of A'/A . In this report an effective A'/A was determined empirically from the experimentally measured gain of shock position to the perturbation variable. The effective A'/A values varied from 3 to 7 times the geometric value. Further work is required to determine a means for estimating an effective A'/A when experimental data are unavailable. This is important because A'/A is common to most inlet analyses.

SUMMARY OF RESULTS

Analysis frequency response results were compared with a method-of-characteristics solution over the frequency range of 5 to 40 hertz. The perturbed variable was Mach number just ahead of the normal shock. Agreement was excellent in both amplitude ratio and phase angle over the entire frequency range. The analysis has the advantages of being simpler to program and requiring much less computer time to solve.

Frequency response comparisons were also made of the analysis with experimental inlet data obtained in a wind tunnel. The perturbation frequency range was 1 to 15 hertz. Data were obtained with the inlet coupled to a turbojet engine, a short pipe, or a long pipe (the pipes having choked exit airflows). In these cases the perturbed variable was Mach number just ahead of the inlet.

Phase angle agreement was generally very good for both shock position and static pressures in the subsonic duct. When the transportation times for the supersonic duct were eliminated, the error in phase angle generally remained small except for the shock position responses with the inlet coupled to the engine or short pipe.

The A'/A parameter (the rate of change of duct area with shock position divided by the duct area - evaluated at the shock operating point) was selected to assure good amplitude ratio agreement for the shock position responses. In general, the comparisons showed good agreement between the shapes of the static-pressure amplitude ratio response curves. However, in some cases a substantial shift in gain was observed. For the inlet-engine case the analysis predicted that static pressures near the subsonic diffuser exit had a low-frequency gain 1.6 times larger than the experimental values. The discrepancy may be due in part to the fact that the engine face is not choked as was assumed for the analysis. A static pressure near the normal shock was in error by as much as 27 percent. This error is believed to be due to the fact that the analysis relates variables across the normal shock which ideally is a discontinuity. In reality this shock may occur as a series of shocks or a shock train.

NASA TM X-3267

EFFECT OF A 180°-EXTENT INLET PRESSURE DISTORTION ON THE
INTERNAL FLOW CONDITIONS OF A TF30-P-3 ENGINE

by Claude E. de Bogdan, John H. Dicus, David G. Evans,
and Ronald H. Soeder

Lewis Research Center

SUMMARY

The measured effects of inlet pressure distortion on the internal flow temperatures and pressures of a Pratt & Whitney TF30-P-3 afterburning turbofan engine are reported. Extensive inner-stage instrumentation combined with stepwise rotation of pressure distortion provided a high degree of circumferential resolution in the data. The steady-state spatial variation in pressures, temperature, and calculated flow velocity and the amplitude and extent of the distorted sectors are given. Data are presented for runs of 77 and 90 percent of low-speed-rotor design speed at pressure distortion levels two-thirds of that required to stall the engine. These data are compared with data taken at clean inlet conditions. Results indicate that the inlet pressure distortion was quickly attenuated within the compressor, except at the hub of the low-pressure compressor. The distorted sectors also swirled and varied in extent as they passed through the engine. Static pressure distortion was attenuated by the large passage volumes at the fan and compressor exits, with a resulting decrease in total pressure distortion in these areas. The overall performance of the compressor system with a distorted inlet did not change substantially from its clean-inlet performance.

Contained in this report are curves showing, for each measuring station, the attenuation (amplification in the case of total temperature) and swirl of the total and static pressure, total temperature, and calculated flow velocity variations within the compressor system. The spatial relationship among the total pressure, static pressure, and temperature distortions that are present within the compressor is shown at various measuring stations. Curves are also presented which compare the average flow velocities obtained with and without inlet distortion at each measuring station. Detailed plots of internal pressures, temperatures, calculated velocities, and average Mach numbers are presented in an appendix to this report.

NASA TM X-3267

SUMMARY OF RESULTS

The effect of operating the TF30-P-3 turbofan engine with 180° distortion in inlet total pressure on the flow conditions measured within the compressor system at low-speed-rotor speeds of 7400 rpm and 8600 rpm are summarized as follows

- 1 The air-jet distortion device used produced a nearly square-wave profile in total pressure and flow velocity at the inlet, with some dropoff in amplitude at the inner radius of the inlet annulus. The corresponding circumferential profile in static pressure was sinusoidal.
- 2 A large buildup in static pressure distortion occurred immediately ahead of and within the first stage of the fan. The buildup resulted in a static pressure distortion amplitude at station 2-1 that was approximately equal to the amplitude of total pressure distortion measured at station 2.
- 3 The amplitudes of the static pressure distortion at the fan and at the low- and high-pressure-compressor exits were equal to the amplitudes of the total pressure distortion measured at these stations at 7400 rpm but varied from 1/2 to 1 times the total pressure distortion amplitude at 8600 rpm.
- 4 A two-lobe circumferential variation in flow velocity of 5 to 30 percent in amplitude occurred within the second and third fan stages, probably caused by the difference in rate of rotation, attenuation, and shape of the total and static pressure defect areas.
- 5 The distorted sectors within the compressor system shifted from a purely circumferential distortion at the engine inlet to a mixed circumferential and radial pattern at all measuring stations within the compressor system.
- 6 The rate at which the distortion in inlet total pressure attenuated within the compressor system increased with increasing rotor speed.
- 7 Some swirl and variation in the circumferential extent of the distorted sectors occurred within the compressor system. In general, all sectors rotated opposite to rotor rotation back to station 2-3, where they reversed and rotated with the rotor back to the low-pressure-compressor exit. The high-pressure-compressor tended to break the pressure sectors into two zones and to rotate the temperature sector from 40° to 80° in the direction of rotor rotation.
- 8 The regions of total and static distorted pressure remained approximately coincident throughout the compressor system. However, the region of elevated total temperature overlapped the minimum pressure region by approximately 90° and did not coincide with the calculated particle flow path through the compressor system.
9. The observed variation in total temperature distortion amplitude did not compare well with that calculated based on variation in heat of compression caused by the circumferential variation in compressor pressure ratio.
10. A significant amplitude of circumferential and radial total pressure distortion was present between the low- and high-pressure compressors. The cause was not apparent, but it may be indicative of the presence of crossflows at station 3 or of some blading characteristic in the hub region of the low-pressure compressor.
- 11 The average flow velocities measured at each station within the compressor system (when operating with inlet distortion) were approximately equal to the velocities measured with a clean inlet.
- 12 A sinusoidal variation in total temperature was measured at the exit of the low-pressure turbine and fan bypass duct (stations 7f and 7g). The variation between the two stations was out of phase by 180° .
- 13 The testing technique of rotating the inlet pressure distortion was an effective means of mapping the spatial variation in flow conditions within the engine.

NASA TM-X-3267

14 The levels of inlet total pressure distortion used in these tests had little effect on the overall operating parameters

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, February 27, 1975,
505-05.

NASA CR-2686
ANALYSIS OF DISTORTION DATA FROM TF30-P-3
MIXED COMPRESSION INLET TEST

By

R. W. King
June 1976

SUMMARY

Pratt & Whitney Aircraft conducted a twelve-month program to reduce and analyze inlet and engine data obtained during the recent NASA/Lewis Research Center testing of a TF30-P-3 with an axisymmetric Mach 2.5 mixed compression inlet. As part of the test, the propulsion system was intentionally mismatched to generate combinations of steady state distortion and turbulence that allowed the engine to drift into surge. During these "drift stall" test runs, inlet and engine pressure data was recorded with high response instrumentation. This data was evaluated to develop methods of correlating inlet pressure distortion data to loss in engine surge margin. Both analog and digital data reduction techniques were used in the evaluation of the distortion analysis techniques. Results of previous TF30 engine tests were used in the development of the distortion analysis methods. In addition to the development of data analysis methods, an evaluation and refinement of a synthesis technique for the estimation of peak instantaneous distortions was conducted.

The program was divided into two tasks. Task I consisted of extensive analysis of ten "drift stall" points. The objective of Task I was to define a set of procedures for the analysis of inlet pressure distortion. This task included analog reduction of inlet and engine parameters, digital analysis of inlet data, and analysis of engine response to stall propagation. Task II consisted of the verification of the procedures defined in Task I and the evaluation of the peak distortion estimating synthesis methods. Task II was conducted by the analysis of an additional twelve "drift stall" points.

A distortion methodology that successfully indicated surge inducing inlet distortion was developed. Analysis of the engine data showed that virtually all of the instabilities were initiated in the low pressure spool compressor, suggesting that a core flow distortion factor should be used for inlet data analysis. It was found that by using a core flow distortion factor, K₀ splitter, stall inducing distortion could best be detected when the inlet pressure data is preconditioned by a filter with a cut-off frequency equal to the low compressor rotor speed. A destabilizing influence of full face in-phase pressure fluctuations was observed.

A method of estimating maximum instantaneous distortion factor levels from steady state total pressure measurements and turbulent RMS measurements was defined. Application of this distortion synthesis method resulted in a reasonably good correlation of estimated to actual stall inducing values of instantaneous distortion.

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RESULTS AND DISCUSSIONAnalysis of Engine Data

The surges which occur in propulsion systems are dynamic events. Significant events leading to and including the surge occur during periods of time measured in terms of milliseconds. Identification and analysis of these short duration events requires installation of high response instrumentation at locations which are close coupled to the engine compression system. High response pressure instrumentation was installed in this manner during NASA-Lewis testing of the TF30-P-3 engine/mixed compression inlet configuration. The data acquired from the compression system was analyzed in support of the inlet distortion data analysis for the purposes of 1) determining which compression system component was critical to system stability and 2) to attempt to identify the origin and propagation of instabilities through the system.

Definition of Terms

The stability limits of a compression system are sometimes referred to in terms used interchangeably as "surge" or "stall" limits. Use of the terms in this manner does not give insight into the actual events occurring during a system instability. For this reason, Pratt & Whitney Aircraft defined a glossary of terms for analyses of the type performed on the NASA/TF30 data. The definitions are given below to facilitate understanding of the TF30 analysis.

Stall

The term stall or flow separation refers to the local deviation of the airflow from contours of the airfoils or walls. A stall may be either abrupt - the flow rapidly transits from being attached to being separated over a large area, or gradual - the extent of the surface which is separated varies smoothly with compressor pressure ratio. A stall region may remain fixed in relation to hardware, or may propagate, as in the case of rotating stall. A compression system may operate stably with some stalled regions present on some airfoils or walls; however, local stalls can induce a system instability.

• Rotating Stall

Rotating stall occurs when a localized stall propagates circumferentially. It usually begins as a "cell" of stalled flow covering a fraction of the circumference and span, but may grow to cover the whole span and over half the circumference. A local reduction in flow accompanies the rotating stall cell. The growth of the stall cell is a compression system instability which generally leads to system surge.

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Surge

Surge is a system instability which originates as a major flow breakdown at some axial location in the compressor that stops or retards the flow around essentially the whole circumference. The flow breakdown in a multispool compressor system may occur in any of the individual compressors and the compressor in which it occurs may be different for various operating conditions.

Surge may develop into a system of multiple stall cells. The surge process includes a flow breakdown, ensuing surge wave, depressurization, and eventual reflowing and repressurizing of the compressor and associated duct volumes. In some cases, such as when conditions which drove the compressor beyond its stability boundary are not removed, the system surge can be cyclic in nature.

The surge wave is a pressure pulse, generated by the flow breakdown, which travels forward as a compression wave (overpressure) and travels rearward as an expansion wave from the origin of the surge. The identification of the origin of the surge wave therefore is sufficient for defining the compressor which caused the major flow breakdown.

Detection and Classification of Instabilities

Pressure-time histories of the data acquired from the high response instrumentation serve as the basis for analysis of the instabilities. A single history provides a record of the pressure fluctuations as a function of time at one spatial location in the compression system. Comparison of the time histories recorded at different locations in the system reveals the time of initial instability and events leading to surge, as well as the surge itself. Rotating stall and surge were the two kinds of instability identified in the TF30 data.

Because of the localized extent and the rotational nature of the rotating stall cell, its presence is indicated by a periodic fluctuation in the pressure time history. As shown in Figure 1, the direction of the fluctuation is dependent on the location of the pressure probe in relation to the cell. A downstream probe shows a reduction in pressure due to the stalled condition of the flow, while the probe upstream of the cell shows a pressure increase due to the back pressure (flow blockage) effect of the cell on the incoming flow. The positions of the upstream and downstream probes define the axial location of the cell within the compression system. The rotational frequency of the cell, used to identify stall type, can be determined by plotting the circumferential location of the probe versus the time of the pressure fluctuation as shown in Figure 2. This method is particularly useful if the duration of the cell is less than one cell revolution, which in turn requires that probes at more than one circumferential location be included on the plot.

The surge characteristics exhibited in the pressure time histories are generated by a major flow breakdown at some axial location in the compression system. Probes downstream of the breakdown point show a rapid decrease in pressure, the upstream probes show a rapid pressure increase. These characteristics, shown in Figure 3, differ from rotating stall characteristics primarily in magnitude of the pressure variations, the surge pressure variations being substantially larger.

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Previous Pratt & Whitney Aircraft analyses of TF30 compressor instabilities have shown that the origin of compression system instabilities as well as the occurrence and order of events leading to surge vary on a case to case basis. It has therefore been found useful to classify each case according to the origin and types of instability leading to surge. This practice was continued in the analysis of the NASA/TF30 data and definitions of the surge event classifications are given below to clarify results of the analysis.

- Initial Instability in Fan

Type 1 - Rotating stall initiating in fan root followed by rotating stall in the low pressure compressor (LPC), followed by a surge wave emanating from the high pressure compressor (HPC).

Type 2 - Rotating stall initiating in fan tip (coupling with fan duct resonant frequency) followed by depressurization of the core.

*Type 3 - Rotating stall initiating in the fan root followed by rotating stall in LPC, followed by rotating stall in HPC, followed by a surge wave emanating from HPC.

- Initial Instability in LPC

Type 1 - Rotating stall initiating in LPC followed by a surge wave emanating from the HPC.

Type 2 - Rotating stall initiating in LPC followed by a surge wave emanating from the LPC.

Type 3 - Rotating stall initiating in LPC followed by HPC rotating stall and followed by a surge wave emanating from HPC.

- Initial Instability in HPC

Type 1 - Rotating stall initiating in HPC followed by surge wave emanating from HPC.

Type 2 - Surge wave emanating from HPC.

