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SUPERSONIC THROUGHFLOW FANS

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SUPERSONIC THROUGHFLOW FANS

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ABSTRACT

Increased need for more efficient long-range supersonic flight has revived interest in the supersonic throughflow fan as a possible component for advanced high-speed propulsion systems. A fan that can operate with supersonic inlet axial Mach numbers would reduce the inlet losses incurred in diffusing the flow from supersonic Mach numbers to a subsonic one at the fan face. In addition, the size and weight of an all-supersonic inlet will be substantially lower than those of a conventional inlet. However, the data base for components of this type is practically nonexistent. Therefore, in order to furnish the required information for assessing the potential for this type of fan, the NASA Lewis Research Center has begun a program to design, analyze, build, and test a fan stage that is capable of operating with supersonic axial velocities from inlet to exit. The objectives are to demonstrate the feasibility and potential of supersonic throughflow fans, to gain a fundamental understanding of the flow physics associated with such systems, and to develop an experimental data base for design and analysis code validation.

This presentation provides a brief overview of past supersonic throughflow fan activities; discusses technology needs; describes the design of a supersonic throughflow fan stage, a facility inlet, and a downstream diffuser; and presents th results from the analysis codes used in executing the design. Also presented is a unique engine concept intended to permit establishing supersonic throughflow within the fan on the runway and maintaining the supersonic throughflow condition within the fan throughout the flight envelope.

SUPERSONIC THROUGHFLOW FANS FOR HIGH-SPEED AIRCRAFT

The NASA Lewis Research Center has embarked on a program to develop the technology for supersonic throughflow fans applicable to high-speed aircraft propulsion systems. We feel this technology could revolutionize high-speed aircraft design and performance.
(ref. 1-2)

SUPERSONIC THROUGHFLOW FANS FOR HIGH-SPEED AIRCRAFT

OUTLINE

This presentation provides a brief overview of past supersonic throughflow activities; discusses technology needs; describes the design and analysis of a supersoni throughflow fan stage, a facility inlet, and a downstream diffuser; and presents a unique engine concept incorporating a supersonic throughflow fan.

OUTLINE

- BACKGROUND
- TECHNOLOGY NEEDS AND STATUS
- NASA STF FAN PROGRAM
- UNIQUE ENGINE CONCEPT
- SUMMARY

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BACKGROUND

Ferri, in 1956, was the first to point out the potential advantages of supersonic inflow compression systems. In 1961, Savage, Boxer, and Erwin studied the starting characteristics in transitioning to supersonic inflow. Under Air Force sponsorship in 1967, General Applied Science Laboratory (GASL), with Detroit Diesel Allison (DDA) as a subcontractor, and United Technologies Research Center (UTRC) conducted design studies and proposed turbojet engine concepts incorporating supersonic throughflow compressors. Also in 1967, Boxer proposed a high-bypass-ratio turbofan engine/ramjet combination with a variable-pitch supersonic inflow compressor. In 1975, Breugelmans conducted the most thorough supersonic throughflow fan experiment to date. In 1978, Franciscus presented the results of his first analysis showing significant payoffs of supersonic throughflow fan engines for supersonic cruise aircraft. (refs. 3-12)

BACKGROUND

- FERRI WAS FIRST TO PROPOSE SUPERSONIC INFLOW COMPRESSORS (1956)
- SAVAGE, BOXER, AND ERWIN STUDIED STARTING CHARACTERISTICS (1961)
- GASL/DDA AND UTRC PROPOSED TURBOJET ENGINE CONCEPTS (1967)
- BOXER PROPOSED A TURBOFAN/RAMJET COMBINATION (1967)
- BREUGELMANS CONDUCTED MOST THOROUGH EXPERIMENT TO DATE (1975)
- FRANCISCUS PRESENTED THE RESULTS OF HIS FIRST STUDY (1978)

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TECHNOLOGY NEEDS

A technology need is to extend and validate the codes. In moving into the supersonic flow regime, where the data base is essentially nonexistent, applying computational methods in the design process should greatly enhance the quality of the experiment. Experiments are needed to obtain data for flow physics modeling and code validation and to demonstrate subsonic performance, transition, and supersonic performance.

TECHNOLOGY NEEDS

- EXTEND AND VALIDATE CODES
  - SUPERSONIC THROUGHFLOWS
  - ENDWALL BOUNDARY LAYER FLOWS
  - BLADE ROW INTERACTIONS
  - UNSTEADY FLOWS
  - DESIGN AND OFF-DESIGN PERFORMANCE PREDICTIONS
- CONDUCT EXPERIMENTS TO OBTAIN DATA FOR FLOW PHYSICS MODELING AND CODE VERIFICATION AND TO DEMONSTRATE PERFORMANCE
  - SUBSONIC PERFORMANCE
  - TRANSITION
  - SUPERSONIC PERFORMANCE
  - CHOKE, STALL, AND UNSTART
  - DISTORTION TOLERANCE
  - FLUTTER AND FORCED RESPONSE

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CODE STATUS

The indicated codes have been modified to accommodate supersonic throughflow velocities. However, because of the lack of data, the codes, with the exception of the three-dimensional parabolized Navier-Stokes code, have not been validated. This code has been validated for duct flows. All of the codes were used in the design and analysis of the NASA supersonic throughflow fan experiment. They will be validated as experiment results become available. (refs. 13-22)

CODE STATUS

THE FOLLOWING DESIGN AND ANALYSIS CODES HAVE BEEN MODIFIED AND APPLIED, BUT NOT YET VALIDATED FOR THE SUPERSONIC FLOW REGIME:

- AXISYMMETRIC DESIGN CODE (CROUSE)
- 1-D STAGE STACKING CODE (STEINKE)
- AXISYMMETRIC OFF-DESIGN CODE (CROUSE)
- QUASI-3-D THIN SHEAR LAYER NAVIER-STOKES CODE (CHIMA)
- 3-D PASSAGE AVERAGE STAGE CODE (ADAMCZYK)
- SUPERSONIC THROUGHFLOW FLUTTER CODE (RAMSEY)
- 3-D EULER CODE WITH 2-D BOUNDARY LAYER MODEL (DENTON)
- 3-D PARABOLIZED NAVIER-STOKES CODE (PNS)*
- 3-D UNSTEADY EULER CODE (WHITFIELD)

*VALIDATED FOR DUCT FLOW

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NASA SUPERSONIC THROUGHFLOW FAN PROGRAM

In discussing the NASA supersonic throughflow fan design, particular attention will be given to the results obtained from the analysis codes and to how they were used to guide the design. The detailed flow physics gleaned from the codes will be highlighted.

