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# SUPERSONIC THROUGHFLOW FANS FOR HIGH-SPEED AIRCRAFT

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## SUMMARY

This paper 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 the 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.

## INTRODUCTION

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 subsonic 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 (refs. 1 and 2). 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.

## BACKGROUND

Ferri, in 1956, was the first to point out the potential advantages of supersonic inflow compression systems for supersonic aircraft propulsion (ref. 3). In 1961, Savage, Boxer, and Erwin (ref. 4) studied the starting characteristics in transitioning to supersonic inflow by operating a transonic rotor at an overdesign tip speed. 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 (refs. 5 and 6). Also in 1967, Boxer proposed a high-bypass-ratio turbofan engine/ramjet combination with a variable-pitch supersonic inflow fan (ref. 7). In 1975, Breugelmans (ref. 8) conducted the most thorough supersonic throughflow fan experiment to date. He tested a rotor designed for an inlet axial Mach number of 1.5. However, a mechanical failure precluded reaching the design speed, and only limited data were obtained. In 1978, Franciscus presented the results of his first analysis showing significant payoffs of supersonic throughflow fan engines for supersonic cruise aircraft (ref. 9). Results from his later analysis, confirming potential payoffs, are presented in references 10 to 12.

## TECHNOLOGY NEEDS

First, there is a need to extend and validate the computational codes for supersonic throughflows to include the associated endwall boundary-layer flows, blade row interactions, unsteady flows, and design and off-design performance predictions. 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 experimental program.

There is also a need to conduct experiments to obtain data for flow physics modeling and code validation and to demonstrate subsonic performance, transition, and supersonic performance. Choke, stall, and unstart (which all occur near Mach 1 for supersonic throughflow) need to be investigated. Fan distortion tolerance and the flutter and forced response must also be experimentally evaluated in the supersonic flow regime.

## CODE STATUS

The following design and analysis codes have been modified and applied to the supersonic throughflow fan, but they have not yet been validated for the supersonic flow regime with the exception of the three-dimensional parabolized Navier-Stokes code. This code was validated for duct flows.

(1) Axisymmetric design code (ref. 13)
(2) One-dimensional stage stacking code (ref. 14)
(3) Axisymmetric off-design code (unpublished work by J.E. Crouse)
(4) Quasi-three-dimensional thin shear layer Navier-Stokes code (ref. 15)
(5) Three-dimensional passage average stage code (refs. 16 and 17)
(6) Supersonic throughflow flutter code (ref. 18)
(7) Three-dimensional Euler code with two-dimensional boundary layer model (ref. 19 and unpublished material by J.D. Denton, ASME Turbomachinery Institute Course on Fluid Dynamics of Turbomachinery, July 18-27, 1983)
(8) Three-dimensional parabolized Navier-Stokes code (ref. 20)
(9) Three-dimensional unsteady Euler code (ref. 21)

Modification of the axisymmetric design code for supersonic flow proved to be a key to the design of the supersonic throughflow fan. This code generates blade geometry which can then be analyzed by the other codes. Each of the codes will be validated as experimental results become available.

# NASA SUPERSONIC THROUGHFLOW FAN

## Fan Aerodynamic Design

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.

Figure 1 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.

The following design criteria were established to guide the design of the baseline fan, the facility inlet, and the facility downstream diffuser:

(1) Set the hub and tip radii constant throughout the fan stage to limit severe three-dimensional effects.

(2) Apply computational two- and three-dimensional inviscid and viscous codes to the fan to (1) ensure started conditions, (2) maintain supersonic throughflow velocities throughout the compression system, (3) ensure that all shock structure is captured within the bladed passages, and (4) control suction and pressure surface gradients to minimize the strength of the internal compression and expansion wave system.

(3) Apply computation codes to the facility inlet to (1) achieve uniform velocity distribution at the fan inlet and (2) minimize endwall boundary layers entering the fan.

(4) Apply computation codes to the facility diffuser to (1) minimize diffusion losses through a series of controlled weak compression waves and (2) ensure started conditions.

In the design of the fan an axisymmetric design code was used to obtain initial blade shapes (fig. 2). The quasi-three-dimensional thin shear layer Navier-Stokes code was then used to analyze the design. The design was adjusted by using the axisymmetric design code, and the process was repeated until the desired loading distributions and wave patterns were achieved. In the lower half of the figure are the calculated Mach number contours from the quasi-three-dimensional thin shear layer Navier-Stokes code. 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.

Figure 3 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.

Figure 4 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.

The predicted performance map for the supersonic throughflow fan (fig. 5) 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. Also shown in the figure is the maximum flow condition calculated for subsonic flow and the unstart condition for the supersonic flow regime. Lines of constant incidence angles of 3°, 0°, and -3° are plotted on the map. The incidence angle range at design speed is greater than that normally predicted for subsonic/transonic fans.

The performance map presented in figure 6 is for the normal 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 Design

In order to test the fan over a range of Mach numbers and to provide transition from subsonic to supersonic flows, a variable-inlet nozzle is needed for the facility. 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 are presented in figure 7. 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.