Instrumentation

Figure 4 shows the five axial high response instrumentation stations that were installed for the NASA/TF30 test. Coverage of the entire fan face was provided by the inlet pressure rakes used in the inlet distortion analysis. Instrumentation internal to the compression system was located in the core flow path which in turn permitted positive identification of instabilities occurring in this region. The core path was instrumented such that the low pressure compressor was isolated into three blade row groups while the high pressure compressor was isolated as a complete unit. The groups in the low compressor were inlet guide vane (IGV) through rotor 3 (fan roots), stator 3 through rotor 6, and stator 6 through rotor 9.

*Note - New classification based on NASA test results.

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The internal instrumentation was circumferentially spaced in the manner shown in Figure 5. The operating probes provided data for at least one circumferential location at each axial station and were adequate for identification of rotating stall cells and determination of rotating frequency

Interpretation of Engine Data

Two methods of generating the pressure-time histories for analysis were explored. Initially, the histories were generated by analog tracing of unfiltered playback of the dynamic pressure components. Subsequently, the histories were generated by analog to digital conversion of filtered playback of the dynamic pressures. The latter method proved to be more desirable because the filtering provided better resolution of the compression system instabilities through attenuation of higher frequency activity. In addition, the digital output format permitted machine plotting of the pressure-time coordinates on grid paper, making reading of the time scale easier than on the analog traces. A real time filter cut-off frequency of 320 Hz and an effective cut rate of 1024 cuts/sec. were selected for the analog to digital process and produced the time histories shown on Pages 86 through 109 of Appendix D. These histories were used in the final analysis of the instabilities.

The time history plots were interpreted by initially identifying the system surge. From this point in time, preceding periods were examined to determine the time of initial instability as well as the location and types of instabilities preceding surge. In general, time histories of P_{12} showed only overpressure resulting from surge, therefore, the analysis was centered around the time histories of the internal engine pressures shown on Pages 86 through 109. The significant events occurring in each case were identified on each plot

Rotating stall cell frequencies were determined from the plots shown on Pages 110 through 130 of Appendix D where probe position was plotted versus time of cell indication. In some cases it appeared that the rotating cells were superimposed on top of the surge waves. When this occurred and cell duration prior to surge was short (less than one rotation) the post surge indication was used to improve resolution of the rotating frequency. This type of rotating stall can be observed in the pressure data for case 408 (Pages 91 and 92), where the presence of the low compressor rotating stall cell is indicated after the time of surge

The 22 surge cases which were analyzed spanned a low rotor corrected speed range of 6000 to 7050 rpm. Within this range, various configurations of bleed position and exhaust area settings were tested as shown in Table I.

Table I shows that the surge events were not identical on a case to case basis. Results which can be derived are as follows

1. Positive identification of instabilities in the core flow path were made in all surge events.
2. Five different sequences of events leading to and including surge were found
3. In all cases except 519, instabilities were identified in the core low spool compressor prior to surge. Those initial instabilities were identified as rotating stall
4. High compressor surge was the only instability positively identified in case 519. This case was the only one analyzed for operation with both 7th and 12th bleeds open.

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- 5 In five cases, high compressor rotating stall was identified prior to surge. These five cases occurred at corrected low rotor speeds of 6500 rpm and lower as shown in Figure 6
- 6 In the majority of cases (16 of 22) the initial instability was identified in the Stator 3 through Rotor 6 (Sta 2 3 - 2.6) row group. These cases covered the full range of low rotor speeds, see Figure 6
7. The initial instability was identified in the IGV through Rotor 3 (Sta 2 0 - 2.3) fan root row group in four cases. These occurred at corrected low rotor speeds of 6180 rpm and below, see Figure 6
8. In case 438 the initial instability was first identified in the Rotor 6 through Rotor 9 (Sta 2.6 - 3.0) group at a corrected low rotor speed of 6970 rpm, see Figure 6

The surge classification system defined from previous TF30 testing provided the key to interpreting the NASA data. The instabilities noted in the data were easily linked to events described in the surge classification system to identify the origin and type of instability. Case 497 was the only one which presented any problem because previous Pratt & Whitney Aircraft experience did not show high compressor rotating stall subsequent to a rotating stall initiated in the fan root. No reason was seen to rule out the possibility of this occurrence and a new classification was defined (Fan/Type 3) on the basis of the NASA data.

Case 519 also stood out as different from the other surge cases because no instability in the low spool was clearly evident prior to surge. This case was unique in that it was the only one analyzed for operation with both 7th and 12th stage bleeds open. Although it was not evident in the data, it was felt that an undetected short duration rotating stall in the low spool compressor should not be completely ruled out as a cause of the surge.

The next area of interest in this analysis was the location of initial instability. The matrix shown in Figure 6 was used to identify trends of results for cases where instability initiated in the low compressor. The matrix included the major engine variables which were exercised during the test and in turn affected the compression system matching.

A significant majority of the instabilities (16 of 21) initiated in the Stator 3 through Rotor 6 group, instabilities originated at this location over the full range of tested rotor speeds. In contrast, instabilities which initiated in the fan roots were concentrated in a speed range of 6000 to 6180 $N_1/\sqrt{\theta_{12}}$ while the one instability that initiated in the Stator 6 through Rotor 9 row group occurred at 6970 $N_1/\sqrt{\theta_{12}}$. The cases where instability initiated outside the Stator 3 through Rotor 6 group qualitatively displayed the expected movement of stalled operation from the front to the rear of the compressor with increasing speed, however, the trend was not felt to be conclusive. In contrast, consideration of all 21 cases led to the conclusion that the location of initial instability was somewhat random and therefore unpredictable. It did appear that the most probable location for initial instability was the Stator 3 through Rotor 6 group.

Compressor Row Matching

The test results showed the low compressor to be critical to engine stability. For purposes of augmenting analysis of the engine test data, a computer model of the fan/low compressor was exercised to determine which blade row in the core compressor might be critical to its stability.

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The model was composed of average $\theta - \psi$ characteristics defined for each blade row of the fan and core compressor. The characteristics were derived from data acquired during rig testing of a Bill of Material TF30-P-3 fan/low compressor. The model was operated by input of the corrected low rotor speed, corrected total airflow and corrected core flow at the fan inlet station, output from the model included both the overall and row operating conditions.

The operating conditions investigated with the model were limited to corrected low rotor speeds of 7000 rpm and above due to the range of definition of the row characteristics; in addition, the fan match was restricted to conditions encountered during static operation because testing of the rig was performed at ambient inlet and discharge conditions. With these limitations in mind, the row operating conditions were examined for operation of the low compressor near the rig surge line with the fan operating on a line near the level experienced during static operation of the engine. A range of rotor speeds of 7000-9000 rpm was investigated as shown in Figures 7 and 8 so that the trend of shifts in row operating points could be qualitatively extrapolated to the 6000 to 7050 rpm corrected speed range covered by the NASA engine test.

Assessment of the low compressor row operating conditions was made by evaluating the position of each rotor and stator row operating point relative to the peak pressure rise of its characteristic. As illustrated in Figure 9, the point was identified as operating on either the choked or stalled side of the $\theta - \psi$ characteristic and the percentage deviation from peak pressure rise ($\% \Delta \psi$) was calculated. The results were plotted as a function of the row location and inlet corrected rotor speed, as shown in Figure 10, to give an overview of the predicted operating conditions within the compressor.

The results of this analysis showed that operation of the low compressor 3rd stator was predicted to progress toward a relatively highly stalled condition as rotor speed was reduced from 9000 to 7000 rpm (see Figure 10), qualitative extrapolation of these results therefore predicted the 3rd stator to be the predominantly stalled row in the 6000 to 7050 rpm speed range tested.

Interpretation of these results required consideration of the findings made from the internal compression system data taken during the test. The data showed that in the majority of cases, the instabilities initiated in the Stator 3 through Rotor 6 blade row group. This, coupled with the model predictions, left the 3rd stator as a row suspected of frequently influencing the stability limit of the compressor, in contrast, it could not be said that instability universally initiated at this location because of the exceptions demonstrated in the test data, namely the instabilities which initiated in the fan roots and in the rear stages. In addition, Pratt & Whitney Aircraft testing of the TF30 compression system has also shown that instabilities initiate at different locations (stages) in the compressor under apparently similar operating conditions

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Application of Previously Developed Methodology

Early testing of the TF30/F111 installation led to development of a factor which evaluated the severity of the predominantly 180° distortion generated by the aircraft inlet. The factor, K_{D2} (see Appendix A) was initially implemented in both the engine development and flight test programs and evaluated the engine stability limits on the basis of low response data. This approach correlated the development test and flight test results separately, but indicated the stability limits of the engine to be lower in the aircraft than in the test cell.

As the aircraft flight testing progressed, it became apparent that the inlet flow environment contained significant nonsteady, high frequency characteristics. These were known to degrade the apparent stability limits of the engine indicated by the K_{D2} level calculated from low response data. It was necessary to establish a method of accounting for the high frequency activity (turbulence) and the K_{D2} - turbulence approach was developed. A sample surge limit curve is illustrated in Figure 11. The approach correlated the stability limits of the engine at constant airflow in terms of the low response K_{D2} and turbulence level $\Delta \bar{P}_t \text{ rms} / P_t$ (0-150 Hz). The cutoff frequency of 150 Hz for turbulence was confirmed by testing of a fan/low compressor rig.

The stability limits of the engine in terms of K_{D2} and turbulence were derived from cell testing of the engine behind a distortion valve and turbulence generator which are shown in Figure 12. The distortion valve defined stability limits at conditions of low turbulence and the turbulence generator at conditions of relatively high turbulence. During this testing the distortion patterns were maintained at one per revolution conditions so that the aircraft environment could be simulated as closely as possible. (See Figure 13.)

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A METHOD TO ACCOUNT FOR VARIATION OF AVERAGE COMPRESSOR INLET PRESSURE DURING INSTANTANEOUS DISTORTION ANALYSES

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Abstract

Instantaneous distortion analyses compare a time-varying value of an index (or "surge margin used up") with a critical level (or "available surge margin" of the compressor) to determine inlet-engine compatibility. Unless freestream conditions or propulsion system controls are changing, it is generally assumed that the available surge margin of the compressor is accurately determined from the steady-state operating point. Results are presented which show that variations of average compressor inlet pressure may occur without changes in freestream conditions or propulsion system controls. The volume dynamics of the compressor will cause these pressure variations to be attenuated and delayed by the time they reach the exit. This will cause the compressor pressure ratio (and available surge margin) to vary with time. A method is presented to calculate the available surge margin as a function of time and incorporate it into an instantaneous distortion analysis. Results show that inlet pressure variations which cause only a small change at the compressor exit can cause a significant variation in the available surge margin.

Introduction

The problems associated with inlet-engine compatibility have received a great deal of attention in recent years (cf ref 1). Wind tunnel and flight test programs have shown that losses in compressor surge margin cannot be predicted using only steady-state total-pressure measurements. Correlations between available compressor surge margin and various steady-state total-pressure distortion parameters have been obtained in engine tests where the magnitudes of the total-pressure fluctuations are small. Inlet-engine compatibility has been investigated by applying these steady-state distortion correlations to a set of fluctuating total-pressure measurements on a quasi-steady, or instantaneous, basis. Since the compressor does not appear to be sensitive to pressure fluctuations above a certain frequency (on the order of engine rev/sec), the total-pressures are normally filtered before instantaneous distortion values are calculated.

To apply the steady-state correlation at each instant in time, it is necessary to calculate both the distortion parameter and the available surge margin as functions of time. Unless freestream conditions or propulsion system controls are changing, it is generally assumed that the available surge margin of the compressor does not change with time and is accurately determined from the steady-state operating point.

Results are presented which show that variations of average compressor inlet pressure may occur without changes in freestream conditions or propulsion system controls. The volume dynamics of the compressor and its terminating volume cause these pressure variations to be attenuated and delayed by the time they reach the exit. The amount of attenuation and delay influence the variation of compressor pressure ratio (and available surge margin) with time.

The objective of this paper is to show that the use of the time-varying value of available surge margin can significantly change the results of an instantaneous distortion analysis. The distortion index used in this paper is of a general form and similar to many in use at the present time. Therefore, the results presented are expected to be indicative of those that would be obtained using many of the current indices.

Data from a supersonic wind tunnel test of an axisymmetric mixed compression inlet and J85-CE-13 turbojet engine are analyzed. Time-varying measurements of compressor-face total pressure are used to calculate exit total-pressure variations (using a compressor transfer function), because adequate exit pressure measurements were not available. Instantaneous values of a distortion index are calculated and compared with both the time-varying and steady-state levels of available surge margin. The differences between the two comparisons are examined to determine the effect of fluctuations in average inlet pressure.

Summary of Results

A method was developed to account for the variation of average compressor inlet pressure during instantaneous distortion analyses. The pressure ratio and surge margin of the J85-13 compressor were determined as a function of time. Compressor discharge total-pressure was calculated by applying a transfer function to the measured inlet total-pressure. Supersonic wind tunnel test data from an inlet-engine combination were used to determine the influence of a time variation in available surge margin on an instantaneous distortion analysis. These data were recorded while freestream conditions and propulsion system controls were not changing. An instantaneous distortion index (surge margin used up) was compared with both the steady state and time-varying values of available surge margin.

The following conclusions were drawn:

1. Small variations of the average compressor inlet total-pressure can have a significant effect on available surge margin. Compressor inlet total-pressure variations of ± 2.5 percent caused exit total-pressure variations of ± 0.75 percent which combined to produce ± 10 percent changes about a 0.16 level of available surge margin.

2. The effect of full-face pulses of compressor inlet total-pressure can be accounted for by an instantaneous distortion analysis which includes the time-variation of available surge margin. In the limiting case of zero distortion, the analysis would predict surge when the available surge margin decreased to zero.

3. When a steady-state distortion correlation is applied to time-varying data on a quasi-steady or instantaneous basis, both the distortion index (margin used up) and the available surge margin should be determined as functions of time. The available surge margin is generally assumed to remain constant when freestream conditions and propulsion system controls are not changing. The wind tunnel data that were examined did not support this assumption. This caused the results of an instantaneous distortion analysis using the unsteady level of available surge margin to be significantly different from the results of an analysis using the steady state level of available surge margin.