NASA

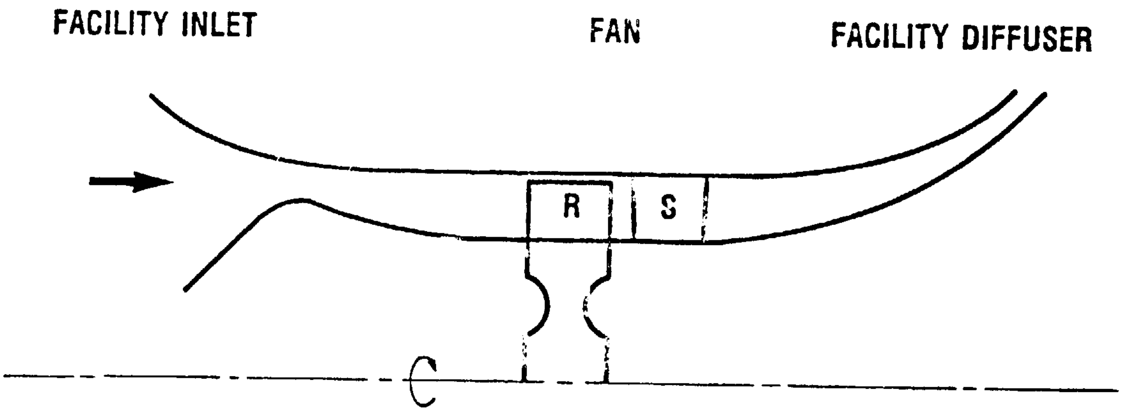
SUPERSONIC THROUGHFLOW FAN

PROGRAM

SUPERSONIC THROUGHFLOW FAN

This figure depicts a supersonic throughflow fan, the facility inlet needed to accelerate the flow to supersonic velocities at the fan face, and the diffuser needed downstream to decelerate the supersonic flow leaving the fan to subsonic conditions downstream. The design fan-face Mach number is 2.0 and the exit Mach number is 2.9. The fan was designed with a constant annulus area. The design pressure ratio and tip speed were selected to be representative of those required of a turbofan engine fan operating at supersonic cruise conditions. (ref. 2)

SUPERSONIC THROUGHFLOW FAN



PRESSURE RATIO 2.45
AXIAL MACH NUMBER
INLET 2.0
EXIT 2.9
ROTOR TIP SPEED .... 1500 ft/sec

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DESIGN CRITERIA

This figure presents the design criteria established to guide design of the fan, th facility inlet, and the downstream diffuser.

DESIGN CRITERIA

APPLY COMPUTATIONAL TWO- AND THREE-DIMENSIONAL INVISCID AND VISCOUS CODES TO

- FAN
  - ENSURE STARTED CONDITION
  - MAINTAIN SUPERSONIC THROUGHFLOW VELOCITIES THROUGHOUT COMPRESSION SYSTEM
  - ENSURE THAT SHOCK STRUCTURE IS CAPTURED WITHIN BLADED PASSAGES
  - CONTROL SUCTION AND PRESSURE SURFACE GRADIENTS TO MINIMIZE STRENGTH OF INTERNAL COMPRESSION AND EXPANSION WAVE SYSTEM
- FACILITY INLET
  - ACHIEVE UNIFORM VELOCITY DISTRIBUTION AT FAN INLET
  - MINIMIZE ENDWALL BOUNDARY LAYERS ENTERING FAN
- FACILITY DIFFUSER
  - MINIMIZE DIFFUSION LOSSES THROUGH A SERIES OF CONTROLLED WEAK COMPRESSION WAVES
  - ENSURE STARTED CONDITION

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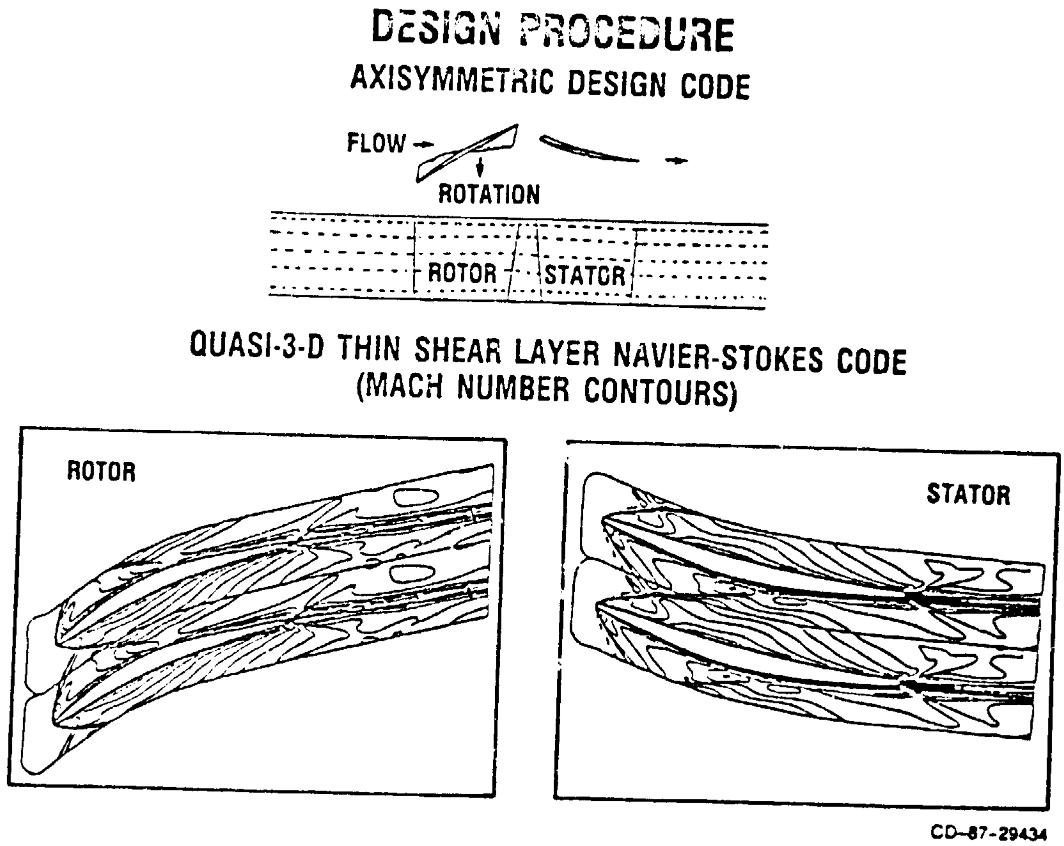
DESIGN PROCEDURE - AXISYMMETRIC DESIGN CODE AND QUASI-3-D THIN SHEAR LAYER NAVIER-STOKES CODE

In the design of the fan an axisymmetric design code was used to obtain initial blade shapes. The quasi-three-dimensional thin shear layer Navier-Stokes code was used to analyze the design. The design was then adjusted by using the design codes, and the process was repeated until the desired loading distributions and wave patterns were achieved. The Mach number contours for the rotor and stator show that the waves off the leading edge are contained within the bladed passage. Also, the expansion waves off the suction surface tend to cancel the compression waves off the pressure-surface leading edge, thus reducing the pressure gradient along the suction surface. At the trailing edge the strength of the expansion and compression waves was minimized by controlling the loading near the trailing edge. (refs. 13,15)