## Facility Variable Diffuser Design

A variable diffuser was designed to diffuse the flow from a fan exit Mach number of approximately 2.9 to Mach 1.8 prior to dumping the flow into the Lewis central exhaust system. The diffusion is taken primarily through two weak compression waves. The variable diffuser will be used to ease starting conditions. A similar analysis was conducted for the diffuser as for the inlet. Again, good agreement was obtained between the Euler and the viscous codes (fig. 8).

### Fan Aeroelastic Analysis

A new supersonic flutter code was developed to analyze the fan blades. This code indicated that a blade redesign was required. The results from an analysis of the flutter potential of the supersonic throughflow fan are presented in figure 9. 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 rotor into the stable operating range.

### Test Package

The layout of the supersonic throughflow fan test package is presented in figure 10. 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. Bleed air for the rotor inlet hub will exhaust down the centerbody and out of the inlet struts. The exit hub bleed air will exhaust out the back of the test package. Photographs of some of the test package hardware are shown in figure 11 in various phases of completion.

## 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.

A conceptual design for a supersonic throughflow turbofan high-speed engine is shown in figure 12 for a Mach 3 design condition. 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. The effect of flight Mach number on the fan geometry is illustrated in figure 13. For the Mach 3 condition, as shown in figure 12, 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.

In order to examine how the inlet would be configured over the flight envelope, a climb flight path was established. The assumed flight path is shown in figure 14. 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.

The fan performance map (fig. 15) 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.

Figure 16 shows the inlet geometry configurations for Mach 0 to 0.3, 1.0, 2.0, and 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 desired 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 number 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.

## CONCLUDING REMARKS

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.

## REFERENCES


1. Wood, J.R., et al.: Application of Advanced Computational Codes in the Design of an Experiment for a Supersonic Throughflow Fan Rotor. ASME Paper 87-GT-160, May 1987 (NASA TM-88915).

2. Schmidt, J.F., et al.: Supersonic Through-Flow Fan Design. AIAA Paper 87-1746, June 1987 (NASA TM-88908).

3. Ferri, A.: Problems Related to Matching Turbojet Engine Requirements to Inlet Performances as Function of Flight Mach Number and Angle of Attack. Air Intake Problems in Supersonic Propulsion, J. Fabri, ed., Agardograph No. 27, AGARD, France, 1956, pp. 48-62.

4. Savage, M.; Boxer, E.; and Erwin, J.R.: Resume of Compressor Research at the NACA Langley Laboratory. J. Eng. Power, vol. 83, no. 3, July 1961, pp. 269-285.

5. Lipfert, F.W.: Supersonic Axial Velocity Compressor Study, Vol. 1, AFAPL-TR-67-68, 1967. (Avail. NTIS, AD-383675.)

6. Gadbois, S.E.; and Dunn, B.M.: Investigation of the Supersonic Through-Flow Compressor Concept, Vol. II, AFAPL-TR-67-60-VOL-2, 1967. (Avail. NTIS, AD-815963.)

7. Boxer, E.: The Variable-Pitch Supersonic Inflow Compressor and Its Application in a Hypersonic Engine. Conference on Hypersonic Aircraft Technology, NASA SP-148, 1967, pp. 401-416.

8. Breugelmans, F.A.E.: The Supersonic Axial Inlet Component in a Compressor. ASME Paper 75-GT-26, Mar. 1975.

9. Franciscus, L.C.: Supersonic Through-Flow Fan Engines for Supersonic Cruise Aircraft. NASA TM-78889, 1978.

10. Franciscus, L.C.: The Supersonic Fan Engine - An Advanced Concept in Supersonic Cruise Propulsion. AIAA Paper 81-1599, July 1981 (NASA TM-82657).

11. Franciscus, L.C.: Supersonic Fan Engines for Military Aircraft. AIAA Paper 83-2541, Oct. 1983 (NASA TM-83499).

12. Franciscus, L.C.: The Supersonic Through-Flow Turbofan for High Mach Propulsion. AIAA Paper 87-2050, June 1987 (NASA TM-100114).

13. Crouse, J.E.; and Gorrell, W.T.: Computer Program for Aerodynamic and Blading Design of Multistage Axial-Flow Compressors. NASA TP-1946, 1981.

14. Steinke, R.J.: STGSTK - A Computer Code for Predicting Multistage Axial Flow Compressor Performance by a Meanline Stage Stacking Method. NASA TP-2020, 1982.

15. Chima, R.V.: Explicit Multigrid Algorithm for Quasi-Three-Dimensional Viscous Flows in Turbomachinery. J. Propulsion Power, vol. 3, no. 5, Sept.-Oct. 1987, pp. 397-405.

16. Adamczyk, J.J.; Mulac, R.A.; and Celestina, M.L.: A Model for Closing the Inviscid Form of the Average-Passage Equation System. J. Turbomachinery, vol. 108, no. 2, Oct. 1986, pp. 180-186.