4. If the average inlet pressure variations are of small enough amplitude and/or high enough frequency, the compressor exit pressure changes will be negligible. In this case, the simplified expression

$$\left[1 - \frac{P_{2,\min}(t)}{\bar{P}_{2s}} = s \text{ available surge margin} \right]$$

will be satisfied at the same time surge is predicted by the complete time-dependent distortion analysis. At times when surge is not predicted, the simplified analysis will not accurately express the difference between margin used up and margin available, unless the inlet pressure variations, $\bar{P}_2(t)$, are also negligible.

Results show that exit pressure variations, $\bar{P}_3(t)$, as small as 0.75 percent will cause the simplified expression to fail to predict surge.

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MODELING AND ANALYSIS OF TF30-P-3 COMPRESSOR

SYSTEM WITH & WITHOUT INLET PRESSURE

DISTORTION

By

R. S. Mazzawy

SUMMARY

The analysis of circumferential inlet flow distortion data taken by NASA-Lewis Research Center personnel from testing of a Pratt & Whitney Aircraft TF30-P-3 afterburning turbofan engine is presented herein. The distortion was generated by a NASA-developed air jet device which was capable of varying the amplitude, circumferential extent, and circumferential position of a low total pressure region. The data included detailed steady state instrumentation measurements for distortion levels below those required to stall the engine, as well as steady state and high response instrumentation measurements to document engine stall.

Data analysis was primarily performed through the use of the P&WA-developed multiple segment parallel compressor model. This model exists as a computer program and provides a detailed blade row by blade row definition of the distorted flow field for the TF30-P-3 compressor system. The required pressure and temperature rise characteristics for each blade row were provided from previous P&WA compressor component rig testing. The results of this program were compared in detail with available pressure and temperature measurements at two low rotor speeds: 7400 rpm and 8600 rpm. Generally good agreement was obtained between the model calculations and the test data. The predicted attenuation and circumferential movement of the distorted region through the compressor were verified by the data. An analysis of the same data by NASA-LeRC personnel was presented in Reference 1 without the assistance of the model. Some of the conclusions reached in that data analysis are also included in this report for comparison purposes.

The engine stall data was analyzed on the basis of classical two-segment parallel compressor theory. A comparison is made between the distortion level which was observed to cause engine stall and the distortion level predicted by using parallel compressor theory. In general, the predicted level was lower than that which was measured experimentally. On the basis of the prediction, however, an estimate was made of the origin of the stall which was in reasonable agreement with the stall site determined from high response records. The data analyzed covered a low rotor speed range from 7300 rpm to 8700 rpm. It was determined in each case that stall was initiated in the front stages of the low pressure compressor.

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DATA ANALYSIS AND RESULTS

UNIFORM INLET DATA

Uniform inlet data from NASA-LeRC tests were analyzed to verify the applicability of the P&WA blade row performance characteristics from TF30 compressor rig testing. The undistorted data analysis revealed that some measurements which are critical for the determination of engine bypass ratio were made with an insufficient number of instrumentation locations. In order to correct this deficiency the available data were supplemented by similar measurements made by P&WA on a number of TF30 engines. The complete data analysis during this phase verified that the blade row characteristics provided an adequate representation of the TF30 engine performance. An exception was the speed-airflow relationship for the fan, but this was corrected by modifying the characteristics to reflect slightly higher total airflow capacity for the engine tested at NASA LeRC relative to the component rig results.

The P&WA characteristics were derived from rig testing with different instrumentation and different Reynold's Number levels than were used in the NASA engine testing. The use of engine airflow for cooling purposes is another difference between the two tests. These differences resulted in real and apparent flow capacity shifts and were necessarily considered when the applicability of the characteristics was evaluated. The most convenient procedure for this task was to adjust the engine data for these differences and make comparisons with compressor rig overall performance maps. The engine core airflow calculation was an important part of this procedure and particular attention was given to using the most accurate technique available.

The TF30-P-3 turbofan is a mixed flow engine since the engine core and fan bypass flows mix together and exit through a common tailpipe and nozzle. This type of configuration precludes the separate measurement of engine and fan flows as is done in compressor rig testing. It is customary, therefore, to measure the total airflow and to calculate the engine bypass ratio (fan duct flow/engine flow) using other measured engine parameters. The calculation used for this purpose is based upon an energy balance between the compressors and turbines, the fuel and air flow entering and the flow leaving the engine. The equations as well as the measured and assumed parameters required for this calculation are outlined in Figure 2.

Initial calculations of engine bypass ratio for NASA LeRC uniform inlet data made on the basis of an energy balance between the compressors, burner, and turbines indicated unusual flow characteristics. Engine flow was calculated to increase as power setting was reduced in the intermediate operating range. It was initially suspected that the assumed primary burner efficiency used for this calculation was in error at reduced power. A thorough investigation revealed that the source of the problem was the use of only two turbine exit temperature rakes. It can be seen in Figure 3 that the right side and left side rake readings are significantly different in the intermediate speed range. An investigation of the distortion data showed a similar problem with the turbine exit temperature measurement over this range. The difference in temperature measurement is attributed to the change in circumferential swirl of the air through the turbine with rotor speed. As swirl changes with speed, the different rakes can be exposed to locally colder or hotter regions in the burner.

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exit profile which are not representative of true average conditions. For this reason, experimental engines tested at P&WA normally have six turbine exit temperature rakes to obtain accurate data. A comparison of the NASA LeRC data with other available engine data indicates that the left side rake measurement is closer to the actual temperature than the average of the two rakes. The left side temperature was therefore corrected to represent an average temperature using other engine experience. Bypass ratios were recalculated and the results were found to be more consistent with the other engine and compressor rig experience.

In summary, the following analysis has been conducted using the NASA LeRC uniform inlet data:

- 1 The engine bypass ratio has been calculated on the basis of an energy balance between the compressors, burner and turbines. Inconsistencies in turbine exit temperature measurements were caused by limited instrumentation coverage. These inconsistencies have been resolved on the basis of other TF30-P-3 engine data with more extensive instrumentation coverage.
- 2 Parasitic airflow (67% for cooling purposes) is removed from the main airflow at station 3.0 (high pressure compressor inlet). This reduction in airflow was accounted for in determining high pressure compressor performance.
- 3 The large (for structural integrity) station 3.0 total pressure rakes cause a known back-pressure effect which raised the indicated total pressure measurement approximately 4% above the true level. The higher Pt 3.0 was accounted for to accurately determine the relative airflow matching and performance of the low pressure compressor and high pressure compressor. Small adjustments in total pressure level for differences in radial instrumentation between engine and rig tests were similarly accountable.
- 4 Different levels of Reynold's Number existed between the rig and engine tests. Flow capacity shifts due to these differences were applied for the fan, low pressure and high pressure compressors.

The resulting adjusted engine data has been plotted on the rig performance maps in Figures 4 through 6. The fan operating line, (Figure 4), falls below the normal sea level operating line because the NASA test was run with a choked exit nozzle, which has the same effect as running unchoked with a larger nozzle area. It is also observed that the NASA total corrected airflow is somewhat higher than that measured in the rig test. The difference, about 1.5%, can be attributed to engine-to-engine variation, and measurement error tolerances.

The low pressure compressor operating line is above the normal operating line, see Figure 5. This result is characteristic of the TF30 engine with a low fan operating line. Relative speed-flow differences at high speed are also expected because of the influence of the bypass ratio (which is relatively higher with the choked jet nozzle) on the low pressure compressor. The agreement of the data on the high pressure compressor map is quite good as shown on Figure 6.

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Predictions of the engine data using P&WA's compressor characteristics are also shown on the figures. Fan predictions were based on compressor characteristics with the fan blade rows modified to reflect the 1.5% greater total corrected airflow measured by NASA. These predictions automatically include the effects of bypass ratio on the low pressure compressor map. The P&WA characteristics are seen to be quite adequate for use in predicting the NASA data for this contract. It should be noted that data were not available from the NASA LeRC engine test to substantiate the level of the rig-generated stall lines shown on the three maps. However, P&WA experience with TF30 engine and dual spool rig testing (Reference 6) supports the assumption that rig and engine stall lines are synonymous at the same Reynolds Number.

DISTORTION ATTENUATION

The circumferential distortion attenuation data analysis done under this contract is based upon the P&WA developed multiple segment parallel compressor model. This model provides a detailed prediction of the distorted flow field which is used for the purpose of interpreting the measured pressure and temperature distortion profiles at the different measurement planes within the engine. The data analysis of Reference 1 was done without the aid of such a calculation. Accordingly, some of the conclusions drawn in that analysis are different than those reached in this present work. These differences will be commented on later in the data analysis section.

Data Analysis

The NASA LeRC TF30-P-3 turbofan tests were conducted to evaluate the response of this engine to circumferential inlet total pressure distortion. The air jet device used to produce the circumferential distortion is described in detail in NASA TMX-1946. Rotation of the distortion in 60° increments provided detailed definition of the distorted flow field. 180° extent distortion rotation data were obtained at two locations on the engine operating line: one at approximately 7400 rpm, the other at approximately 8600 rpm. The data were normalized by NASA LeRC for variations in inlet total pressure. Additionally, the P&WA data analysis consisted of

- 1 averaging data over the six distortion positions,
- 2 calculating the compressor performance parameters,
- 3 executing the P&WA multiple segment parallel model compressor program with appropriate input from the distortion rotation data including inlet pressure profile,
- 4 comparing the compressor performance parameters from the P&WA compressor model predictions with those calculated from the test data and with P&WA compressor rig experience,
- 5 comparing the flow field profiles as measured and as predicted by the P&WA compressor model at the axial locations used by NASA LeRC to measure flow properties within the compression system.

On this basis, the best estimate for the stall site would be either S3, R5 or R6 at the lowest speed (7300 rpm) and R5 or R6 at the other speeds. Diffusion factors calculated for the high pressure compressor are not high as can be seen in Figure 55. The engine design point levels are again shown for comparison. The low-levels verify that the high pressure compressor did not initiate the stall. The high pressure compressor, however, will be additionally distorted by the rotating stall from the low pressure compressor. The additional loading which the rotating stall imposes on the high pressure compressor and which causes the final engine surge is not reflected in the parallel compressor calculation.

A comparison shows that the stall sites from the high response records and the diffusion factor analysis are in qualitative agreement. It is difficult to estimate the exact stall location because of the distribution of the high response instrumentation. Furthermore, the diffusion factor analysis is based upon a mean diameter calculation and does not reflect radial variations in blade loading. The significant point is that basic parallel compressor theory gives a reasonable prediction for the origin of stall for the TF30 engine. This was true despite the fact that the predicted distortion level required to stall the engine was in disagreement with the test data.

NASA CR-134996

SUMMARY OF RESULTS

The data analyses performed on the basis of the multiple segment and classical parallel compressor model predictions for attenuation and sensitivity with 180° circumferential pressure distortion are summarized as follows

- 1 The square wave inlet total pressure distortions result in non-square inlet velocity distortions. The primary reasons for this are the inlet air angle variation caused by circumferential flow redistribution upstream of the fan and unsteady flow effects
- 2 Circumferential crossflow within the compression system resulted in increased attenuation in the front stages
- 3 The low mass flow region moves circumferentially as it travels through the compression system by an amount equal to the swirl of the acoustic path. This amounts to approximately 10-20 degrees in the fan and 65 degrees in the core in the direction of rotor rotation. The static and total pressure distortion swirl about the same distance
- 4 The total temperature distortion is primarily created by the attenuation within the front stages. The temperature distortion swirls approximately 35 degrees in the fan and 165 degrees in the core in the direction of rotor rotation. This is comparable to the circumferential displacement of a fluid particle as it passes through the TF30 compression system
- 5 The static pressure uniformity at station 3.0 indicates that the low and high pressure compressors are decoupled by the crossflow cavities at station 3.0. The good prediction of the distortion attenuation with this station 3.0 boundary condition verifies the decoupling

ASME Paper 81-GT-211

PRINT 16/2/1-158 TERMINAL=89
 81A30100# ISSUE 12 PAGE 1928 CATEGORY 2 RPT#:
 ASME PAPER 81-GT-211 81/03/00 10 PAGES
 UNCLASSIFIED DOCUMENT

UTTL: Theoretical study of flow instabilities and inlet distortions in axial compressors
 AUTH: A/FERRAND, P.: B/CHAUVIN, J. PAA: B/(Aix-Marseille I, Universite, Marseille, France) SAP: MEMBERS, \$2.00: NONMEMBERS \$4.00. American Society of Mechanical Engineers, Gas Turbine Conference and Products Show, Houston, Tex., Mar. 9-12. 1981, 10p
 MAJS: /*COMPUTATIONAL FLUID DYNAMICS/*FLOW DISTORTION/*FLOW STABILITY/*INLET FLOW/*LAPLACE TRANSFORMATION/*TURBOCOMPRESSORS
 MINS: / COMPRESSOR BLADES/ DYNAMIC RESPONSE/ MACH NUMBER/ PRESSURE DISTRIBUTION
 ABA: (Author)
 ABS: This paper describes a method of evaluation of the single and multistage compressors response to steady and unsteady inlet distortions. It allows also the evaluation of the appearance of unstable regimes and their characterization (rotating stall and surge). It is based on a linearized approach using mean line calculations. The compressor is considered as a series of vaned and vaneless spaces and the corresponding equations are solved by use of Fourier series for time independent variables and by Laplace's transform for time dependent variables. An analogy between the compressor's response and a servo-mechanism is developed using Nyquist's diagram. Results are compared with experimental data which prove the validity of the approach. A parametric study indicates which parameters can be modified to improve the flow stability.