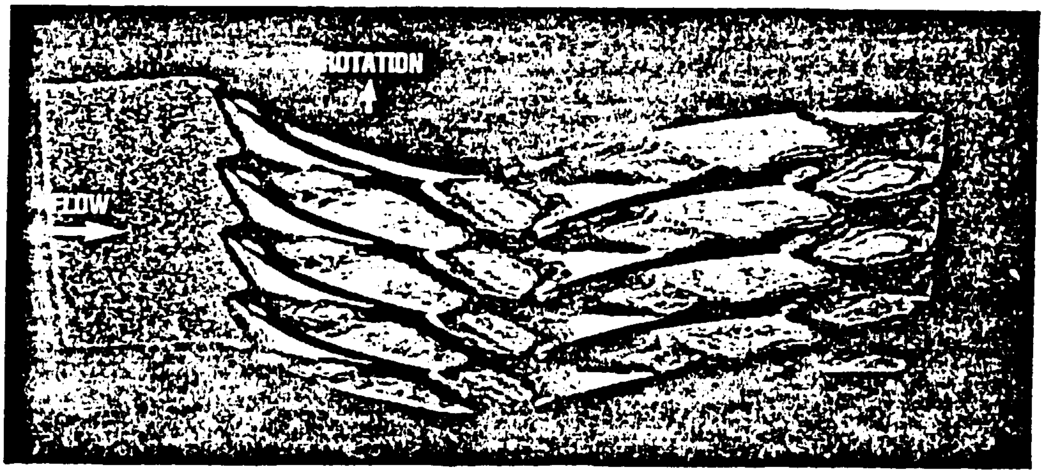


3-D UNSTEADY EULER CODE SHOWING INTERACTIVE WAVE PATTERNS

This figure shows results obtained from a three-dimensional unsteady Euler code used to study the rotor/stator flowfield interactions with supersonic throughflow. Computer graphics were used to obtain the interactive wave patterns for a given index of the rotor relative to the stator. The picture can be thought of as a schlieren photograph with the light patterns being expansion waves and the dark patterns, compression waves. The effect of the time-dependent flowfields behind the rotor on the stator flowfield can best be seen from the next figure. (ref. 22)

3-D UNSTEADY EULER CODE
INTERACTIVE WAVE PATTERNS

ROTOR STATOR



LIGHT = EXPANSION
DARK = COMPRESSION

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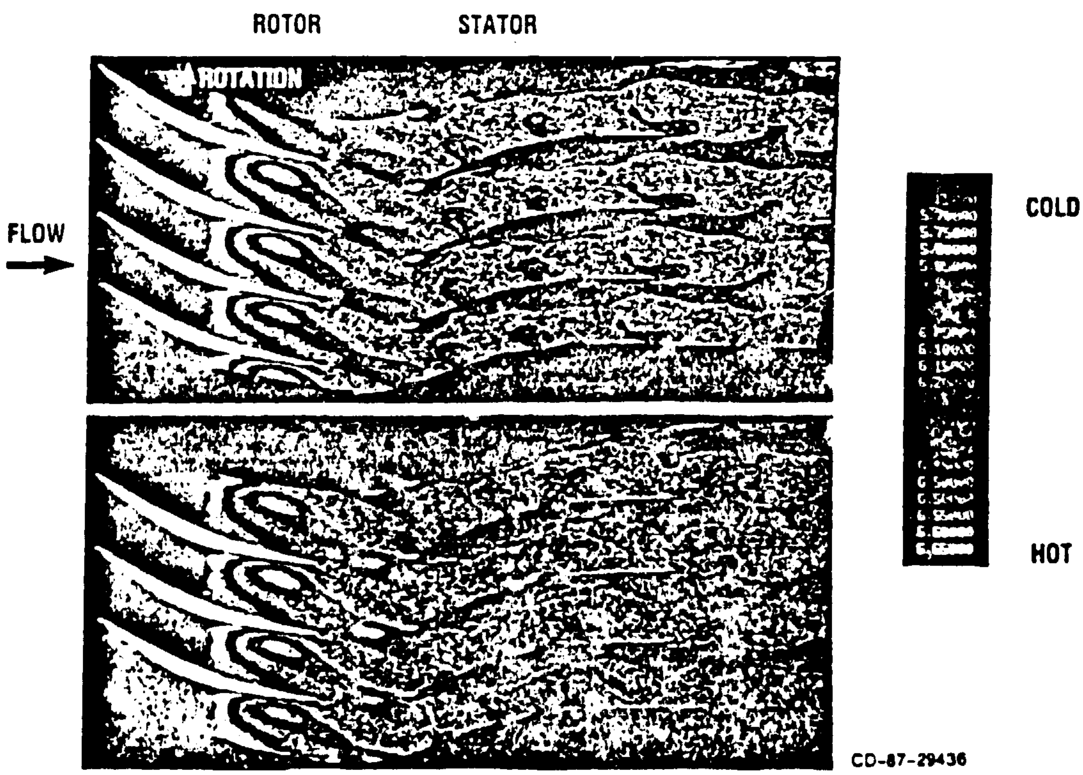
3-D UNSTEADY EULER CODE SHOWING STAGNATION ENTHALPY

This figure shows the stagnation enthalpy, and thus temperature, for two different indexes of the rotor blades relative to the stators. The interactive wave patterns within and exiting the rotor result in a time-dependent flowfield entering the stator. This unsteady flowfield relative to the stator appears to result in cyclic movement fore and aft of the stator leading-edge compression wave, which emanates from the pressure surface. Wave motion is nonlinear, with more energy being added when the shock moves forward than is subtracted when the shock moves rearward. Further analysis is needed to fully understand this phenomenon. The cyclic nature of the local temperature is apparent from the difference in the magnitudes of the local white (highest temperature) regions. However, the unsteady aspects of the flowfield can best be seen from a motion picture generated from the three-dimensional unsteady Euler code analysis. (ref. 22)