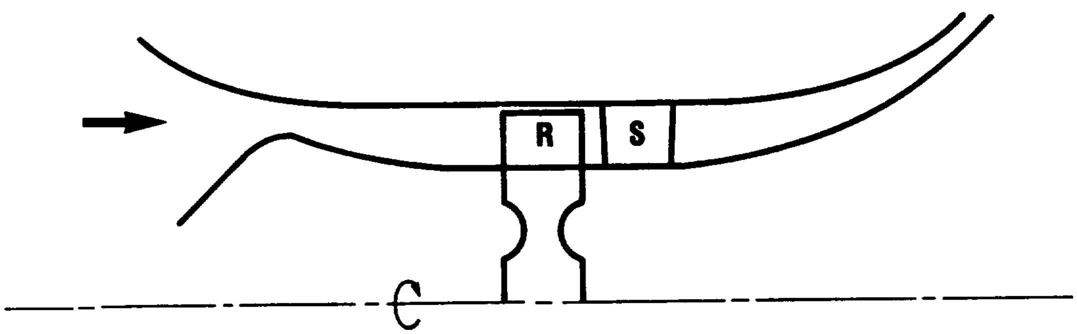
17. Celestina, M.L.; Mulac, R.A.; and Adamczyk, J.J.: A Numerical Simulation of the Inviscid Flow Through a Counterrotation Propeller. J. Turbomachinery, vol. 108, no. 2, Oct. 1986, pp. 187-193.

18. Ramsey, J.K.; and Kielb, R.E.: A Computer Program for Calculating Unsteady Aerodynamic Coefficients for Cascades in Supersonic Axial Flow. NASA TM-100204, 1987.

19. Denton, J.D.: An Improved Time-Marching Method for Turbomachinery Flow Calculations. J. Eng. Power, vol. 105, no. 3, July 1983, pp. 514-524.

20. Buggeln, R.C., et al.: Development of a Three-Dimensional Supersonic Inlet Flow Analysis. NASA CR-3218, 1980.

21. Whitfield, D.L., et al.: Three-Dimensional Unsteady Euler Solutions for Propfans and Counter-Rotating Propfans in Transonic Flow. AIAA Paper 87-1197, June 1987.




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

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Figure 1. - Supersonic throughflow fan.

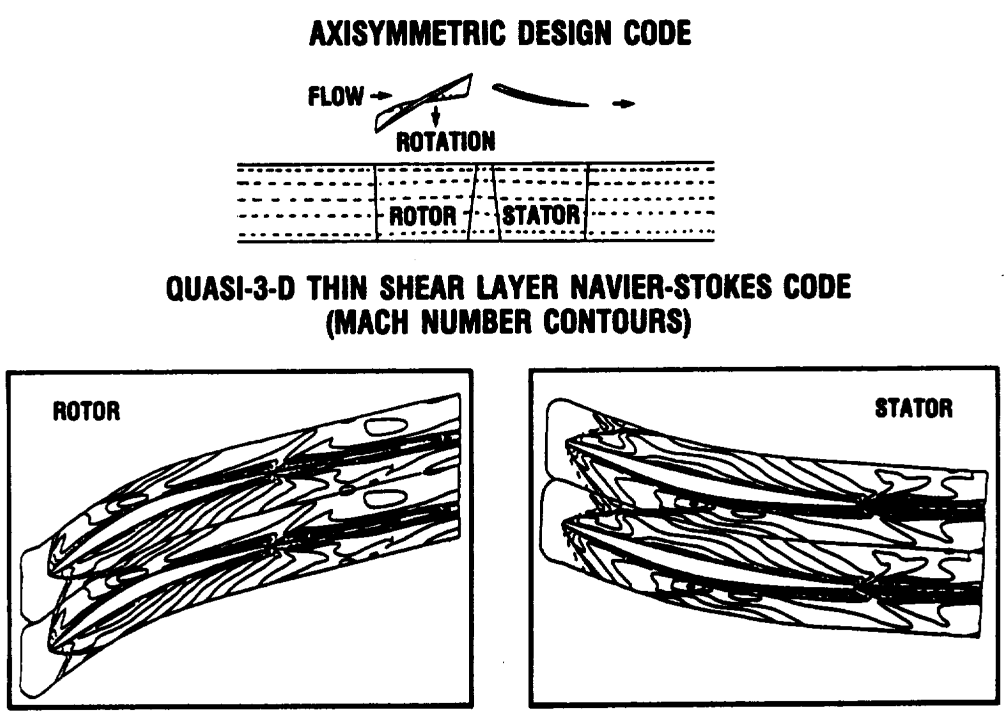


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Figure 2. - Design procedure.