Journal of Fluid Mechanics
Vol. 98, June 12, 1980

80444862* ISSUE 19 PAGE 3451 CATEGORY 2
80/06/12 36 PAGES UNCLASSIFIED DOCUMENT

UTTL The effect of finite turbulence spatial scale on the amplification of turbulence by a contracting stream

AUTH A/GOLDSTEIN, M E . B/DURBIN P A PAA A/(NASA Lewis Research Center, Cleveland, Ohio); B/(Cambridge University, Cambridge, England)

CORP National Aeronautics and Space Administration Lewis Research Center, Cleveland, Ohio Cambridge Univ (England)

Journal of Fluid Mechanics, vol 98, June 12, 1980, p. 473-508

MAJS /*BOUNDARY LAYER SEPARATION/*FLOW DISTORTION/*FLOW VELOCITY/*FREE FLOW/*INLET FLOW/*TURBULENT FLOW

MINS / ATMOSPHERIC TURBULENCE/ BOUNDARY LAYER FLOW/ ENGINE NOISE/ HOMOGENEOUS TURBULENCE/ NOISE PREDICTION (AIRCRAFT)/ POTENTIAL FLOW/ SMALL PERTURBATION FLOW/ SPATIAL DISTRIBUTION/ VELOCITY DISTRIBUTION

ABA. A L W

ABS The turbulence downstream of a rapid contraction is calculated for the case when the turbulence scale can have the same magnitude as the mean-flow spatial scale. The approach used is based on the formulation of Goldstein (1978) for turbulence downstream of a contraction, with the added assumptions of a parallel mean flow at downstream infinity and turbulence calculated far enough downstream so that the nonuniformity of the mean flow field has decayed, and by treating the inverse contraction ratio as a small parameter. Consideration is given to the large-contraction-ratio and classical rapid-distortion theory limits, and to results at an arbitrary contraction ratio. It is shown that the amplification effect of the contraction is reduced when the spatial scale of the turbulence increases, with the upstream turbulence actually suppressed for a contraction ratio less than five and a turbulence spatial scale greater than three times the transverse dimensions of the downstream channel.

AIAA Paper 80-1109

80A41182# ISSUE 17 PAGE 3110 CATEGORY 2 RPT#:
 AIAA PAPER 80-1109 80/06/00 11 PAGES UNCLASSIFIED
 DOCUMENT

UTTL Evaluation of a statistical method for determining
 peak inlet flow distortion using F-15 and F-18 data

AUTH A/STEVENS, C H , B/OLIPHANT R C , C/SPONG, E. D.
 PAA C/(McDonnell Aircraft Co., St Louis Mo)
 AIAA, SAE, and ASME, Joint Propulsion Conference,
 16th, Hartford, Conn , June 30-July 2, 1980, AIAA 11
 P

MAJS /*COST REDUCTION/*F-15 AIRCRAFT/*F-18 AIRCRAFT/*FLOW
 DISTORTION/*INLET FLOW/*STATISTICAL ANALYSIS

MINS / DESIGN ANALYSIS/ DYNAMIC PRESSURE/ ENGINE DESIGN/
 NOZZLE DESIGN/ PREDICTION ANALYSIS TECHNIQUES/
 PRESSURE SENSORS/ SCALE MODELS/ WIND TUNNEL TESTS

ABA (Author)

ABS Methods have been developed for significantly reducing
 the cost of determining inlet peak dynamic distortion
 values for advanced design purposes. These methods are
 not intended to replace the data acquisition and
 reduction systems required for final assessments of
 inlet/engine compatibility on aircraft development
 programs. However, they do satisfy the critical need
 for a prediction procedure for advanced design
 investigations that enables us to predict peak
 distortion levels using small scale models in small
 wind tunnels. Cost reductions are achieved by taking
 advantage of the statistical characteristics of the
 dynamic pressure and distortion data. Comparisons with
 measured distortion data show good agreement, thereby
 validating the statistical approach.

AIAA Paper 80-1108

8041181# ISSUE 17 PAGE 3241 CATEGORY 60 RPT#.
AIAA PAPER 80-1108 80/06/00 10 PAGES UNCLASSIFIED
DOCUMENT

UTTL An analog editing system for inlet dynamic flow
distortion DYNADEC - Past, present and future
AUTH A/MAROUS J J , B/SEDLOCK D PAA A/(USAF,
Aeronautical Systems Div , Wright-Patterson AFB, Ohio)
 , B/(USAF, Wright Aeronautical Laboratories,
Wright-Patterson AFB, Ohio)
AIAA, SAE, and ASME, Joint Propulsion Conference,
16th, Hartford, Conn., June 30-July 2, 1980. AIAA 10
p
MAJS. /*ANALOG COMPUTERS/*DATA REDUCTION/*ENGINE INLETS/*
FLOW DISTORTION/*INLET FLOW
MINS: / DYNAMIC PRESSURE/ FIGHTER AIRCRAFT/ HYBRID COMPUTERS
/ PARALLEL PROCESSING (COMPUTERS)
ABA: V.T
ABS. An analog/digital (hybrid) editing system DYNADEC
(Dynamic Data Editing and Computing) used to screen
inlet dynamic pressure distortion data is described
An overall configuration is presented, some of the
improvements that have been made over the past decade
are pointed out, and some future changes and uses of
the system are discussed

Journal of Mechanical Engr.
Science, Vol. 22, Feb. 1980

80A40420 ISSUE 16 PAGE 2966 CATEGORY 34
80/02/00 2 PAGES UNCLASSIFIED DOCUMENT
UTTL Generation of streamwise vorticity in an asymmetric
swirling flow
AUTH A/GREITZER E M . B/HAWTHORNE W R PAA A/(MIT,
Cambridge, Mass). B/(MIT, Cambridge, Mass .
Cambridge University, Cambridge England)
Journal of Mechanical Engineering Science, vol 22,
Feb 1980, p 41, 42
MAJS /*DUCTED FLOW/*INLET FLOW/*NONUNIFORM FLOW/*
OSCILLATING FLOW/*THREE DIMENSIONAL FLOW/*
TURBOCOMPRESSORS/*VORTICITY
MINS / FLOW CHARACTERISTICS/ FLOW DISTORTION/ FLOW GEOMETRY
/ SECONDARY FLOW/ TURBULENT FLOW
ABA V.L
ABS. The behavior of a circumferentially nonuniform
swirling flow, which is of interest in connection with
the problem of the response of axial compressors to
inlet flow distortion, is examined. An analogy is
demonstrated between the behavior of classical
secondary flow, such as flow in a bent duct, and the
inherently three-dimensional effects that occur in an
asymmetric swirling flow

AIAA Paper 80-1110

80A38918# ISSUE 16 PAGE 2909 CATEGORY 7 RPT#
AIAA PAPER 80-1110 80/06/00 11 PAGES UNCLASSIFIED
DOCUMENT

UTTL An evaluation of statistical methods for the prediction of maximum time-variant inlet total pressure distortion

AUTH A/SAILERS, M E PAA A/(ARO Inc Arnold Engineering Development Center Arnold Air Force Station, Tenn)
AIAA, SAE and ASME, Joint Propulsion Conference 16th, Hartford, Conn June 30-July 2, 1980, AIAA 11 p

MAJS /*INLET PRESSURE/*PERFORMANCE PREDICTION/*PRESSURE DISTRIBUTION/*STATISTICAL ANALYSIS/*TIME DEPENDENCE/*TURBINE ENGINES

MINS / AIRCRAFT STRUCTURES/ ENGINE DESIGN/ FLOW DISTORTION/ INLET FLOW/ PERIODIC VARIATIONS/ PRESSURE MEASUREMENTS / STRUCTURAL DESIGN

ABA A T

ABS. The paper presents an evaluation of statistical methods for the prediction of maximum time-variant inlet total pressure distortion. Of the three methods investigated the Motycka method shows the most promise, the predicted distortion values and patterns agree with those measured however, the method must be modified to account for the nature of the inlet total pressure fluctuations. The Jacobs method predicts distortion values with reasonable accuracy, but the ADC requirement makes the method unsuitable for the most inlet programs. The Melick method is recommended for use during early subscale model inlet tests for the determination of maximum time-variant distortion values, also this method can be used as an online analysis tool for configuration selection and test direction at any stage of aircraft development.

AIAA Paper 80-0386

80A20970*# ISSUE 6 PAGE 944 CATEGORY 2 RPT#
 AIAA PAPER 80-0386 80/01/00 12 PAGES UNCLASSIFIED
 DOCUMENT

UTTL An analytical and experimental study of a short
 S-shaped subsonic diffuser of a supersonic inlet

AUTH A/NEUMANN, H E . B/POVINELLI L A . C/COLTRIN, R.
 E PAA C/(NASA Lewis Research Center, Cleveland,
 Ohio)

CORP National Aeronautics and Space Administration Lewis
 Research Center, Cleveland Ohio
 American Institute of Aeronautics and Astronautics,
 Aerospace Sciences Meeting 18th Pasadena Calif .
 Jan 14-16 1980, 12 p

MAJS /*DIFFUSERS/*DUCTED FLOW/*FLOW GEOMETRY/*FOREBODIES/*
 INLET FLOW/*SUPERSONIC FLOW

MINS / COMPRESSIBLE FLOW/ FINITE DIFFERENCE THEORY/ FLOW
 DISTORTION/ MACH NUMBER/ PRESSURE DISTRIBUTION/
 SUPERSONIC WIND TUNNELS/ TURBULENT FLOW

ABA (Author)

ABS An experimental investigation of a subscale HiMat
 forebody and inlet was conducted over a range of Mach
 numbers to 1.4. The inlet exhibited a transitory
 separation within the diffuser but steady state data
 indicated reattachment at the diffuser exit. A finite
 difference procedure for turbulent compressible flow
 in axisymmetric ducts was used to successfully model
 the HiMAT duct flow. The analysis technique was
 further used to estimate the initiation of separation
 and delineate the steady and unsteady flow regimes in
 similar S-shaped ducts.

West Germany Paper

79A43480 ISSUE 18 PAGE 3506 CATEGORY 64 CNT#:
N00024-78-C 5384 79/05/25 7 PAGES UNCLASSIFIED
DOCUMENT

UTTL A spatial decay estimate for the Navier-Stokes
equations
AUTH A/ELCRAT, A R B/SIGILLITO, V G PAA A/(Wichita
State University Wichita, Kan). B/(Johns Hopkins
University Laurel Md)
Zeitschrift fuer angewandte Mathematik und Physik.
vol 30 May 25 1979 p 449-455
MAJS /CHANNEL FLOW/ FLOW DISTORTION/ FLOW GEOMETRY/
NAVIER STOKES EQUATION
MINS. / BOUNDARY VALUE PROBLEMS/ FLOW VELOCITY/ INLET FLOW/
INTEGRAL EQUATIONS/ VELOCITY DISTRIBUTION
ABA P.T H
ABS An expression is derived which gives the exponential
decay, in the distance from a fixed reference plane,
for an energy-type functional of solutions of the
Navier-Stokes equations. The primary motivation for
the work is in flows through channels. Since it is
assumed that the initial velocity is zero and that the
velocity vanishes on the boundary of the flow region
except on the intersection of the region with the
reference plane, the result gives information on the
downstream decay of the effect of a disturbance at the
inlet region on the rest state. The decay constant
depends in a complicated manner on the geometry of the
flow region.

AIAA Paper 79-0104

79A19533*# ISSUE 6 PAGE 927 CATEGORY 2 RPT#
 AIAA PAPER 79-0104 79/01/00 11 PAGES UNCLASSIFIED
 DOCUMENT

UTTL Reynolds number, scale and frequency content effects
 on F-15 inlet instantaneous distortion

AUTH A/STEVENS C H , B/SPONG E D , C/NUGENT, J
 D/NEUMANN H E PAA B/(McDonnell Douglas Corp ,
 St Louis Mo) C/(NASA, Flight Research Center,
 Edwards Calif), D/(NASA Lewis Research Center,
 Cleveland Ohio)

CORP McDonnell-Douglas Corp , St Louis, Mo , National
 Aeronautics and Space Administration Flight Research
 Center Edwards, Calif , National Aeronautics and
 Space Administration Lewis Research Center,
 Cleveland, Ohio
 American Institute of Aeronautics and Astronautics,
 Aerospace Sciences Meeting, 17th New Orleans, La.,
 Jan 15-17 1979 11 p

MAJS /*BANDPASS FILTERS/*F-15 AIRCRAFT/*FLOW DISTORTION/*
 FREQUENCY DISTRIBUTION/*INLET FLOW/*REYNOLDS NUMBER

MINS / DATA BASES/ INTAKE SYSTEMS/ PRESSURE DISTRIBUTION/
 PRESSURE RECOVERY/ SCALE MODELS/ SUPERSONIC INLETS/
 WIND TUNNEL TESTS

ABA (Author)

ABS. An inlet instantaneous distortion study program
 sponsored by NASA was recently completed using an F-15
 fighter aircraft Peak distortion data from subscale
 inlet model wind tunnel tests are shown to be
 representative of full-scale flight test peak
 distortion The effects on peak distortion are
 investigated for engine presence, Reynolds number,
 scale and frequency content Data are presented which
 show that (1) the effect of engine presence on total
 pressure recovery, peak distortion, and turbulence is
 small but favorable, (2) increasing the Reynolds
 number increases total pressure recovery, decreases
 peak distortion, and decreases turbulence, and (3)
 increasing the filter cutoff frequency increases the
 calculated values of both peak distortion and
 turbulence

ASME Paper 78-GT-154

79A10270# ISSUE 1 PAGE 2 CATEGORY 2 RPT# ASME
 PAPER 78-GT-154 78/10/00 11 PAGES UNCLASSIFIED
 DOCUMENT

UTTL Small perturbation analysis of nonuniform rotating
 disturbances in a vaneless diffuser

AUTH A/INOLE, M B/CUMIPSTY N A PAA B/(Cambridge
 University, Cambridge England)
 (American Society Of Mechanical Engineers Gas Turbine
 Conference and Products Show London, England Apr
 9-13, 1978.) ASME Transactions Journal of
 Engineering for Power vol 100, Oct 1978 p
 711-721

MAJS /*CENTRIFUCAL COMPRESSORS/*FLOW DISTORTION/*IMPELLERS
 /*ROTATING FLUIDS/*SMALL PERTURBATION FLOW/*VANELESS
 DIFFUSERS

MINS / INCOMPRESSIBLE FLUIDS/ INLET FLOW/ INVISCID FLOW/
 NAVIER-STOKES EQUATION/ PHASE SHIFT/ POTENTIAL FLOW/
 RADIAL FLOW/ STATIC PRESSURE

ABA (Author)

ABS The behavior of the distorted flow discharged from a
 centrifugal impeller within a vaneless diffuser is
 examined theoretically by assuming small disturbances
 to a main flow. The inlet static pressure distribution
 is found in the calculation, and allowance is made for
 circumferential nonuniformity in the relative flow
 angle. The flow is treated as incompressible and
 inviscid. The analysis shows that the decay of
 irrotational disturbances is more rapid with
 increasing disturbance wave number (e.g., more
 impeller blades), and that the effect of the main flow
 condition on this behavior is very small. With
 rotational disturbances, however, the decay is slower
 than in the irrotational case and the effect of wave
 number is less. However, the phase angle between
 radial and tangential velocity fluctuations is found
 to have a strong influence on the decay processes for
 rotational disturbances. It is shown that the present
 small-perturbation theory predicts results very
 similar to the Dean and Senoo (1960) theory for
 impellers with large blade numbers (over 20). For
 small numbers of blades the large circumferential
 nonuniformity in relative flow angle appears at
 smaller radii and the inaccuracy of the Dean and Senoo
 theory becomes pronounced.