3-D UNSTEADY EULER CODE
STAGNATION ENTHALPY



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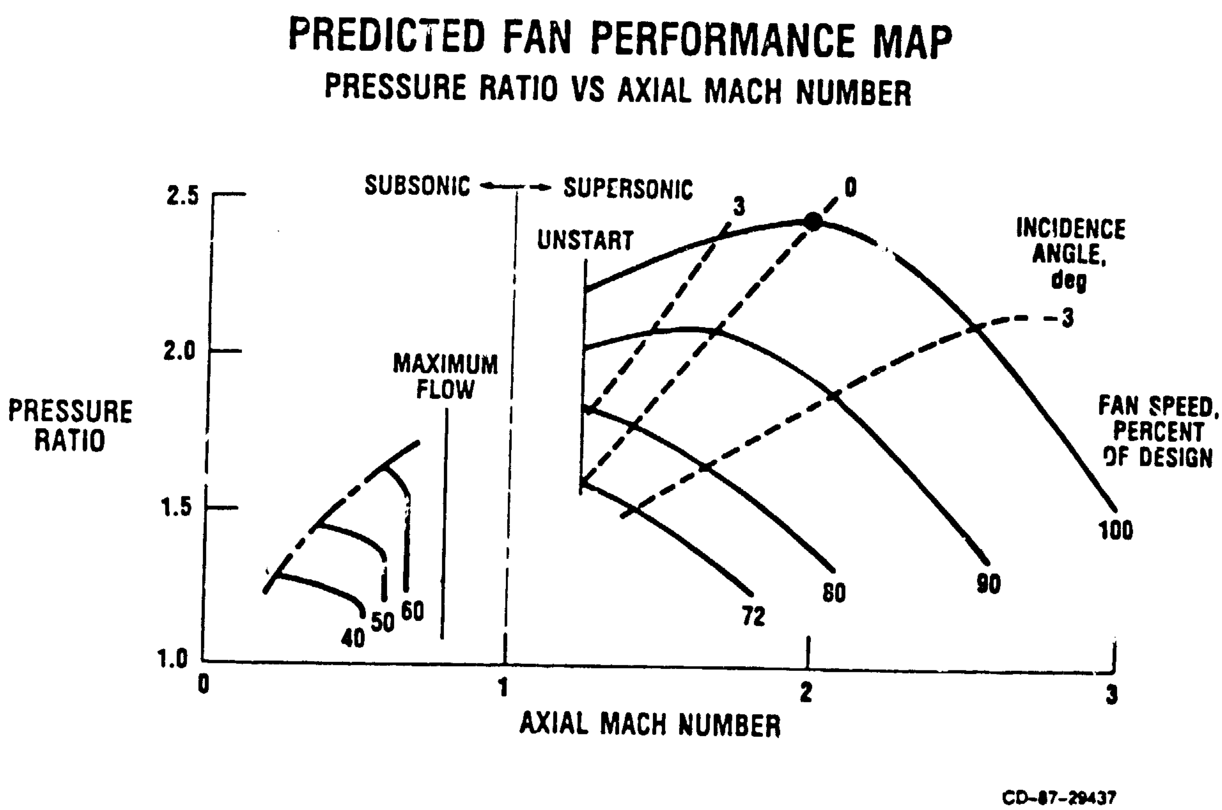
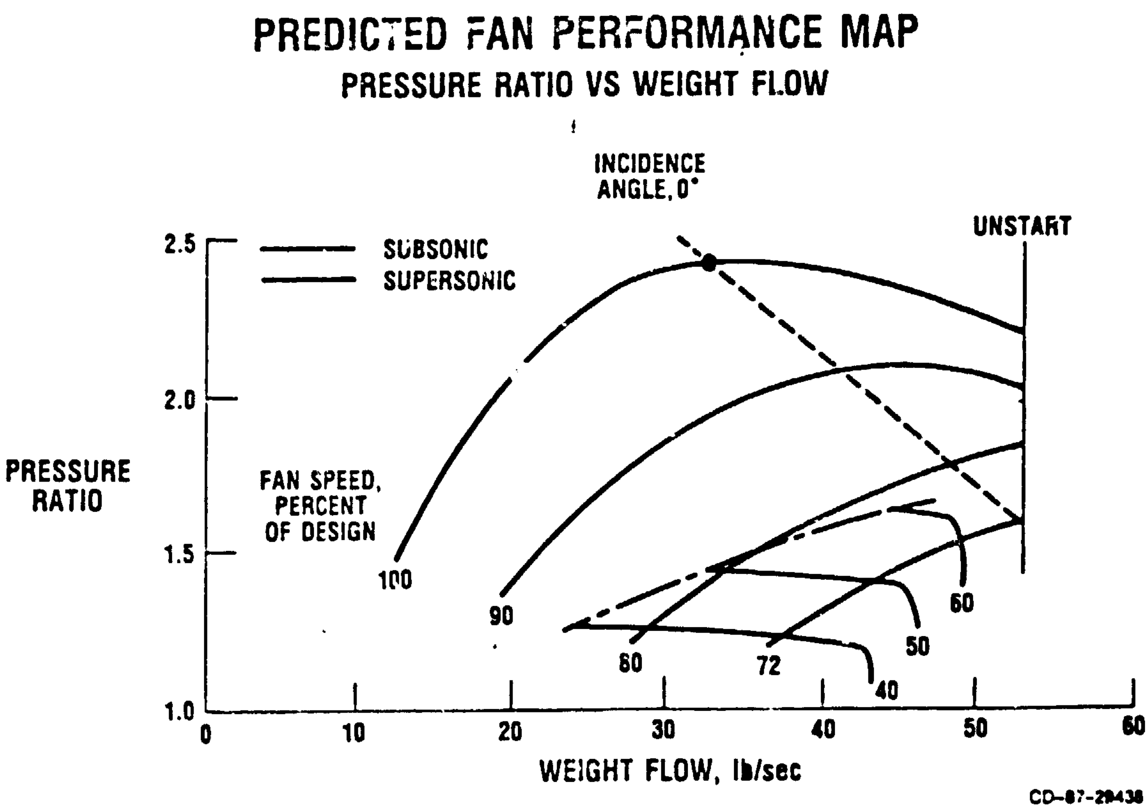
PREDICTED FAN PERFORMANCE MAP - PRESSURE RATIO VERSUS AXIAL MACH NUMBER

The predicted performance map for the supersonic throughflow fan was derived by using a combination of codes including the off-design axisymmetric code and the quasi-three-dimensional viscous code. Presenting the performance as a function of inlet axial Mach number results in a performance map similar to subsonic/transonic fan maps.