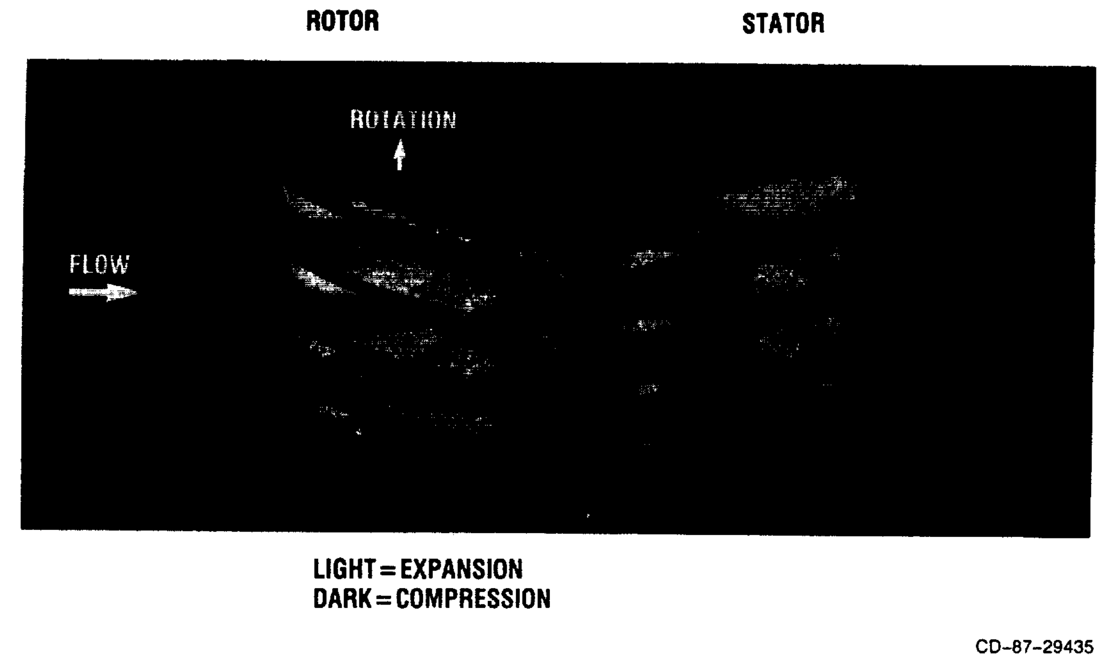


Figure 3. - Three-dimensional unsteady Euler code (interactive wave patterns).

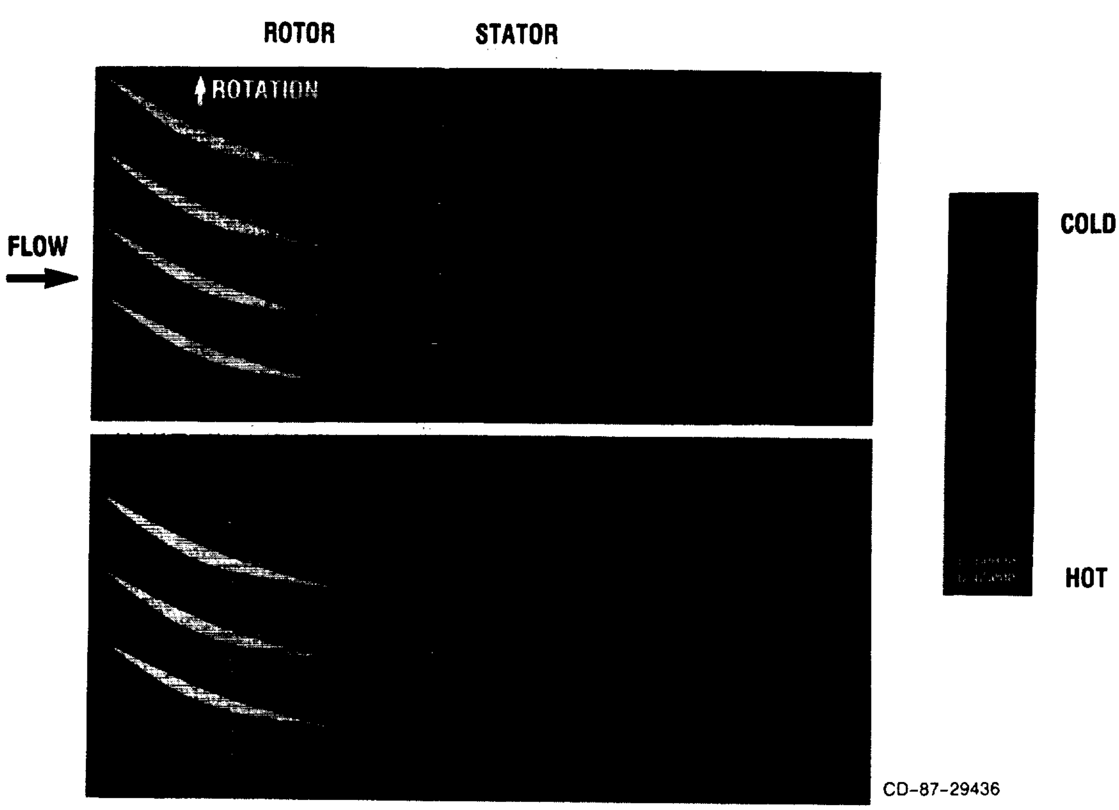


Figure 4. - Three-dimensional Euler code (stagnation enthalpy).