AIAA Paper 78-970

78A43525 # ISSUE 19 PAGE 3406 CATEGORY 2 RPT#
 AIAA PAPER 78-970 CNT# NAS2 6901 NAS2-8878
 78/07/00 15 PAGES UNCLASSIFIED DOCUMENT

UTTL Estimating maximum instantaneous distortion from inlet
 total pressure rms measurements

AUTH A/MELICK H C , JR B/YBARRA, A H , C/BENCZE, D
 P PAA B/(Vought Corp , Dallas, Tex) C/(NASA,
 Ames Research Center, Moffett Field Calif)

CORP Vought Corp Dallas Tex , National Aeronautics and
 Space Administration Ames Research Center, Moffett
 Field Calif
 American Institute of Aeronautics and Astronautics and
 Society of Automotive Engineers, Joint Propulsion
 Conference 14th, Las Vegas Nev July 25 27, 1978.
 AIAA 15 p

MAJS /*AIRCRAFT ENGINES/*ENGINE INLETS/*FLOW DISTORTION/*
 INLET FLOW/*INLET PRESSURE/*PRESSURE MEASUREMENTS/*
 TURBULENT FLOW

MINS / CHANNEL FLOW/ DATA REDUCTION/ FLUID DYNAMICS/ GRAPHS
 (CHARTS)

ABA V P

ABS. In the present paper a new mathematical model of
 inlet turbulence is developed by application of basic
 fluid dynamics and statistical concepts. The model
 provides an understanding of the turbulent inlet flow
 as well as a means of describing the flow in
 quantitative terms. Specifically the maximum
 instantaneous distortion produced by inlet unsteady
 flow can be estimated by the simple measurement of rms
 data. Practical application of these techniques leads
 to a data/acquisition/reduction system that is at
 least one and maybe two orders of magnitude less
 expensive than conventional methods. Each data point
 can be reduced in terms of the mean strength of the
 turbulent vortices. By storing these two parameters
 (that are representative of the unsteady flow with the
 steady state information), the maximal instantaneous
 distortion can be reconstructed for other distortion
 factors at any time subsequent to the test.

AIAA Paper 77-876

77A38570 # ISSUE 17 PAGE 2825 CATEGORY 7 RPT #: AIAA PAPER 77-876
77/07/00 12 PAGES UNCLASSIFIED DOCUMENT

UTTL: Comparison of estimated with measured maximum instantaneous distortion using flight data from an axisymmetric mixed compression inlet - - - for YF - 12C aircraft.

AUTH: A/BRILLIANT. H. M.: B/BAUER, C. A. PAA: A/(U. S. Air Force Academy, Colorado Springs, Colo.): B/NASA Flight Research Center, Edwards, Calif.)

CORP: Air Force Academy Colo.: National Aeronautics and Space Administration. Hugh L. Dryden Flight Research Center, Edwards, Calif. American Institute of Aeronautics and Astronautics and Society of Automotive Engineers, Propulsion Conference 13th. Orlando, Fla., July 11-13, 1977. AIAA 12 p.

MAJS: /*DYNAMIC PRESSURE/*ENGINE INLETS/*FLOW DISTORTION/*YF-12 AIRCRAFT
MINS: / ANGLE OF ATTACK/ COMPUTER TECHNIQUES/ FLIGHT TESTS/ MACH NUMBER/
PREDICTION ANALYSIS TECHNIQUES

ABA: G.R.

ABS: YF- 12C flight-measured inlet dynamic distortion data are compared with predictions made on the basis of the method reported by Melick et al. (1976). The YF-12C aircraft is a twin engine aircraft capable of speeds above 3. The inlets have a translating spike to control the inlet throat area. A bypass system is used to control the terminal shock of the inlet for operation in the mixed compression mode. The dynamic data were obtained with the aid of 24 high frequency response total pressure sensors. The model of Melick et al. is discussed along with the computer program used to implement the model. It is found that the predictions of maximum instantaneous distortion are within 20 percent of the measured values which had been obtained at Mach numbers of 1.8, 2.1, 2.5, and 3.0.

International Symposium on Air
Breathing Engines 3rd. Munich,
West Germany, Mar. 1976
Proceedings

77A17260 ISSUE 5 PAGE 638 CATEGORY 7 76/00/00
19 PAGES UNCLASSIFIED DOCUMENT

UTTL Investigation of the flow pattern at the engine face
and methods of the flow pattern simulation at
supersonic flight speed

AUTH A/OGORODNIKOV, D A , B/IANCHUK, V A
In International Symposium on Air Breathing Engines,
3rd, Munich, West Germany, March 7-12, 1976,
Proceedings (A77-17226 05-07) Cologne, Deutsche
Gesellschaft fuer Luft- und Raumfahrt, 1976 p
703 721

MAJS /*AIRCRAFT ENGINES/*DUCTED FLOW/*FLIGHT SIMULATION/*
FLOW DISTORTION/*FLOW DISTRIBUTION/*INLET FLOW/*
SUPERSONIC AIRCRAFT

MINS / BOUNDARY LAYER SEPARATION/ DIFFUSERS/ PRESSURE
OSCILLATIONS/ SHOCK WAVES/ SPOILERS/ STEADY FLOW/
SUPERSONIC INLETS/ TIME DEPENDENCE/ TURBOJET ENGINES

ABA S D

ABS. Steady-state distortions and fluctuations of a
nonuniform time-dependent fluctuating flowfield at an
aircraft engine face at supersonic flight speed affect
significantly the engine operational stability. The
principal characteristics of this distortion flow and
the possibilities of its simulation are examined for a
normal shock wave in a model of slightly divergent
diffuser duct. The model is provided with transducers
for measurement of steady-state distortion flow and
total pressure fluctuations in different sections
downstream of the terminal shock wave. A similar
distortion flow study is performed for a cylindrical
duct with a smooth inlet and a spoiler of varying
height. Regions are identified where the flow pattern
corresponding to the flow downstream of the terminal
shock wave may be simulated by means of spoilers with
good approximation.

AIAA Paper 76-705

76A38219# ISSUE 18 PAGE 2776 CATEGORY 2 RPT #: AIAA PAPER 76-705
76/07/00 6 PAGES UNCLASSIFIED DOCUMENT

UTTL: Determination of maximum expected instantaneous distortion patterns from statistical properties of inlet pressure data.

AUTH: A/MOTYCKA, D. L. PAA: A/(United Technologies Corp., Pratt and Whitney Aircraft Div., East Hartford, Conn.) American Institute of Aeronautics and Astronautics and Society of Automotive Engineers, Propulsion Conference, 12th, Palo Alto, Calif., July 26-29, 1976. AIAA 6p.

MAJS: /*DATA REDUCTION/*FLOW DISTORTION/*GAS TURBINE ENGINES/*INLET PRESSURE/*STATISTICAL ANALYSIS

MINS: / DATA PROCESSING/ FLOW CHARTS/ POWER SPECTRA/ PROBABILITY DENSITY FUNCTIONS/ RANDOM PRECESSES/ SPECTRUM ANALYSIS/ STANDARD DEVIATION S.D.

ABA:

ABS: An inexpensive and time-saving procedure is proposed which uses random numbers to synthesize instantaneous inlet distortion in turbine engines from statistical properties of inlet pressure data. The statistical properties include amplitude probability density, standard deviation, mean and power spectral density. Determination of the statistical properties of each pressure can be done with simple meter readings; if more precision is desired a spectral analyzer may be used. Not only did the levels of synthesized distortion factors agree well with the test data, but pattern comparisons were excellent. It is concluded that maximum instantaneous distortion patterns can be accurately synthesized by random number processing and that power spectral density variations influence the distribution of extreme values of the distortion factors versus sampling times.

AIAA Paper 75-59

75A20264*# ISSUE 7 PAGE 913 CATEGORY 5 RPT#: AIAA PAPER 75-59 CNT#: NAS3-16815 NAS3-17842 75/01/00 10 PAGES UNCLASSIFIED DOCUMENT

UTTL: Design and testing of new center inlet and S-duct for B-727 airplane with refanned JT8D engines

AUTH: A/TING, C.T.: B/KALDSCHMIDT, G.: C/SYLTEBO, B.E. PAA: C/(Boeing Commercial Airplane Co., Seattle, Wash.)
American Institute of Aeronautics and Astronautics, Aerospace Sciences Meeting, 13th, Pasadena, Calif., Jan. 20-22, 1975. 10p.

MAJS: /*AIRCRAFT ENGINES/*BOEING 727 AIRCRAFT/*DUCTED FLOW/*ENGINE DESIGN/*ENGINE INLETS/*ENGINE TESTS

MINS: / AIR DUCTS/ AIR FLOW/ ANGLE OF ATTACK/ BOUNDARY LAYER FLOW/ FLOW DISTORTION/ INLET FLOW/ POTENTIAL FLOW/ PRESSURE RECOVERY/ VORTEX GENERATORS

ABA: (Author)

ABS: The work described in this paper was part of the NASA refan program. The airflow requirement of the refanned JT8D (-100 series) engine increased about 50% above that of the basic JT8D. This required a redesign of the center inlet and S-duct of the Boeing 727 airplane. The paper describes the design constraints for the S-duct and the analytic method used to define the lines of the new duct. Model tests that were conducted at static angle-of-attack and crosswind conditions are described with a variety of flow control devices. Test results showed that the new inlet and S-duct have a pressure recovery comparable to that of the existing inlet. By employing corotating vortex generators less flow distortion was obtained for the core region than the existing duct has with its counter rotating vortex generators.

AIAA Paper 74-1197

75A10342# ISSUE 1 PAGE 2 CATEGORY 2 RPT# AIAA
PAPER 74-1197 74/10/00 9 PAGES UNCLASSIFIED
DOCUMENT

UTTL Statistical averages of subsonic inlet distortion
AUTH A/CLARK, L T PAA A/(Boeing Commercial Airplane
Co , Seattle, Wash)
American Institute of Aeronautics and Astronautics and
Society of Automotive Engineers, Propulsion
Conference 10th San Diego Calif Oct 21-23 1974.
AIAA 9 p Research supported by the Boeing
Commercial Airplane Co

MAJS /AERODYNAMIC INTERFERENCE/ENGINE INLETS/FLOW
DISTORTION/INLET FLOW/PRESSURE GRADIENTS/SUBSONIC
AIRCRAFT

MINS / COMBUSTION STABILITY/ COMPRESSOR EFFICIENCY/ ENGINE
TESTS/ PRESSURE RECOVERY/ PRESSURE SENSORS/
STATISTICAL ANALYSIS/ TURBINE ENGINES

ABA (Author)

ABS Results of an experimental investigation of dynamic
distortion in a typical subsonic aircraft inlets are
discussed. The purpose of the investigation was to
study the possibilities for representing the dynamic
distortion in terms of statistical averages. Results
suggested that a sufficient measure of the distortion
can be obtained using cross correlation of the signals
from transducers placed only in the regions of
significant dynamic activity. The method would be
practical for subsonic inlets where the distortion is
localized in predictable places in the inlet. A very
good agreement was found between the gradient of mean
total pressure and unsteady activity. Contour maps of
the mean total pressure gradient can be used to
accurately locate unsteady regions. Cross correlations
between pressure signals within the turbulent region
were used to provide the necessary measure of the size
of the turbulence.

AFFDL-TR-70-20

81N74335 CATEGORY 2 RPT# AFFDL-TR-70-20 CNT#
F33657-69-C-0369 AF PROJ 668A 70/05/00 170 PAGES
UNCLASSIFIED DOCUMENT
UTTL Large scale inlet distortion investigation TLSP
Technical Report, 20 Sep 1968 - 30 Apr 1969
AUTH A/KUTSCHENREUTER P H JR , B/MOORE, M T.,
C/BURNETT, G A
CORP General Electric Co , Cincinnati Ohio. AVAIL NTIS
Wright-Patterson AFB Ohio AFFDL
MAJS /*DYNAMIC CHARACTERISTICS/ ENGINE INLETS/*FLOW
DISTORTION/*INLET PRESSURE/ PERFORMANCE TESTS/
PRESSURE RECOVERY/*STEADY STATE
MINS / ANALYSIS (MATHEMATICS)/ BOUNDARY LAYER CONTROL/
COWLINGS/ ENGINE DESIGN/ FREQUENCY DIVISION
MULTIPLEXING/ SUPERSONIC DIFFUSERS/ SUPERSONIC INLETS/
TURBOFAN ENGINES/ WIND TUNNEL TESTS

AAAF-NT-80-18

81N16991* ISSUE 8 PAGE 993 CATEGORY 2 RPT#: AAAF-NT-80-18
ISBN-2-7170-0614-1 CNT#: DRET-79-069 80/00/00 22 PAGES IN FRENCH
UNCLASSIFIED DOCUMENT DCAF E003393

UTTL: Highly separated transonic flow --- and flow visualization techniques
AUTH: A/FARCY, A.: B/Mercier, V.: C/LEBLANC, R.: D/GOETHALS, R.
CORP: Centre d'Essais Aeronautique Toulouse, Poitiers (France). AVAIL.NTIL
SAP: HC A02/MF A01: CEDOCAR, Paris FF 25 (France and EEC) FF 30
(others) Paris Association Aeronautique et Astronautique de France
Presented at 16th Colloq. d'Aerodyn. Appl., Lille, 13-15 Nov. 1979
MAJS: /*FLOW DISTORTION/*FLOW VISUALIZATION/*SEPARATED FLOW/*SUPERSONIC
INLETS/*TRANSONIC FLOW
MINS: /ANGLE OF ATTACK/ FLOW VELOCITY/ HOT-WIRE ANEMOMETERS/ PRESSURE
SENSORS/ SHADOWGRAPH PHOTOGRAPHY/ SHOCK LAYERS/ TURBULENT FLOW/ WIND
TUNNEL APPARATUS
ABA: Author (ESA)
ABS: A highly separated transonic flow was studied under laboratory
conditions in order to show the usefulness of combining various flow
visualization and analysis techniques in defining jet engine intake
flow characteristics. The experimental setup consisted of a section
of rectangular tubing placed at an angle of incidence of 40 deg to a
transonic jet. Measurements were obtained by filmed visualization,
pressure sensors, laser anemometry and hot wire anemometry. Results
support a general description of the different flow parameters; i.e.,
the existence of shocks and their stability, the existence of a
bubble at the lower leading edge of the air inlet, its dimensions and
particularly the reattachment of its boundary, and flow rate
coefficient approximated from pressure data. An attempt to describe
flow turbulence is also made.