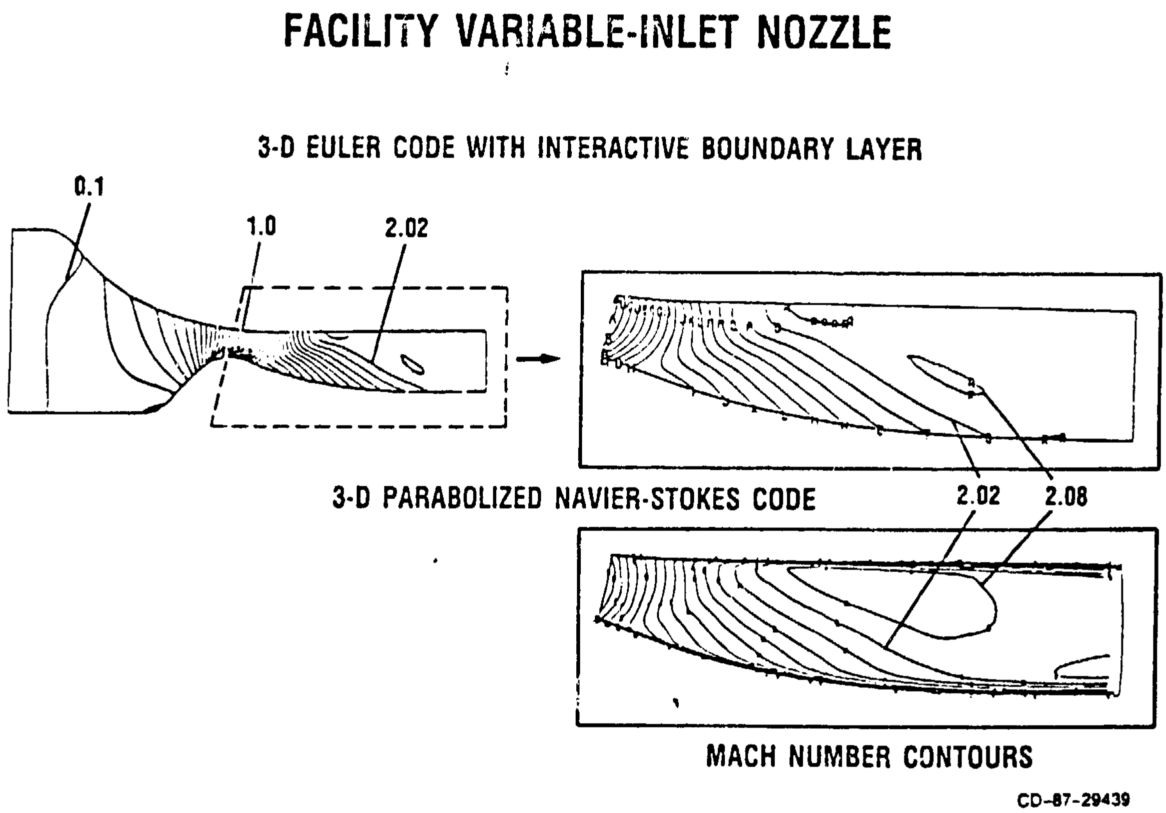
PREDICTED FAN PERFORMANCE MAP - PRESSURE RATIO VERSUS WEIGHT FLOW

This figure reflects the reduction in flow capacity on the supersonic throughflow side of the map as the inlet axial Mach number is increased.



FACILITY VARIABLE-INLET NOZZLE MACH NUMBER CONTOURS

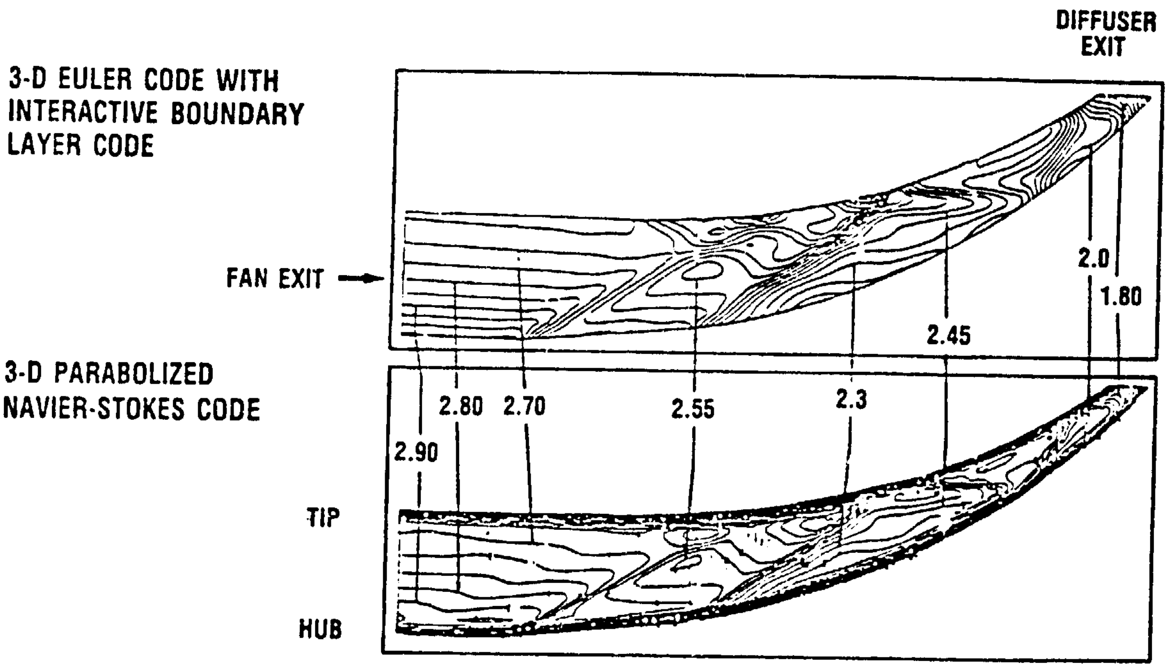
This figure presents the results obtained from a three-dimensional Euler code with an interactive boundary layer routine and from a three-dimensional parabolized Navier-Stokes code in analyzing the flowfield of the variable-inlet nozzle. The nozzle was positioned to achieve the design axial Mach number of 2.0. Good agreement existed between the codes. The codes indicated that at the design condition the flow was radially uniform at the fan face and the wall boundary layers were relatively thin. (refs 19-21)



FACILITY VARIABLE DIFFUSER MACH NUMBER CONTOURS

A similar analysis was conducted for the diffuser as for the inlet. Again, good agreement was obtained between the Euler and the viscous codes. The diffuser is designed to diffuse the flow from a fan exit Mach number of approximately 2.9 to Mach 1.8, primarily through two weak compression waves. (refs. 19-21)

FACILITY VARIABLE DIFFUSER



MACH NUMBER CONTOURS

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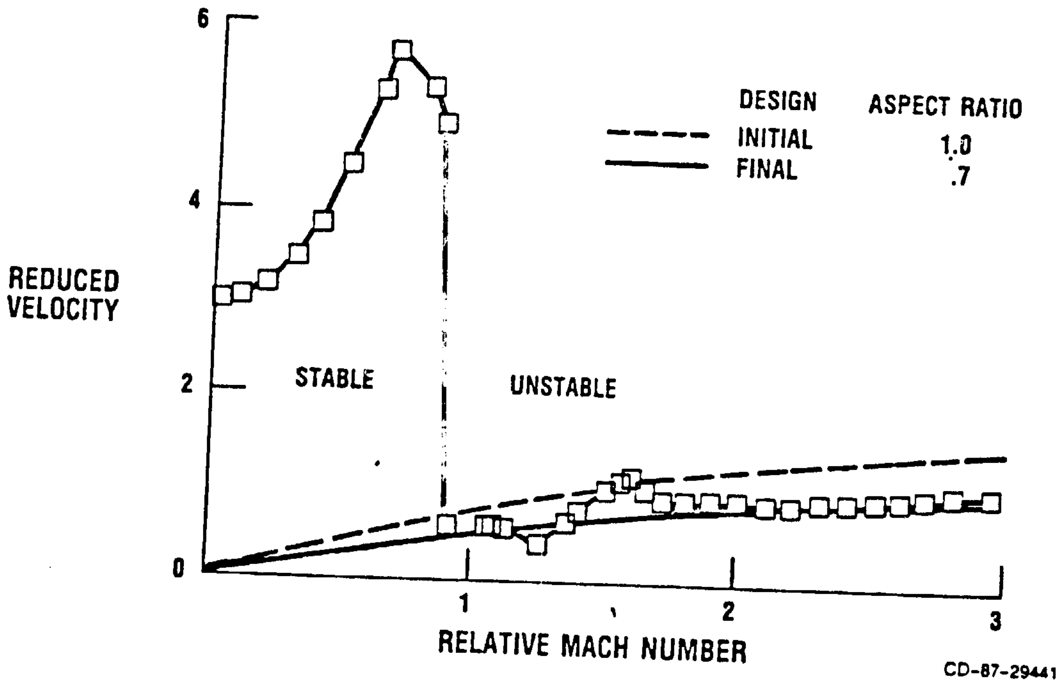
This figure presents the results from an analysis of the flutter potential of the supersonic throughflow fan. Note the large reduction in stable operating range indicated by the supersonic throughflow flutter analysis at supersonic relative velocities. Even though the initially designed rotor blade was relatively low in aspect ratio, the analysis indicated a potential for supersonic torsional flutter The design aspect ratio was further reduced in the final design to bring the roto into the stable operating range. (ref. 18)