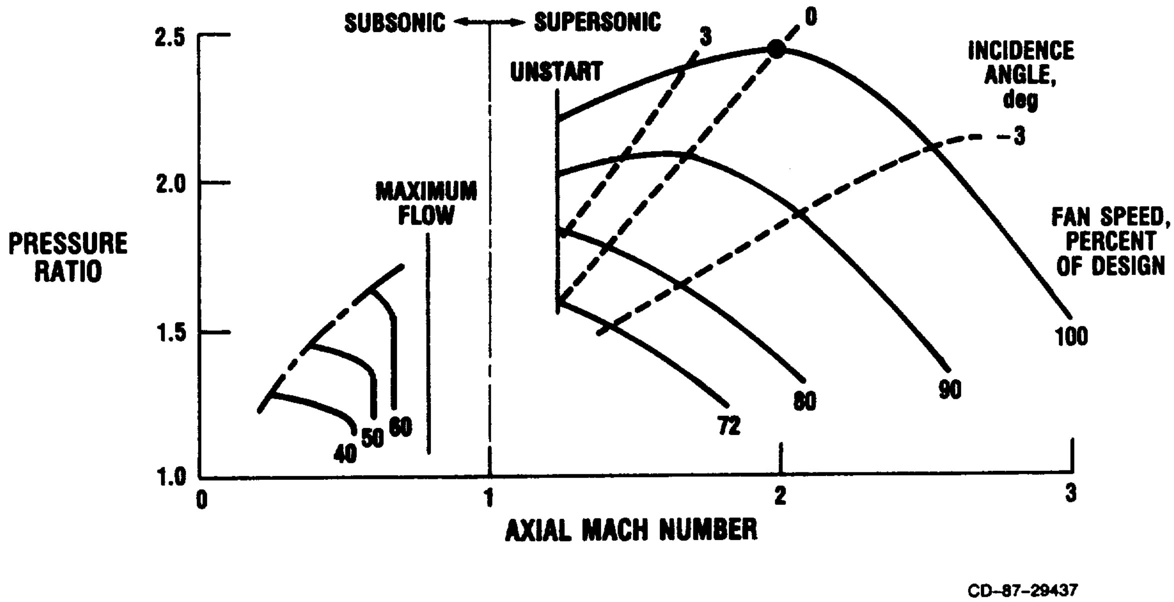


Figure 5. - Predicted fan performance map (pressure ratio versus axial Mach number).

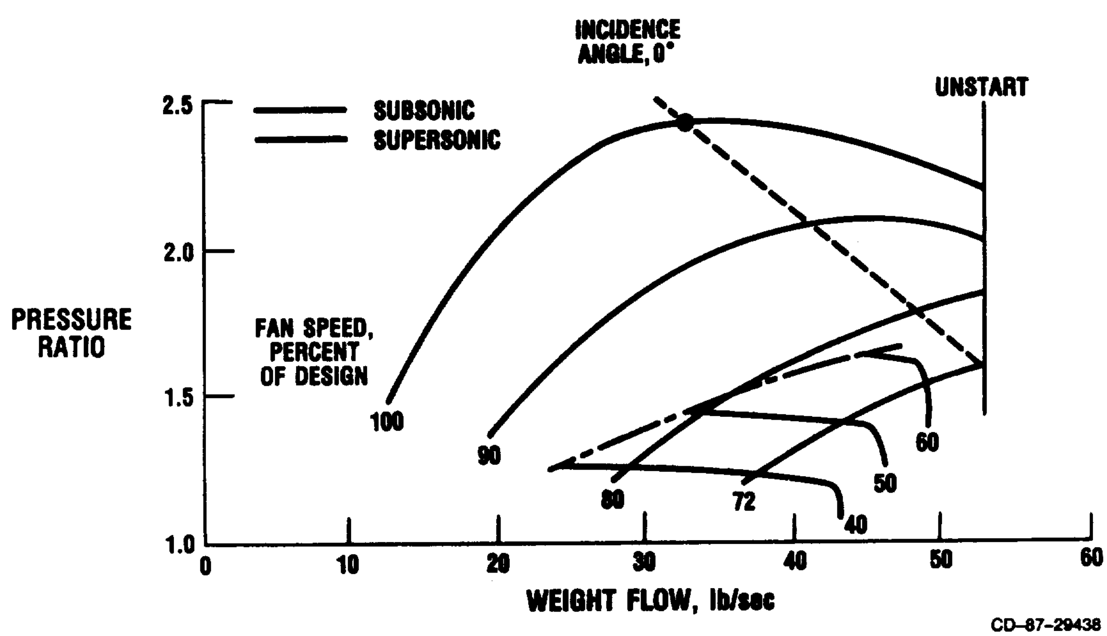


Figure 6. - Predicted fan performance map (pressure ratio versus weight flow).



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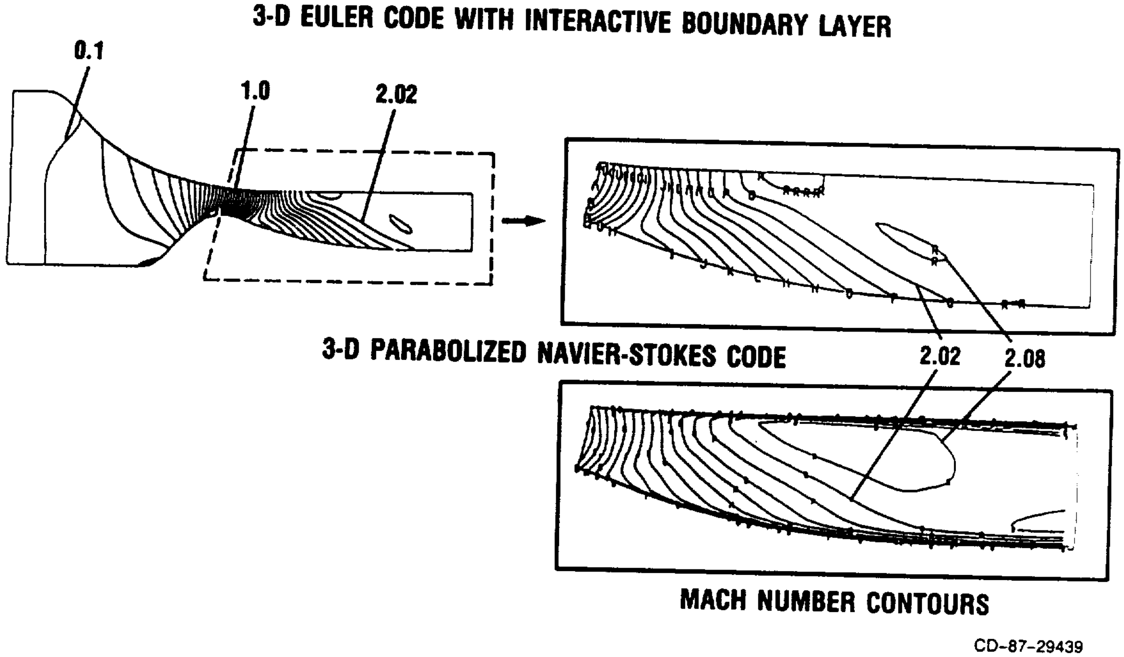