AD-A089817
AEDC-TR-79-77

81N11040# ISSUE 2 PAGE 156 CATEGORY 7 RPT#: AD-A089817 AEDC-TR-79-77
80/09/00 56 PAGES UNCLASSIFIED DOCUMENT

UTTL: An evaluation of statistical methods for the prediction of maximum time-variant inlet total pressure distortion TLSP: Final Report, 1 Oct. 1978 24 Sep. 1979

AUTH: A/SANDERS, M. E.: B/CHRISTENSON, R. E.

CORP: ARO, Inc., Arnold Air Force Station, Tenn. AVAIL. NTIS SAP: HC A04/MF A01 AEDC Sponsored by Air Force

MAJS: /*DISTORTION/*INLET PRESSURE/*PREDICTION ANALYSIS TECHNIQUES/
*STATISTICAL ANALYSIS/*TURBOFAN ENGINES

MINS: / FLOW DISTORTION/ INLET FLOW/ JET ENGINES/ PERFORMANCE PREDICTION/
STATISTICAL TESTS

ABA: GRA

ABS: An analysis was conducted to determine the accuracies and limitations of three statistical methods used to predict engine-face maximum time-variant total pressure distortion. The statistical methods have all been proposed as low-cost alternatives to the time-consuming and costly deterministic method generally used for reducing engine-face time-variant total pressure data. The statistical methods are evaluated by comparing their predicted distortion values and patterns to those measured with the deterministic method. Data comparisons from tests of four different inlet models, covering a wide range of Mach numbers, mass flow ratios, model attitudes, and distortion factors were used during the analysis. The results show good agreement between the measured and predicted values for all three statistical methods. The distortion pattern predictions, however, were inadequate at conditions with high total pressure fluctuation (turbulence). It is recommended that improvements continue to be made in the statistical methods, particularly adjustments for high turbulence conditions, and that the Melick method be used as an on-line distortion analysis tool for inlet performance tests.

Von Karman Inst. of Fluid Dyn.
Unsteady Flow in Turbomachines
Vol. 2, 26p., 1980

80N12333# ISSUE 3 PAGE 324 CATEGORY 34
79/02/02 26 PAGES UNCLASSIFIED DOCUMENT DCAF
FO02628

UTTL Unsteady pressure distortion --- in compressor inflow
AUTH. A/PEACOCK, R E
CORP Cranfield Inst of Tech , Bedfordshire (England)
AVAIL NTIS SAP HC A15/MF A01
In Von Karman Inst for Fluid Dyn Unsteady Flow in
Turbomachines, Vol 2 26 p (SEE N80-12331 03-34)
MAJS /*FLOW DISTORTION/*INLET FLOW/*PRESSURE DISTRIBUTION/*
TURBOCOMPRESSORS/*UNSTEADY FLOW
MINS / MASS FLOW/ MATHEMATICAL MODELS/ PERFORMANCE
PREDICTION/ PULSE GENERATORS/ WAVE REFLECTION
ABA K L
ABS A model is proposed for the solution of unsteady flows
in a compressor embedded in ductwork and subjected to
repetitive or nonrepetitive pulses Various methods
for generating the unsteady pulses are discussed A
three-stage aircraft type compressor was subjected to
pulsating flows in order to determine experimentally
the effects of pulsating flow on compressor
performance

Ph. D. Thesis,
North Carolina State University

79N33203 ISSUE 24 PAGE 3178 CATEGORY 7
79/00/00 160 PAGES UNCLASSIFIED DOCUMENT
UTTL An experimental study of the response of a
turbomachine rotor to a low frequency inlet distortion
TLSP Ph D Thesis
AUTH. A/HARDIN L W
CORP North Carolina State Univ at Raleigh SAP Avail
Univ Microfilms Order No 7923049
MAJS /*FLOW DISTORTION/*INLET FLOW/*ROTOR AERODYNAMICS/*
TURBOMACHINERY
MINS / AERODYNAMIC STALLING/ FLOW VELOCITY/ PRESSURE
DISTRIBUTION/ THREE DIMENSIONAL FLOW
ABA. Dissert Abstr
ABS An experiment was conducted to measure the response of
an isolated turbomachine rotor to a distortion in
inlet axial velocity. A once-per-revolution sinusoidal
variation in axial velocity with an amplitude of
approximately twenty percent of the average axial
velocity was generated by an upstream screen. The
response of the rotor was studied using pressure
transducers and skin friction gages mounted on one of
the rotor blades and a velocity probe at the rotor
exit plane as well as with standard stationary frame
pneumatic instrumentation

MBB-UFE-1359-0

79N23936# ISSUE 15 PAGE 1929 CATEGORY 2 RPT #: MBB-UFE-1359-0
77/09/26 80 PAGES In GERMAN: ENGLISH summary UNCLASSIFIED DOCUMENT
DCAF E002631

UTTL: Aerodynamic problems in engine airframe integration on fighter
airplanes
AUTH: A/LOTTER, K. W.
CORP: Messerschmitt-Boelkow-Blohm G.m.b.H., Ottobrunn (West Germany). CSS:
(Unternehmensbereich Flugzeuge Entwicklung.) AVAIL.NTIS SAP: HC
A05/MF A01 Presented at 85th Wehrtech. Symp. Luftfahrttechn. III
Moderne Triebwerkstechnol., Mannheim. West Ger., 21-22 Sep. 1977
MAJS: /*AERODYNAMIC INTERFERENCE/*ENGINE INLETS/*FIGHTER AIRCRAFT
MINS: / AFTERBODIES/ AIRFRAMES/ FLOW DISTORTION/ INLET NOZZLES/ NOZZLE
DESIGN/ THRUST REVERSAL
ABA: Author (ESA)
ABS: Different types of intake are discussed together with engine mass
flow/air intake matching problems. Emphasis is given to
intake/engine compatibility and instantaneous distortion
measurements. The sources and consequences of intake malfunctions
are illustrated; some typical supersonic fighter aircraft intakes are
described. Nozzle concepts are presented and interference effects
between propulsive jet and external flow are shown. The afterbody
drag of fighter aircraft is given. Improvements with
convergent/divergent nozzles are shown. Thrust reversal interference
effects are mentioned. Future trends are presented.

AD-A064776
 NCSU/EDC-78-6
 AFOSR-79-0073TR

79N23089# ISSUE 14 PAGE 1806 CATEGORY 7 RPT#: AD-A064776
 NCSU/EDC-78-6 AFOSR-79-0073TR CNT#: F44620-76-C-0055 AF AFOSR-280275
 78/05/00 31 PAGES UNCLASSIFIED DOCUMENT

UTTL: An experimental study of the response of a turbo-machine rotor to a low frequency inlet distortion TLSP: Interim Report. Jan. 1976 - Dec. 1978

AUTH: A/HARDI, L.

CORP: North Carolina State Univ. at Raleigh.: United Technologies Research Center, East Hartford, Conn. CSS: (Engineering Design Center.)
 AVAIL.NTIS SAP: HC A08/MF A01
 Prepared in Cooperation with United Technol. Res. Center

MAJS: /*FLOW DISTORTION/*GAS FLOW/*INLET FLOW/*ROTORS

MINS: / AERODYNAMIC CHARACTERISTICS/ AERODYNAMIC STALLING/ AIRFOILS/
 DYNAMIC RESPONSE/ LEADING EDGES/ LOW FREQUENCIES/ PRESSURE/ TRAILING EDGES/ TURBOCOMPRESSORS/ TURBOMACHINE BLADES

ABA: GRA

ABS: As part of a joint technical effort involving North Carolina State University and United Technologies Research Center, an experiment was conducted to measure the response of an isolated turbomachine rotor to a distortion in inlet axial velocity. A once-per-revolution sinusoidal variation in axial velocity with an amplitude of approximately twenty percent of the average axial velocity was generated by an upstream screen. The response of the rotor was studied using pressure transducers and skin friction gages mounted on one of the rotor blades and a velocity probe at the rotor exit plane as well as with standard stationary frame pneumatic instrumentation. The rotor was operated in undistorted flow to establish the quasi-steady behavior of the compression system. When the air inlet angle was reduced past a certain limit, the rotor began to experience rotating stall. When the rotor was operated in distorted flow, the pressures on the surface of the instrumented blade were observed to vary as a function of the instantaneous inlet angle. These variations were greatest at the leading edge of the airfoil and became smaller toward the trailing edge. This concentration of activity in the leading edge region is more pronounced than has been observed for isolated airfoils. As the instrumented blade traversed the distortion, it was observed to operate transiently at inlet angles below the quasi-steady stall point in an apparently unstalled condition.

AD-A059437
MDC-00651
AFOSR-78-1277TR

79N14043# ISSUE 5 PAGE 551 CATEGORY 2 RPT #: AD-A059437 MDC-Q0651
AFOSR-78-1277TR CNT#: F49620-77-C-0082 78/05/00 31 PAGES UNCLASSIFIED
DOCUMENT

UTTL: Unsteady transonic diffuser flows in a two-dimensional diffuser TLSP:
Annual Report. 1 Apr. 1977 - 31 Mar. 1978

AUTH: A/SAJBEN, M.: B/KROUTIL, J. C.

COPR: McDonnell-Douglas Research Labs., St. Louis, MO. AVAIL. NTIS SAP: HC
A03/MF A01

MAJS: /*AIR BREATHING ENGINES/*NONUNIFORM FLOW/*TRANSONIC FLOW/*VANELESS
DIFFUSERS

MINS: / BOUNDARY LAYER FLOW/ DYNAMIC TESTS/ FLOW DISTORTION/ INLET FLOW/
MISSILES/ SHOCK WAVES

ABA: GRA

ABS: The present study is part of a continuing investigation of unsteady,
transonic diffuser flows, with application to dynamic distortion in
the inlets of fighter aircraft and airbreathing missile propulsion
systems. The investigation is focused on a simple, two-dimensional
diffuser configuration displaying a weak shock shortly downstream of
the throat. Flows in this model were found to display self-excited
oscillations involving the shock and the entire subsonic flow behind
it. The shock displacement amplitudes are comparable to the throat
height and occur at frequencies characterized by the flow speed and
the length of the divergent diffuser section. The time-mean and
fluctuating properties of this flow and the dependence of these
properties on shock strength have been documented.

NASA-CR-2766

77N12041*# ISSUE 3 PAGE 287 CATEGORY 5 RPT#: NASA-CR-2766 R76AEG272
CNT#: NAS3-19522 76/12/00 103 PAGES UNCLASSIFIED DOCUMENT

UTTL: Analysis of pressure distortion testing TLSP: Final Report
AUTH: A/KOCH. K. E.: B/REES. R.L.
CORP: General Electric Co., Cincinnati, Ohio. AVAIL.NTIS SAP: HC A06/MF
AO1 Washington NASA

MAJS: /*ENGINE INLETS/*FLOW DISTORTION/*MATHEMATICAL MODELS/*PRESSURE
DISTRIBUTION/*PRESSURE DROP

MINS: / AIR FLOW/ COMPRESSORS/ ENGINE DESIGN/ STEADY STATE

ABA: Author

ABS: The development of a distortion methodology, method D was documented,
and its application to steady state and unsteady data was
demonstrated. Three methodologies based upon DIDENT, a NASA-LeRC
distortion methodology based upon the parallel compressor model were
investigated by applying them to a set of steady state data. The
best formulation was then applied to an independent data set. The
good correlation achieved with this data set showed that method E,
one of the above methodologies, is a viable concept. Unsteady data
were analyzed by using the method E methodology. This analysis
pointed out that the method E sensitivities are functions of pressure
defect level as well as corrected speed and pattern.

AGARD Unsteady Phenomena
in Turbomachinery

76N25190 ISSUE 16 PAGE 2018 CATEGORY 2 76/04/00 30 PAGES UNCLASSIFIED
DOCUMENT

UTTL: The effect of turbulent mixing on the decay of sinusoidal inlet
distortions in axial flow compressors

AUTH: A/MOKELKO, H.

CORP: Motoren- und Turbinen-Union Muenchen G.m.b.H. (West Germany). in
AGARD Unsteady Phenomena in Turbomachinery 30 p (SEE N76-25169 16-07)

MAJS: /*FLOW DISTORTION/*PERTURBATION THEORY/*TURBOCOMPRESSORS/*TURBULENT
MIXING

MINS: / INLET FLOW/PREDICTION ANALYSIS TECHNIQUES/ PRESSURE OSCILLATIONS/
UNSTEADY FLOW

ABA: Author

ABS: A small perturbation actuator disc theory is presented for the
prediction of the decay of sinusoidal flow distortions in high hub
tip ratio axial compressors with steady circumferential inlet
maldistribution. The theory accounts for the turbulent mixing of the
flow upstream and within the compressor. Decay rates and
circumferential phase shifts of first, second, fourth and eighth
order cosine wave pressure and velocity perturbations are calculated
for equal amplitudes and phases of the four total pressure
disturbances upstream of the compressor. The results are compared
with interstage traverse data obtained from a 4-stage axial flow
compressor. A comparison between corresponding analytical results
obtained from the same theory neglecting viscosity and the
experimental data is also performed. It is found that turbulent
mixing has little influence on the development of the first order
disturbance but that the influence grows rapidly as the order of the
disturbance increases.

AGARD Unsteady Phenomena in
Turbomachinery76N25187 ISSUE 16 PAGE 2018 CATEGORY 2 76/04/00 16 PAGES UNCLASSIFIED
DOCUMENT

UTTL: The relationship between steady and unsteady spacial distortion - - -
in turbocompressor intake flow

AUTH: A/FREEMAN. C.

CORP: Rolls-Royce Ltd., Derby (England). CSS: (Engine Div.) In AGARD
Unsteady Phenomena in Turbomachinery 16 p (SEE N76-25169 16-07)

MAJS: /*FLOW DISTORTION/*INLET PRESSURE/*STEADY FLOW/*TURBOCOMPRESSORS

MINS: / FLUCTUATION THEORY/ MATHEMATICAL MODELS/ PRESSURE REDUCTION/
UNSTEADY FLOW

ABA: Author

ABS: Simple theories of turbulence are used to develop a model that
relates the fluctuating spacial distortion to the time average
spacial distortion. This model uses the relationship between the
fluctuating total pressure, the fluctuating velocities, the Reynolds
stress and the mean velocity gradient. These fluctuating total
pressures are then used with a correlation coefficient to determine
the amplitude of the fluctuating average total pressure over part of
the compressor face. Comparisons are made between the method and
experiment to show that the method describes many features of the
flow.