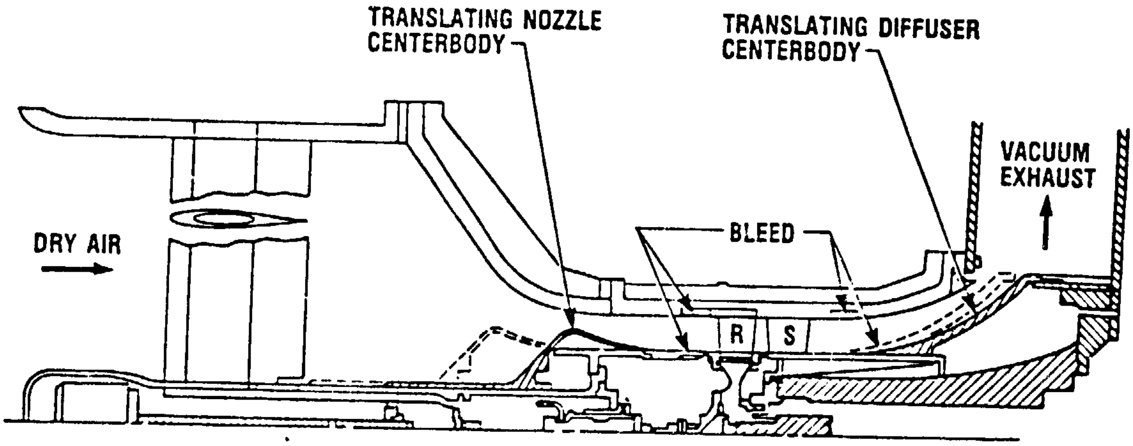
AEROELASTIC ANALYSIS OF SUPERSONIC THROUGHFLOW FAN TORSIONAL FLUTTER



TEST PACKAGE

This figure is a layout of the supersonic throughflow fan test package. The variable-inlet nozzle and the variable downstream diffuser will be used to provide control over the fan-face Mach number and the diffusion of the supersonic fan exit velocities to subsonic conditions entering the exhaust system. Boundary layer bleed capability is provided at the inlet to the fan and the diffuser.

TEST PACKAGE



PRESSURE RATIO 2.45
INLET AXIAL MACH NUMBER 2.0
ROTOR TIP SPEED .... 1500 ft/sec
TIP DIAMETER .... 20 in.

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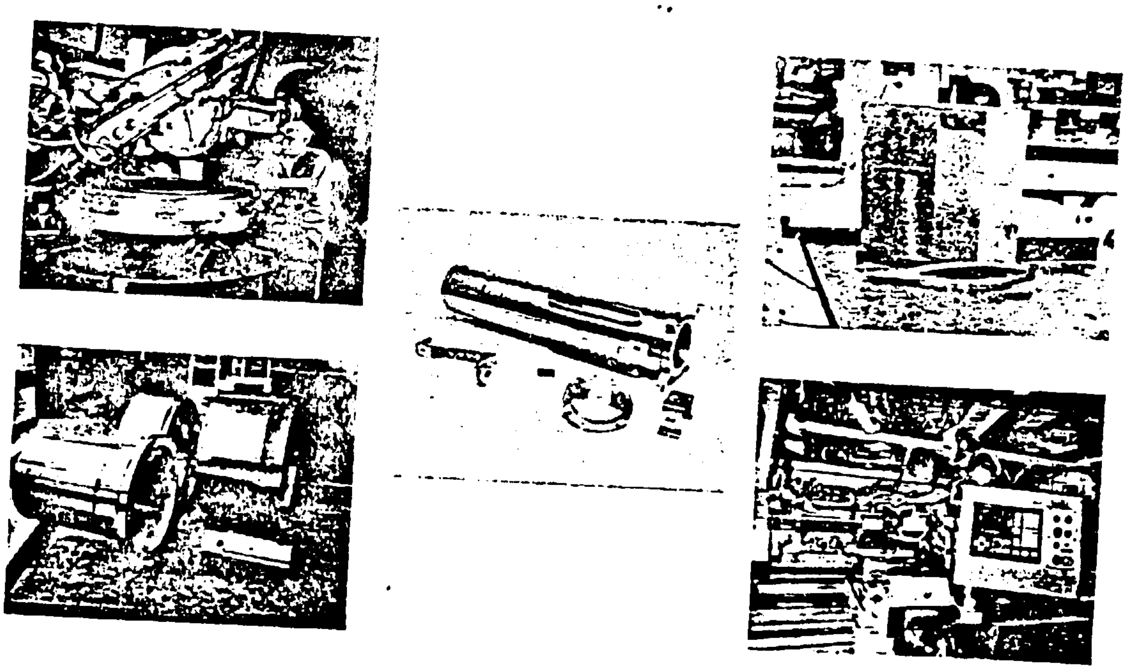
HARDWARE FABRICATION

This figure shows some of the fan hardware in various phases of completion.

HARDWARE FABRICATION



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UNIQUE HIGH-SPEED ENGINE CONCEPT

Some of the problems raised in connection with supersonic throughflow fans for supersonic aircraft are how to "fly" such an engine system, how the fan can be made to transition to the supersonic side of the performance map and at what flight speed this would occur, how to select the design point, and unstart. The following figures present a unique engine concept that solves these problems. The transition to supersonic throughflow within the fan component is made while the airplane is on the runway. This concept's many advantages will be discussed.

UNIQUE HIGH-SPEED ENGINE CONCEPT

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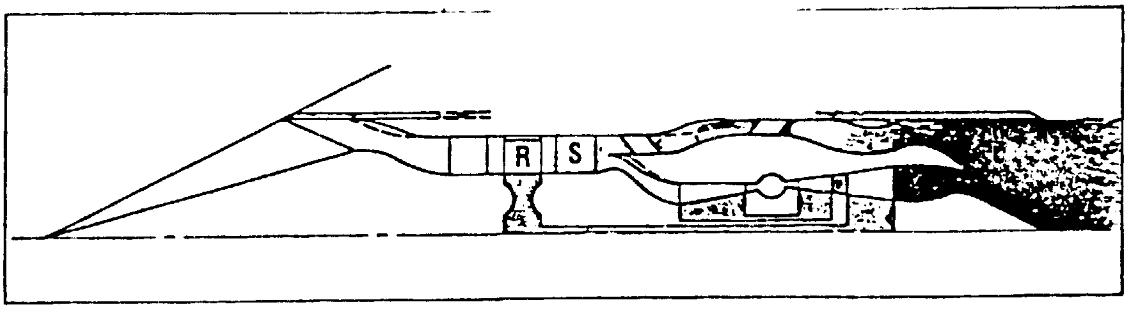
SUPERSONIC THROUGHFLOW TURBOFAN HIGH-SPEED ENGINE CONCEPT - MACH 3 DESIGN