Figure 7. - Facility variable-inlet nozzle.

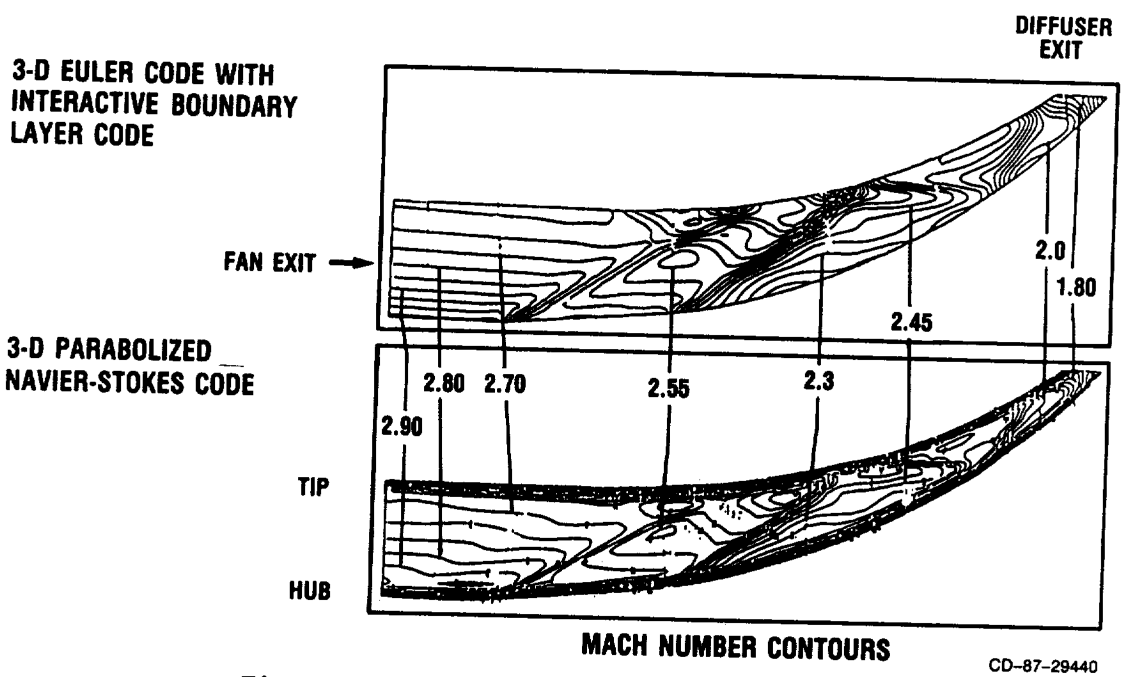


Figure 8. - Facility variable diffuser.

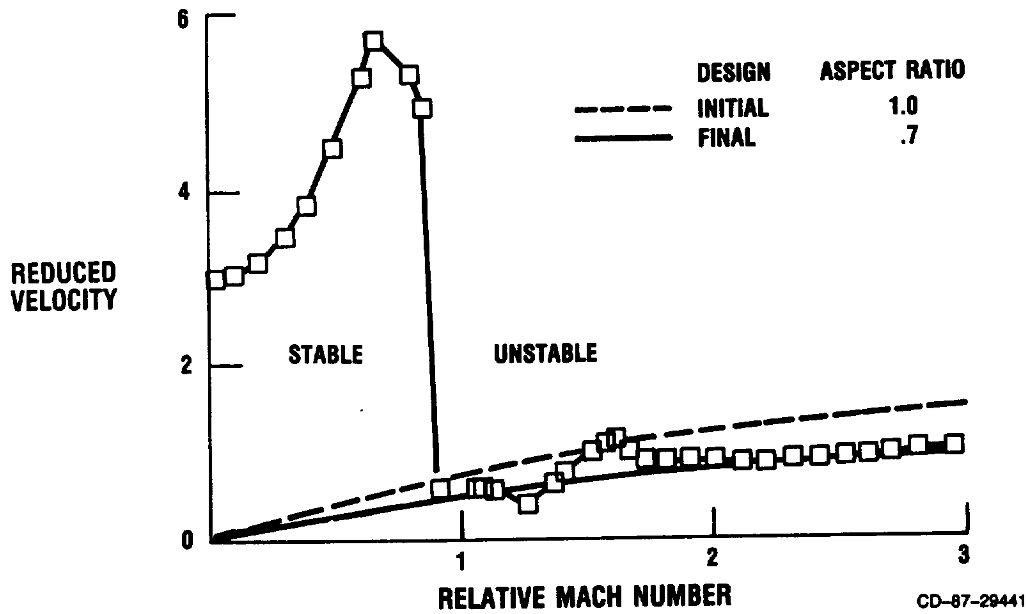


Figure 9. - Aeroelastic analysis of supersonic throughflow fan torsional flutter.