Pennsylvania State University

76N25186 ISSUE 16 PAGE 2018 CATEGORY 7 CNT#: NSG-3031
N00014-67-A-0226-0005 NR PROJ. 098-038 PROJ. SQUID 76/04/00 13 PAGES
UNCLASSIFIED DOCUMENT

UTTL: Axial flow rotor unsteady response to circumferential inflow
distortions
AUTH: A/BRUCE, E.P.
CORP: Pennsylvania State Univ., University Park, CSS: (Dept. of Mechanical
Engineering). IN AGARD Unsteady Phenomena in Turbomachinery 13 p (SEE
N76-25169 16-07)
MAJS: /*INLET FLOW/*PRESSURE REDUCTION/*TURBOCOMPRESSORS/*UNSTEADY FLOW
MINS: / AXIAL COMPRESSION LOADS/ CASCADE FLOW/ FLOW DISTORTION/ INLET
PRESSURE
ABA: Author
ABS: The unsteady response of an axial flow fan rotor to steady,
circumferential inflow velocity and stagnation pressure distortions
is assessed by two different methods. These are: (1) investigation
of the unsteady normal force and pitching moment on a chordwise
element of a rotor blade; and (2) investigation of the variation of
the stagnation pressure distortion between the inlet and exit of the
rotor. Experimental measurements of these unsteady characteristics
are presented as a function of the geometry of the rotor -- stagger
angle solidity and steady angle of incidence -- for sinusoidally
varying circumferential distortions with different numbers of
distortion cycles. These measurements are compared with several
theoretical analyses. While these comparisons indicate some of the
deficiencies which exist in the theories, the existence of an
unsteady cascade effect and the ability of the theories to adequately
predict the trend of the unsteady response due to variations in
reduced frequency, rotor stagger angle, solidity and mean incidence
angle is clearly demonstrated.

AGARD Unsteady Phenomena in
Turbomachinery

76N25177* ISSUE 16 PAGE 2017 CATEGORY 2 CNT#: NAS1-12505 76/04/00 18
PAGES UNCLASSIFIED DOCUMENT

UTTL: Application of rotor mounted pressure transducers to analysis of inlet turbulence --- flow distortion in turbofan engine inlet

AUTH: A/HANSON, D. B.

CORP: Hamilton Standard, Windsor Locks, Conn. In AGARD Unsteady Phenomena in Turbomachinery 18 p (SEE N76-25169 16-07)

MAJS: /*COMPRESSOR ROTORS/*FLOW DISTORTION/*INLET FLOW/*PRESSURE SENSORS

MINS: / PRESSURE MEASUREMENTS/ PRESSURE OSCILLATIONS/ ROTOR BLADES (TURBOMACHINERY)/ TURBOFAN ENGINES/ TURBULENT BOUNDARY LAYER

ABA: Author

ABS: Miniature pressure transducers installed near the leading edge of a fan blade were used to diagnose the non-uniform flow entering a subsonic tip speed turbofan on a static test stand. The pressure response of the blade to the inlet flow variations was potted in a form which shows the space-time history of disturbances ingested by the rotor. Also periodically sampled data values were auto- and cross-correlated as if they had been acquired from fixed hot wire anemometers at 150 equally spaced angles around the inlet. With a clean inlet and low wind, evidence of long, narrow turbulence eddies was easily found both in the boundary layer of the fan duct and outside the boundary layer. The role of the boundary layer was to follow and amplify disturbances in the outer flow. These eddies frequently moved around the inlet with a corkscrew motion as they passed through.

AGARD-CP-177

76N25169# ISSUE 16 PAGE 2016 CATEGORY 7 RPT#: AGARD-CP-177 AD-A028614
76/04/00 565 PAGES UNCLASSIFIED DOCUMENT

UTTL: Unsteady phenomena in turbomachinery
CORP: Advisory Group for Aerospace Research and Development. Paris
(France). AVAIL.NTIS SAP: HC \$13.50 Presented at the 46th Meeting
of the AGARD Propulsion and Energetics Panel. Monterey, Calif.,
22-26 Sep. 1975
MAJS: /*CONFERENCES/*FLOW DISTORTION/*TURBOCOMPRESSORS/*UNSTEADY FLOW
MINS: / CASCADE FLOW/ ENGINE DESIGN/ INLET FLOW/ ROTATING STALLS/ ROTOR
BLADES (TURBOMACHINERY)/ TURBOMACHINERY/ TURBULENT WAKES
ANN: Turbomachinery unsteady aerodynamics are reviewed with emphasis on
flow distortion phenomena inside subsonic, transonic and supersonic
axial flow compressor stages.

NASA TM X-3246

75N26968# ISSUE 18 PAGE 2199 CATEGORY 2 RPT#: NASA-TM-X-3246 E-8213
75/07/00 78 PAGES UNCLASSIFIED DOCUMENT

UTTL: Effect of inlet ingestion of a wing tip vortex on compressor face
flow and turbojet stall margin

AUTH: A/MITCHELL, G. A.

CORP: National Aeronautics and Space Administration. Lewis Research Center.
Cleveland, Ohio AVAIL.NTIS SAP: HC \$4.75
Washington

MAJF: /*INGESTION (ENGINES)/*INTAKE SYSTEMS/*J-85 ENGINE/*VORTICES/*WING
TIPS

MINS: / ENGINE FAILURE/ FLOW DISTORTION/ INLET PRESSURE/ SUPERSONIC
COMPRESSORS/ SUPERSONIC INLETS/ TWO DIMENSIONAL FLOW

ABA: Author

ABS: A two-dimensional inlet was alternately mated to a coldpipe plug
assembly and a J85-GE-13 turbojet engine and placed in a Mach 0.4
stream so as to ingest the tip vortex of a forward mounted wing.
Vortex properties were measured just forward of the inlet and at the
compressor face. Results show that ingestion of a wing tip vortex by
a turbojet engine can cause a large reduction in engine stall
margin. The loss in stall compressor pressure ratio was primarily
dependent on vortex location and rotational direction and not on
total-pressure distortion.

NASA TM X-3169

EXPERIMENTAL INVESTIGATION OF A SIMPLE DISTORTION INDEX UTILIZING
STEADY-STATE AND DYNAMIC DISTORTIONS IN A MACH 2.5
MIXED-COMPRESSION INLET AND TURBOFAN ENGINE

by William G. Costakis

Lewis Research Center

SUMMARY

A wind tunnel investigation was conducted to determine the amplitude and spatial distribution of steady-state and dynamic distortion produced in an inlet with 45 percent of the overall supersonic area contraction occurring internally. The inlet support strut location and/or the overboard bypass flow rate had a significant effect on the spatial distribution of distortion. Because of this effect the majority of the stall points exhibited four-per-revolution patterns of distortion.

Data from this test were used to formulate a simple index that combines steady-state and dynamic distortions. Distortion results obtained with this index correlated well with exhaust nozzle area. The exhaust nozzle area of a TF30-P-3, as modified for use in this test, can be controlled in a scheme to avoid engine stall. A considerable increase in engine distortion tolerance can be achieved by opening the 7th-stage bleed. The engine exhibited higher tolerance to distortion for multiple patterns of distortion per revolution than for a one-per-revolution pattern of distortion.

REFERENCES

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2. Ellis, S. H., and Brownstein, B. J. A Procedure for Estimating Maximum Time-Variant Distortion Levels with Limited Instrumentation. Paper 72-1099, AIAA, Nov.-Dec. 1972.
3. Costakis, William G. Analog Computer Implementation of Four Instantaneous Distortion Indices. NASA TM X-2993, 1974.

NASA TM X-3169

4. Burstadt, Paul L., and Calogeras, James E. Instantaneous Distortion in a Mach 2.5, 40-Percent-Internal-Contraction Inlet and Its Effect on Turbojet Stall Margin. NASA TM X-3002, 1974.
5. Burcham, Frank W., Jr., and Hughes, Donald L. Analysis of In-Flight Pressure Fluctuations Leading to Engine Compressor Surge in an F-111A Airplane for Mach Numbers to 2.17. Paper 70-624, AIAA, June 1970.
6. Wasserbauer, Joseph F., Shaw, Robert J., and Neumann, Harvey E. Design of a Very-Low-Bleed Mach 2.5 Mixed-Compression Inlet with 45 Percent Internal Contraction. NASA TM X-3135, 1974.
7. Batterton, Peter G., Arpasi, Dale T., and Baumbick, Robert J. Digital Integrated Control of a Mach 2.5 Mixed-Compression Supersonic Inlet and Augmented Mixed-Flow Turbofan Engine. NASA TM X-3075, 1974.
8. Webb, John A., Jr., Mehmed, Oral, and Hiller, Kirby W. Improved Design of High-Response Slotted-Plate Overboard Bypass Valve for Supersonic Inlets. NASA TM X-2812, 1973.

AN ANALYSIS OF THE INFLUENCE OF UNSTEADY
CASCADED AIRFOIL BEHAVIOR ON AXIAL
FLOW COMPRESSORS WITH UNSTEADY
AND DISTORTED INFLOW*

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Abstract

The influence of unsteady and circumferentially distorted flow on the stall of axial flow compressors has been studied in earlier works assuming that the stage characteristic curves of the compressor behave in a quasi-steady-state manner. This assumption becomes invalid at high flow oscillation frequencies because the forces exerted on the fluid by the cascaded airfoils do not instantaneously adjust to changes in the incoming flow. The departure from the quasi-steady-state behavior of cascaded airfoils has been analyzed using potential flow theory by Goethert, Reddy and Schorr. Their results, reviewed in this paper, produced a correction for unsteady flow effects to the cascade lift coefficients. Their analysis illustrated the difference in behavior between an oscillating airfoil in a steady stream and a stationary airfoil in an oscillating stream. Also, a difference between the magnitude of the unsteady effects on a cascade and on an isolated airfoil was demonstrated. To quantitatively evaluate the impact of the unsteady cascade effects on overall compressor behavior, the Goethert-Reddy results were incorporated into a digital computer compressor math model. The math model results indicated the quasi-steady-state assumption to be valid up to approximately 300 HZ. The unsteady cascade results were also used as a correction to parallel compressor theory to estimate the influence of circumferential distortion on compressor stall. The estimations were compared to experimental results and remarkably good agreement was indicated.

Summary

Early analyses of compressor stall caused by unsteady and circumferentially distorted flow were performed using mathematical models of dynamic compressor behavior which were based on the quasi-steady-state stage characteristic assumption. That is, it was assumed that the steady-state stage pressure ratio - flow function relationship was maintained during dynamic excursions of compressor operation. Goethert, Reddy and Schorr analyzed the unsteady potential flow of an inviscid, incompressible fluid through a cascade of staggered flat plate airfoils. The results of their analysis provided insight into the range of validity of the quasi-steady-state stage characteristics assumption. Their principal unsteady cascade results were reviewed and are as follows:

Reference 56
C.94

1. With planar oscillatory compressor inflow, or with circumferentially distorted inflow, the unsteady cascade flow is represented, relative to the moving rotor blade, as a fluctuating flow around a stationary blade. Further, it was shown that the unsteady lift response of a stationary blade in an fluctuating flow differs strongly from that of an oscillating blade in a steady stream, especially at high oscillatory frequencies.
2. The mutual interference between cascaded blades in unsteady flow is important, and, the unsteady lift behavior of a cascade departs from quasi-steady-state behavior much more slowly than does that of an isolated airfoil
3. Quantitative values of the unsteady lift coefficient compared to the quasi-steady lift coefficient were calculated and, the value of unsteady lift coefficient at very low reduced frequency was shown to tend toward the steady-state value which is physically expected. The unsteady contribution to lift coefficient at very high reduced frequency tended toward zero which also is physically anticipated.

The quantitative results of the Goethert-Reddy unsteady cascade analysis were approximated for incorporation into a digital computer math model of overall compressor dynamic response. The math model was then used to evaluate the range of validity of the quasi-steady-state stage characteristic assumption by imposing an oscillation in compressor inlet pressure on the model and observing the response. The results indicated that the quasi-steady-state assumption begins to cause an overprediction of the ability of the fluctuation to cause stall above approximately 300 Hz for the specific compressor and operating condition investigated.

The unsteady cascade results were also applied as a correction to the parallel compressor theory used to estimate the influence of circumferential distortion on compressor stall. Estimates in loss of stall pressure ratio due to distortion were compared to experimental data for a J-85-13 engine and very good agreement was demonstrated.

In general, the review of the unsteady cascade behavior and the calculated examples of overall compressor behavior indicated that any non-uniform or unsteady flow phenomenon occurring in the compressor brings the unsteady cascade behavior into play. However, the results are negligible for phenomenon producing low reduced frequencies (or extremely high reduced frequencies). Many phenomena do occur in a range requiring consideration of unsteady cascade effects. These include, planar oscillations in excess of approximately 300 Hz, (for multi-stage compressors) circumferential distortion with low pressure regions less than 90 degs. and rotating stall, blade flutter and other "post stall" excursions of compressors not analyzed in this paper.

NASA TM X-68189

INSTANTANEOUS DISTORTION INVESTIGATION

by James E. Calogeras

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio

INTRODUCTORY REMARKS

The purpose of this presentation is to review some of the results obtained in an inlet-engine compatibility test run in the 10x10 SWT of the NASA-Lewis Research Center. This program was run to measure the time-variant distortions produced in a supersonic inlet and to relate a unique distortion peak, occurring in an instant of time, to the origin of stall in a compressor. The major stumbling block in this type of effort is the determination of a proper increment of time over which to average pressures before computing distortions. It is reasonable to expect that the proper averaging time is related to the particular compressor in question. Indeed, some results of similar investigations have indicated that the proper averaging time was solely dependent on the response characteristics of the compressor. Nonetheless, the most significant point in this presentation is that the proper averaging time may not be solely dependent on a particular compressor, and, in fact, may vary with operating conditions, even for the same inlet-engine combination.