Shown is a supersonic throughflow turbofan engine. However, the basic concept also applies to turbojet and airturboramjet cycles. The concept incorporates a short annular inlet with a variable capture and throat area to ease transitioning to supersonic throughflow within the fan on the runway and to maintain supersonic flow at the fan face throughout the flight envelope. Located downstream of the fan are annular supersonic/subsonic diffusers, one in the bypass duct and the other in the core inlet duct. The core inlet also features a variable capture area to help in flow matching and in optimizing performance. The flow entering the core compressor is diffused to subsonic conditions throughout the flight envelope. However, the duct flow is diffused subsonically for only subsonic flight Mach numbers and remains supersonic for supersonic flight. The variable-geometry features in the diffuser and nozzle are intended to achieve these goals. Duct burning may or may not be required. Some of the advantages of this concept are no forward transmission of fan noise on takeoff, ease of meeting pressure ratio requirements for takeoff and aircraft acceleration through Mach 1, potentially good subsonic cruise performance for overland operation, and, it is hoped, no variable geometry in the fan rotor.

The flowpath shown for the turbofan is consistent with the Mach 3 design. The design flight Mach number has a significant effect on the fan geometry.

SUPERSONIC THROUGHFLOW TURBOFAN HIGH-SPEED ENGINE CONCEPT

MACH 3 DESIGN



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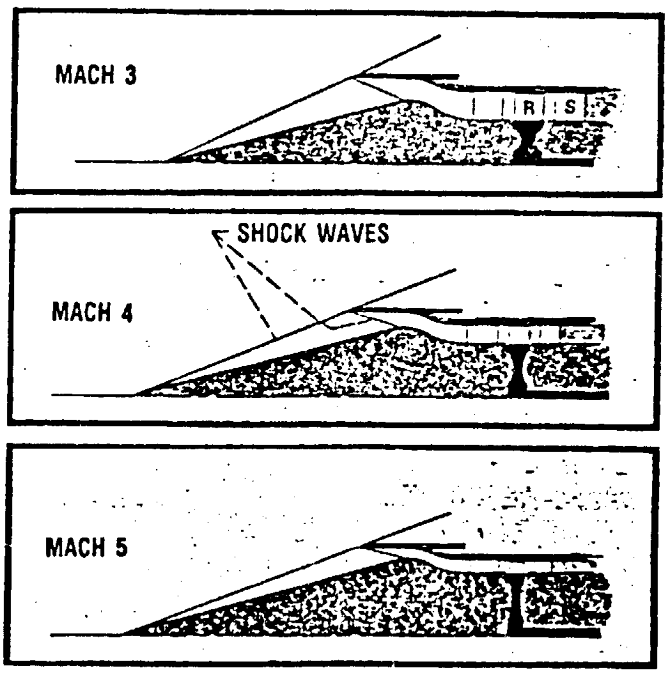
VARIOUS FLIGHT MACH NUMBER DESIGNS FOR SUPERSONIC THROUGHFLOW TURBOFAN

This figure illustrates the effect of flight Mach number on the fan geometry. For the Mach 3 condition, as shown in the previous figure, the fan hub/tip ratio is about 0.7. As the flight Mach number is increased, the passage height decreases and the hub/tip ratio increases. At Mach 5 the fan hub/tip ratio is above 0.8. The reduction in the fan tip diameter in relation to the inlet diameter is adequate to achieve the desired change in throat area with acceptable axial translation of the nozzle. By limiting the reduction in fan diameter, the gooseneck is minimized and the strength of the expansion and compression waves is reduced during supersonic operation. To achieve the low hub/tip ratio typical of transonic fans while achieving satisfactory supersonic operation would require a prohibitive gooseneck.

VARIOUS FLIGHT MACH NUMBER DESIGNS FOR SUPERSONIC THROUGHFLOW TURBOFAN



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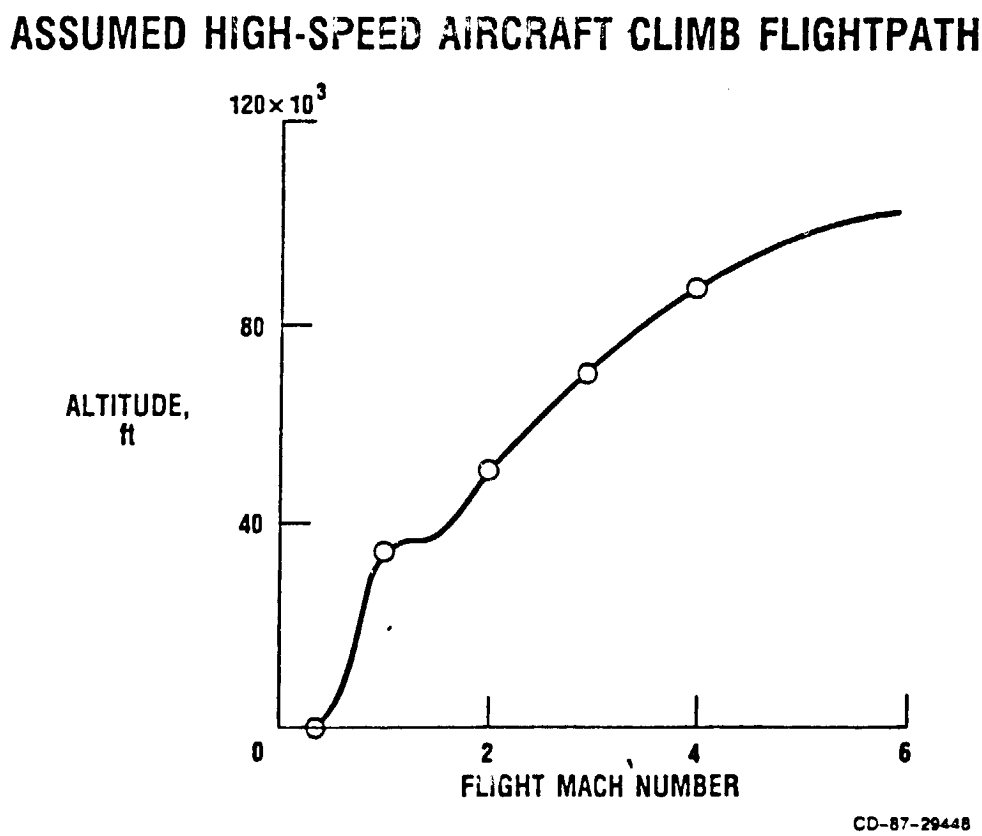
ASSUMED HIGH-SPEED AIRCRAFT CLIMB FLIGHTPATH