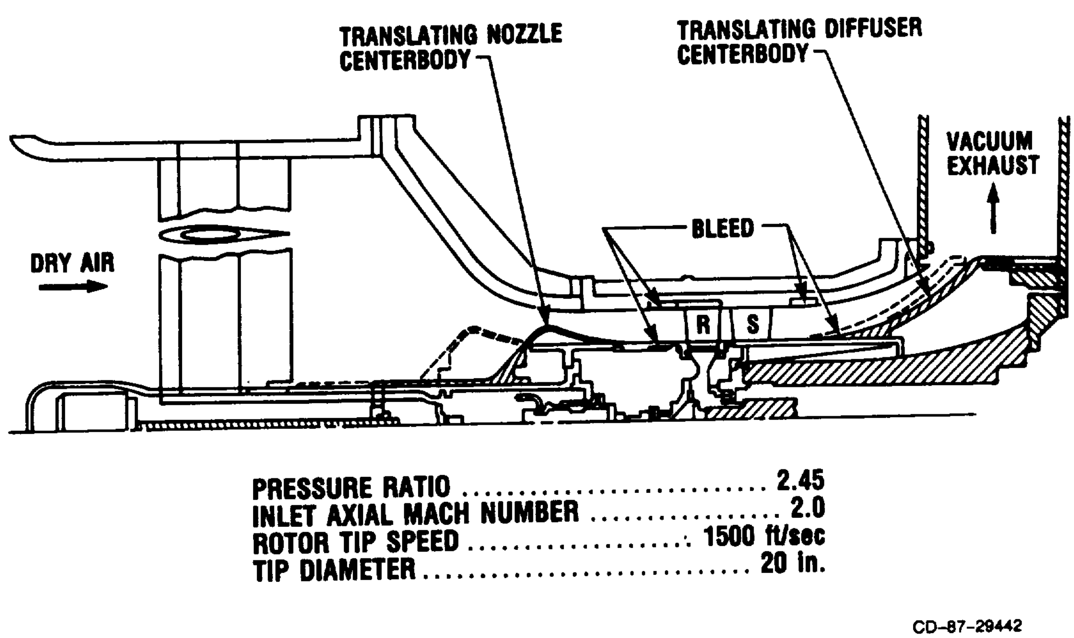


Figure 10. - Test package.

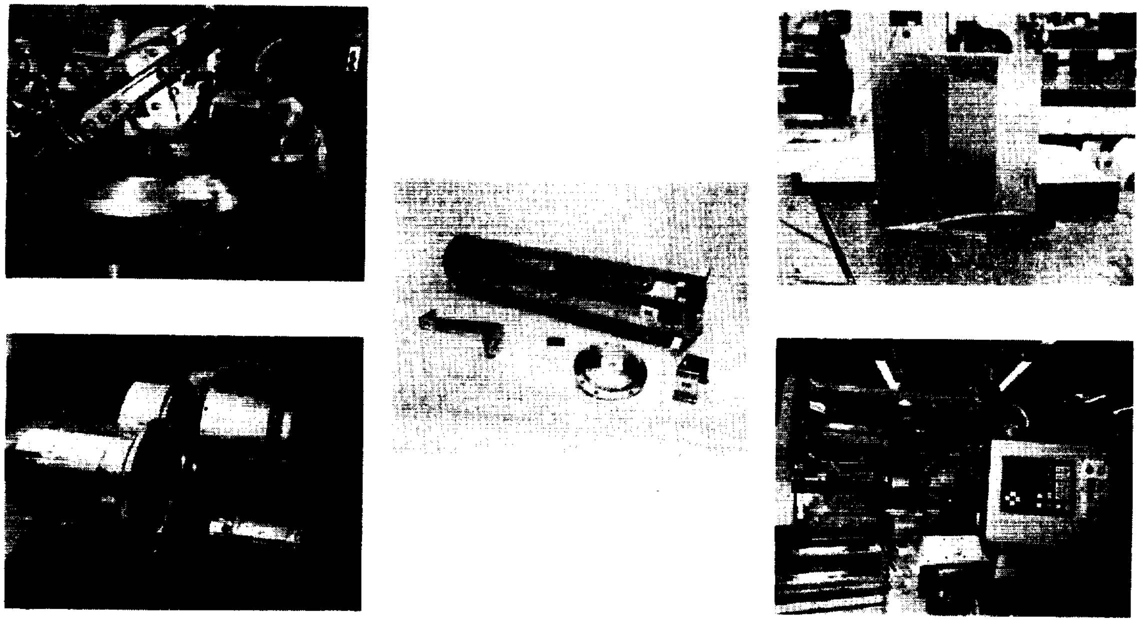

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Figure 11. - Hardware fabrication.