CONCLUDING REMARKS

Of 29 compressor stall points recorded in an investigation of supersonic inlet-engine compatibility, seven were extensively analyzed using the instantaneous distortion approach. Results indicate that this approach can be used to identify the unique distortion peak which is related to compressor stall. But the pressure averaging time required in this type analysis was found to vary considerably over the range of data analyzed. It may ultimately be necessary to use a mean averaging time in the evaluation of propulsion system compatibility, particularly for a system covering a wide range of operating conditions.

AIAA Paper 72-1099

**A PROCEDURE FOR ESTIMATING MAXIMUM TIME-VARIANT
DISTORTION LEVELS WITH LIMITED INSTRUMENTATION**

by

S. H. ELLIS

Pratt and Whitney Aircraft
West Palm Beach, Florida
and

B. J. BROWNSTEIN

Air Force Aero Propulsion Laboratory
Wright Patterson AFB, Ohio

Abstract

Inlet data measured with complete high-response instrumentation have been used to establish a new procedure for assessing propulsion system flow stability from tests with limited instrumentation. *** This procedure is not accurate enough for stability verification testing, but is intended for use during the early stages of propulsion system selection. The most-severe time-variant inlet distortion levels are predicted within an uncertainty band that is a function of the number of high-response probes and the turbulence level. The appropriate distortion index is used to translate the effects of circumferential and radial distortions into predicted stall margin loss. Substantiating data and a representative stability assessment are presented.

VII Summary and Conclusions

A procedure has been established and substantiated for estimating propulsion system flow stability from inlet tests with limited high-response instrumentation. It is applicable to all engines and all inlet data analyzed thus far.

The accuracy of estimating stall margin loss is insufficient for stability verification, but is adequate for early inlet studies.

This procedure is recommended for assessing flow stability during propulsion system selection, and for automatic data validation during analysis of time-variant inlet data.

NASA TM X-2081

EFFECT OF DYNAMIC VARIATIONS IN ENGINE-INLET PRESSURE ON THE
COMPRESSOR SYSTEM OF A TWIN-SPOOL TURBOFAN ENGINE

by John E. McAulay

Lewis Research Center

SUMMARY

An investigation was conducted to determine the effects of dynamic inlet pressure variations on the compressor system of a turbofan engine. These inlet pressure variations were produced by rapid changes in secondary airflow injected as many small contrastream jets ahead of the engine. The air jet system was mechanically capable of frequencies up to 200 hertz; however, due to the jet system and engine-inlet duct flow characteristics, measurable amplitudes in engine-inlet and fan-compressor system pressures were not achieved above 80 hertz.

The air jet system was used to produce cyclic variations in engine-inlet pressure with essentially zero instantaneous spatial distortion (uniform flow), 180° circumferential-extent distortion, and rotating distortion. In addition, single pressure pulses were introduced with uniform flow and 180° distortion.

Uniform cyclic inlet pressure variations resulted in transient changes in the fan-compressor stage group pressure ratios and in fan-compressor stall at high amplitudes (i. e., normalized inlet pressure amplitudes of 0.28 or greater). The changes in pressure ratio during the cyclic inlet pressure variation were largest for the fan tip, fan hub, and the high-compressor rear stage groups. Oscillating spatial distortions at the engine face resulted in compressor stall at substantially lower inlet pressure amplitudes than those required to produce stall with spatially uniform oscillating pressure variations. Stall was produced at lower values of circumferential distortions with the 180° pulsed distortions than with the 180° steady-state distortions. A comparison of the instantaneous distortion required to produce compressor stall for the oscillating 180° distortion, rotating distortion, 180° single pulse, and 180° steady-state distortion showed that stall tolerance is a function of the instantaneous distortion, the rate of change of the inlet pressure, and the dwell-time of the fan-compressor rotor blading in the low-pressure region of an engine-inlet distortion.

NASA TM X-2081

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NASA TM X-1946

TECHNIQUE FOR INDUCING CONTROLLED STEADY-STATE AND DYNAMIC
INLET PRESSURE DISTURBANCES FOR JET ENGINE TESTS

by Carl L. Meyer, John E. McAulay, and Thomas J. Biesiadny

Lewis Research Center

SUMMARY

An investigation was conducted to evaluate a technique wherein secondary air was injected through an array of small nozzles uniformly distributed in an engine inlet duct to achieve momentum interchange with the primary air forward of the compressor face location. High-response servo-operated valves provided steady-state and dynamic control of secondary airflow.

Through control of secondary-air distribution and flow rate, the technique provides a way of inducing variable-amplitude steady-state or dynamic pressure distortions or dynamic uniform pressure oscillations without excessive random pressure amplitude. Dynamic pressure distortions were not evaluated in this investigation. The amplitude of induced dynamic pressure oscillations attenuated appreciably at frequencies above 20 hertz. Work is currently in progress with the purpose of improving amplitude capabilities at the higher frequencies.

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NASA TM X-1928

EXPERIMENTAL INVESTIGATION OF THE EFFECTS OF PULSE PRESSURE
DISTORTIONS IMPOSED ON THE INLET OF A TURBOFAN ENGINE

by Leon M. Wenzel
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SUMMARY

A YTF-30-P-1 turbofan was operated in an altitude chamber. A distortion device capable of effecting pulse depressions in inlet pressure was installed ahead of the engine. The pulses could be varied in duration and amplitude. The portion of the inlet duct subjected to distortion could be varied circumferentially from 60° to 360° in 60° sectors.

Engine stall sensitivity as a function of pulse duration and distortion sector angle was mapped. The amplitude of pulse necessary to stall the engine was found to be an inverse function of pulse duration. The engine was most sensitive to distorted sectors of 180° and 240° .

Transient recordings of engine pressures and pressure ratios during stall are presented. Also, data are presented which indicate the circumferential propagation rate of a pressure pulse traveling through the engine to be 20° per stage.

SUMMARY OF RESULTS

A YTF-30-P-1 turbofan engine was operated in an altitude chamber. Air jets, driven by high-response servovalves, were installed in the engine inlet duct. The jets were operated to effect single pulse depressions in engine inlet pressure. The distorted portion of the inlet was varied circumferentially by sextants from 60° to 360° . Pulse duration was varied from 6 to 100 milliseconds.

The engine stall sensitivity as a function of pulse duration and distortion sector angle was mapped. The pulse amplitude required to stall the engine was found to be an inverse function of pulse duration. The engine was most sensitive to distortion sectors of 180° and 240° .

Transient recordings of engine pressures and pressure ratios during stall are presented. Data are also presented which indicate the circumferential propagation of an inlet pressure pulse as it travels axially through the engine. The propagation rate was measured to be 20° per stage.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, August 27, 1969,
720-03.

Reference 61
C.101

NASA TM X-2993

ANALOG COMPUTER IMPLEMENTATION OF FOUR
INSTANTANEOUS DISTORTION INDICES

by William G. Costakis
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SUMMARY

A program was conducted to test the compatibility of a J85-GE-13 engine and an axisymmetric mixed-compression inlet. The original program was designed to measure the time-variant distortion produced in a supersonic inlet and its relation to stall. Dynamic distortion data obtained from these tests were used to establish the feasibility of using on-line generated indices as control signals. An analog computer was used in the analysis for the study of long periods of time prior to stall. Dynamic distortion data obtained from these tests were analyzed over time increments of 1.3 seconds. The analog program of four instantaneous distortion indices and their results are presented. A modified approach in determining the extent of distortion is also presented.

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SUPERSONIC WIND TUNNEL INVESTIGATION OF INLET-ENGINE COMPATIBILITY

by Robert E. Coltrin and James E. Calogeras

Lewis Research Center
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Cleveland, OhioAbstract

Results are presented from an experimental investigation in the Lewis 10- by 10-Foot Supersonic Wind Tunnel of inlet-produced dynamic distortion and its effects on the stall margin of a J-85 turbojet engine. Compressor face total pressure fluctuations were surveyed by a rotating rake, and resultant recordings analyzed as to both amplitude and frequency content. The magnitude of dynamic distortion was generally found to vary with terminal shock strength, but was favorably influenced by vortex generator placement in the subsonic diffuser. Results of spectral analysis indicate a flat power spectrum to 300 Hz which then decreased by 75 percent at 1000 Hz. In most cases, a bypass cavity resonance was superimposed on this spectrum at about 270 Hz. Cross-spectra information indicated that the low frequency coherence between two probes was high for close probe spacing but fell off rapidly with increased distance. RMS pressure fluctuations of 5 percent of the steady-state total pressure were found to cause complete stall margin degradation at certain engine speeds.

I Introduction

It has long been recognized that the flow from a supersonic diffuser may have a serious total pressure distortion at the compressor face which can influence the engine stall margin. More recently it has become apparent that total pressure fluctuations are superimposed upon the steady distortion which may further alter the compressor stall margin. A major share of this dynamic distortion appears to originate from the unsteady interaction of the diffuser terminal shock with the inlet boundary layer which produces localized three-dimensional transients in total pressure at the diffuser exit. The effects of dynamic distortion on compressor stall margin were first observed during wind tunnel tests of the XB-70 propulsion system. A subsequent investigation was made which artificially created distortion dynamics in an altitude engine test facility utilizing a choke point and subsonic diffuser ahead of a turbojet. Shock waves downstream of this choke point interacted dynamically with the duct boundary layer, and the effects of resultant dynamic distortion were determined on the stall margin of a GE-J93 turbojet engine.

Only limited data are presently available which define the dynamic distortion that actually exists within an inlet. Therefore, the present study was undertaken in the Lewis 10- by 10-Foot Supersonic Wind Tunnel to define the character of dynamic distortion existing in a typical supersonic inlet. A Mach 2.50 inlet was used with a nacelle in which either a cold pipe-choked plug assembly or a J85-GE-13 turbojet engine could be installed. Both steady state and dynamic distortion measurements were made using first the cold-pipe assembly and then the engine. Engine effects on these parameters were then determined by comparing them at

similar inlet-operating conditions. The effects of inlet configuration variables, such as throat bleed and vortex generators, on dynamic distortion were investigated for critical and supercritical inlet operation. The character of this dynamic distortion was determined by analyzing both its amplitude and its frequency content. The engine was stalled at representative inlet conditions to determine the effect of dynamic distortion on compressor stall margin.

VI. Concluding Remarks

A wind tunnel investigation was made to determine

- (a) The magnitude and frequency spectra of the distortion produced in a Mach 2.50 design inlet with 60 percent internal contraction; and
- (b) The effect of this dynamic distortion on the compressor stall margin of a J-85 turbojet engine.

The following results were obtained

1. The magnitude of the dynamic distortion was found to vary with terminal shock position
2. Increased throat bleed reduced the dynamic distortion for near critical inlet operation.
3. Installation of vortex generators in the subsonic diffuser reduced dynamic distortion over the full inlet-operating range.
4. Power spectral information shows that the random dynamic level was flat to about 300 Hz and then decreased by about 75 percent at 1 kHz. For most bleed configurations, a bypass cavity resonance was superimposed on the general spectrum at about 270 Hz.
5. Installation of a dual vane cascade at the entrance to each bypass cavity removed the 270 Hz resonance.
6. The low-frequency coherence between two probes was high for close probe spacing but fell off rapidly with increased distance.
7. The presence of the engine did not noticeably effect the dynamic distortion magnitudes or spectrums
8. RMS pressure fluctuations of 5 percent of the steady-state total pressure were found to cause complete stall margin degradation at certain engine speeds

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EXPERIMENTAL INVESTIGATION OF DYNAMIC DISTORTION IN A MACH
2.50 INLET WITH 60 PERCENT INTERNAL CONTRACTION AND ITS
EFFECT ON TURBOJET STALL MARGIN

by James E. Calogeras
Lewis Research Center

SUMMARY

A wind tunnel investigation was made to determine the amplitude and spatial distribution of dynamic distortion produced in an inlet with 60 percent of the overall supersonic area contraction occurring internally, and to determine the effect of this dynamic distortion on compressor stall margin. The effects of subsonic diffuser vortex generators, inlet throat bleed, and engine presence on dynamic distortion were also investigated. Results showed that compressor stall margin was not seriously reduced by an average rms amplitude of the fluctuating component of total pressure equal to 2 percent of the steady-state total pressure. An increase in this dynamic distortion rms amplitude to 5 percent of the steady-state total pressure could, however, cause complete degradation of the stall margin at certain engine speeds. The use of vortex generators on the cowl and centerbody surfaces was found to be very effective in reducing dynamic distortion over the entire inlet-operating range. A 300-hertz resonance existed at most of the inlet-operating conditions. This resonance originated in the cavities associated with the inlet bypass ports which were placed just upstream of the compressor face station. This resonance had a strong effect on the spatial distribution of high local dynamic distortion lobes. The presence of the engine had no significant effect on either the average magnitude of steady-state and dynamic distortions or their respective spatial distributions.

NASA TM X-1842

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AIAA Paper 68-649

APPLICATION OF STATISTICAL PARAMETERS IN
DEFINING INLET AIRFLOW DYNAMICS

by

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Abstract

During wind tunnel and flight testing of several inlet duct designs, duct pressure time histories were recorded on magnetic tape. Preliminary examination of the data indicated the pressure variations to be random processes having Gaussian distribution patterns. Reduction of this data by electronic analog instruments was made to obtain specific statistical parameters; namely, variance, time-auto-correlations and cross-correlations, and power spectral densities. This paper discusses the utilization of these statistical parameters in helping to define the dynamic properties of the inlet duct and the state of the airflow turbulence. Examples are given illustrating the methods and techniques involved. A short discussion of the advantages and disadvantages of electronic analog data reduction is also given along with some of the major problems and pitfalls encountered during this investigation.

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16 Abstract <p>Publications prior to March 1981 were surveyed to determine inlet flow dynamic distortion prediction methods and to catalogue experimental and analytical information concerning inlet flow dynamic distortion prediction methods and to catalogue experimental and analytical information concerning inlet flow dynamics at the engine-inlet interface of conventional aircraft (excluding V/STOL). The sixty-five publications found are briefly summarized and tabulated according to topic and are cross-referenced according to content and nature of the investigation (e.g., predictive, experimental, analytical and types of tests). Three appendices include lists of references, authors, organizations and agencies conducting the studies. Also selected materials -- summaries, introductions and conclusions -- from the reports are included. Few reports were found covering methods for predicting the probable maximum distortion. The three predictive methods found are those of Melick, Jacox and Motycka. The latter two require extensive high response pressure measurements at the compressor face, while the Melick Technique can function with as few as one or two measurements.</p>			
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