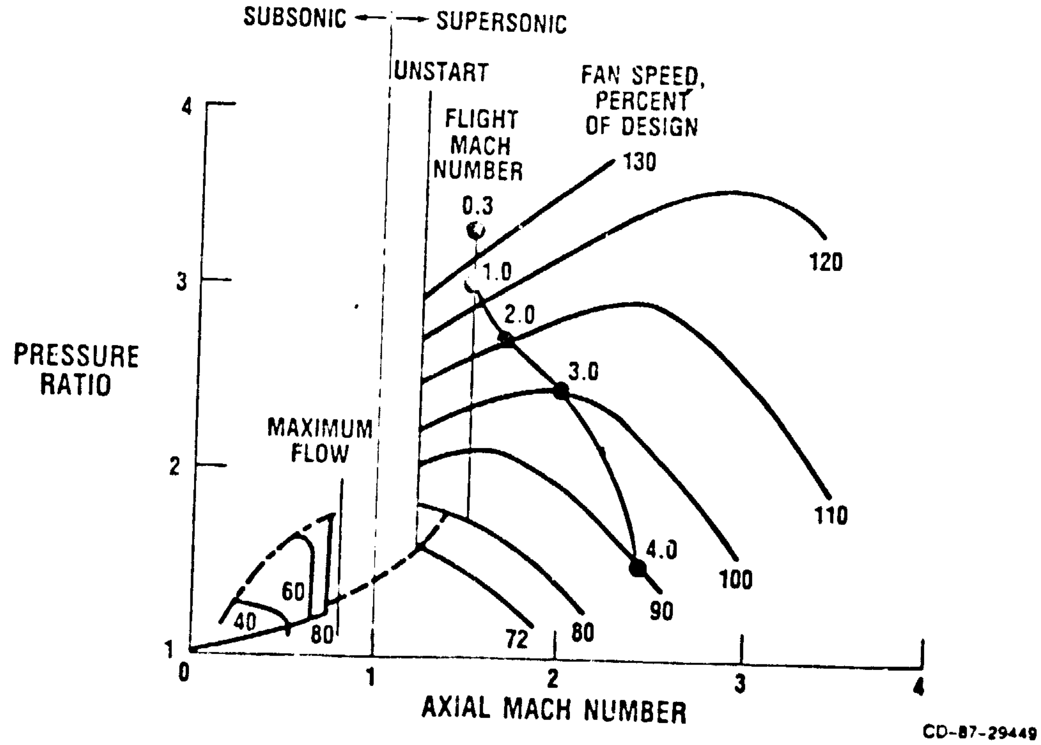
This figure shows the assumed flightpath used in examining how the inlet would be configured over the flight envelope. Mach 0.3 was assumed for takeoff, transition to supersonic flight Mach numbers at 35 000 ft, and Mach 3.0 cruise at 70 000 ft.



SUPERSONIC THROUGHFLOW TURBOFAN FAN OPERATING LINE

This fan performance map shows the startup, transition, and flight operating lines. The design point was assumed to be Mach 3.0 cruise with a fan-face Mach number of 2.0 and a pressure ratio of approximately 2.5, consistent with that derived from mission analysis studies conducted by Franciscus for a Mach 3.0 transport aircraft. These same studies indicated fan pressure ratio requirements of approximately 3.3 and 3.0 for takeoff and aircraft transition to supersonic flight Mach numbers, respectively. The inlet is set to the lower design fan-face Mach number during takeoff and transition to maximize the flow and to minimize inlet bleed requirements. Predicted maximum subsonic flow before startup is shown along with the predicted unstart boundary on the supersonic side of the performance map. Note the large supersonic flow range. The startup method is to increase speed to approximately 80 percent and then close the inlet nozzle slightly. In so doing, it is predicted that the normal shock will transition through the fan. The concept requires a low load line to keep the fan out of stall during subsonic operation, even though the fan incidence will be high just prior to transitioning. As the normal shock passes through the fan, the operating point will jump to the supersonic side along the 80 percent speed line. The inlet will be adjusted to achieve the desired fan-face Mach number, and the speed will then be increased to takeoff conditions. The reverse procedure would be employed during landing.

SUPERSONIC THROUGHFLOW TURBOFAN FAN OPERATING LINE

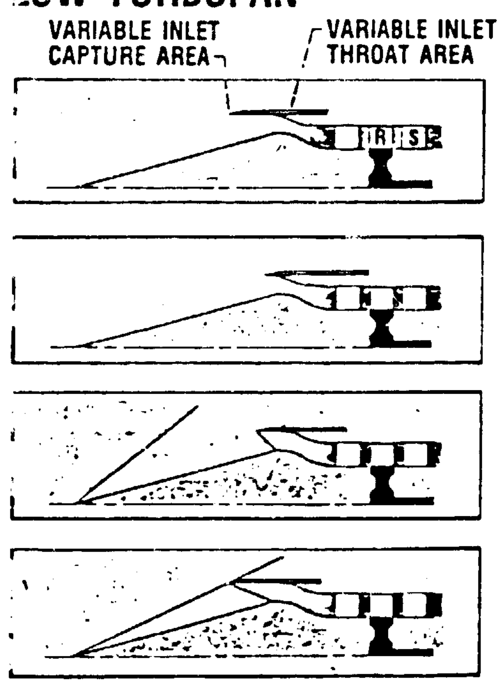


INLET GEOMETRY CONFIGURATIONS FOR SUPERSONIC THROUGHFLOW TURBOFAN

This figure shows the inlet geometry configurations for Mach 0 to 0.3, 1.0, 2.0, an 3.0 (the assumed design point). From Mach 0 to 0.3 the inlet cowl is extended forward so that the flowpath will converge to accelerate the air to Mach 1.0 at the throat. The throat area is set relative to the fan inlet area to achieve the desir fan-face Mach number, in this case 1.5. As the flight Mach number is increased to supersonic conditions, a bow wave is formed off the inlet spike. As the Mach numbe becomes supersonic behind the shock, the inlet cowl is set to control the position of the internal reflected wave. At the design point of Mach 3.0 the cowl is positioned such that the leading-edge bow wave is attached to the cowl lip. The throat is opened up at Mach 2.0 and 3.0 flight to achieve the desired fan-face Mach number

INLET GEOMETRY CONFIGURATIONS FOR SUPERSONIC THROUGHFLOW TURBOFAN

FLIGHT MACH NUMBER	FAN-FACE MACH NUMBER
0-0.3	1.5
1.0	1.5
2.0	1.7
3.0	2.0



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SUMMARY

In summary, mission studies conducted by Franciscus have shown significant benefits from supersonic throughflow fans. The design and analysis conducted on the NASA supersonic throughflow fan shows promise for such a stage. However, an experiment is strongly needed to demonstrate transition and to validate the computational codes. Off-design analysis is continuing with emphasis on the use of Chima's quasi-three-dimensional viscous code. Rotor/stator interactions will be investigated at off-design conditions by using Whitfield's code. The fan is now in fabrication, and testing is scheduled for the end of 1989.

SUMMARY

- OFF-DESIGN PERFORMANCE ANALYSIS CONTINUING
  - SUBSONIC/TRANSITION
  - SUPERSONIC
- UNSTEADY AERODYNAMIC ANALYSIS CONTINUING
  - ROTOR/STATOR INTERACTION
- STF FAN, FACILITY INLET, AND DOWNSTREAM DIFFUSER IN FABRICATION
- TESTING PROJECTED FOR END OF 1989

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