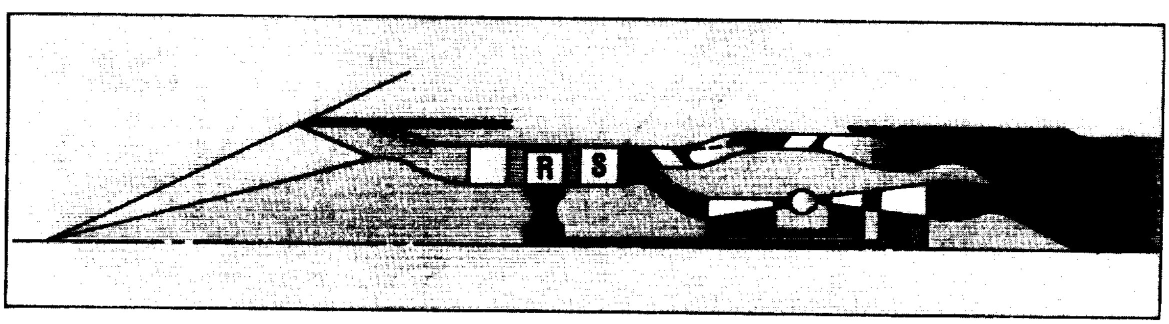


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Figure 12. - Supersonic throughflow turbofan high-speed engine concept (Mach 3 design).

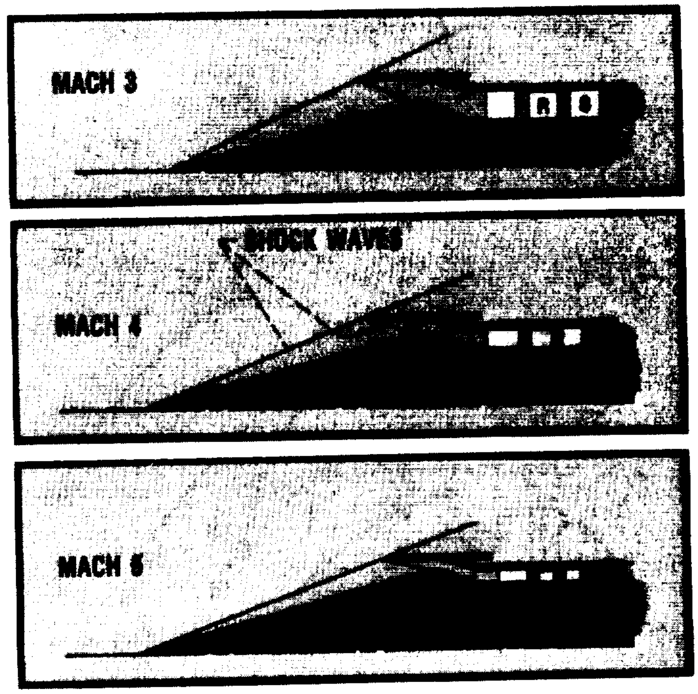


Figure 13. - Various flight Mach number designs for supersonic throughflow turbofan.

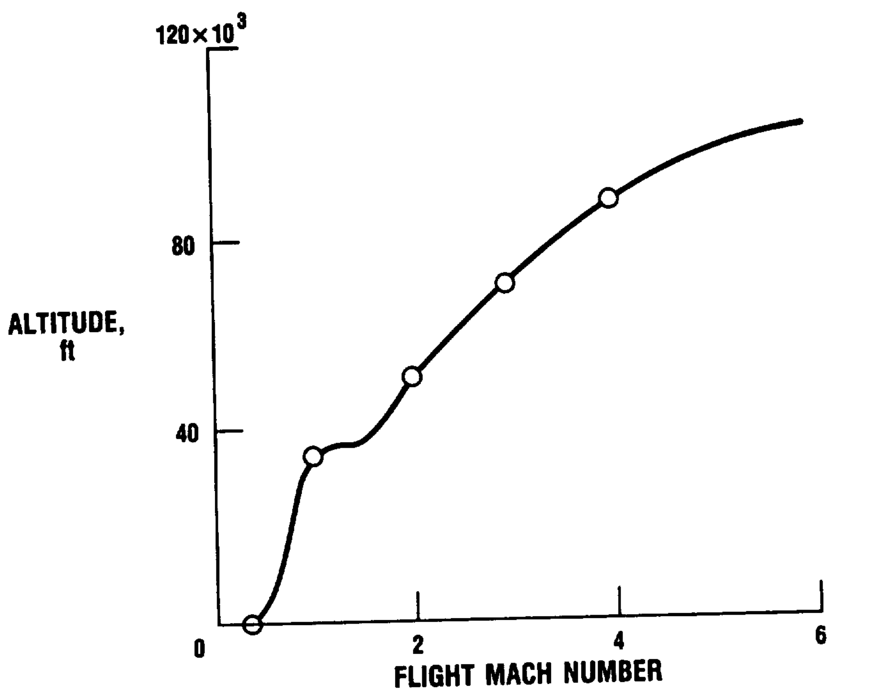


Figure 14. - Assumed high-speed aircraft climb flight path.

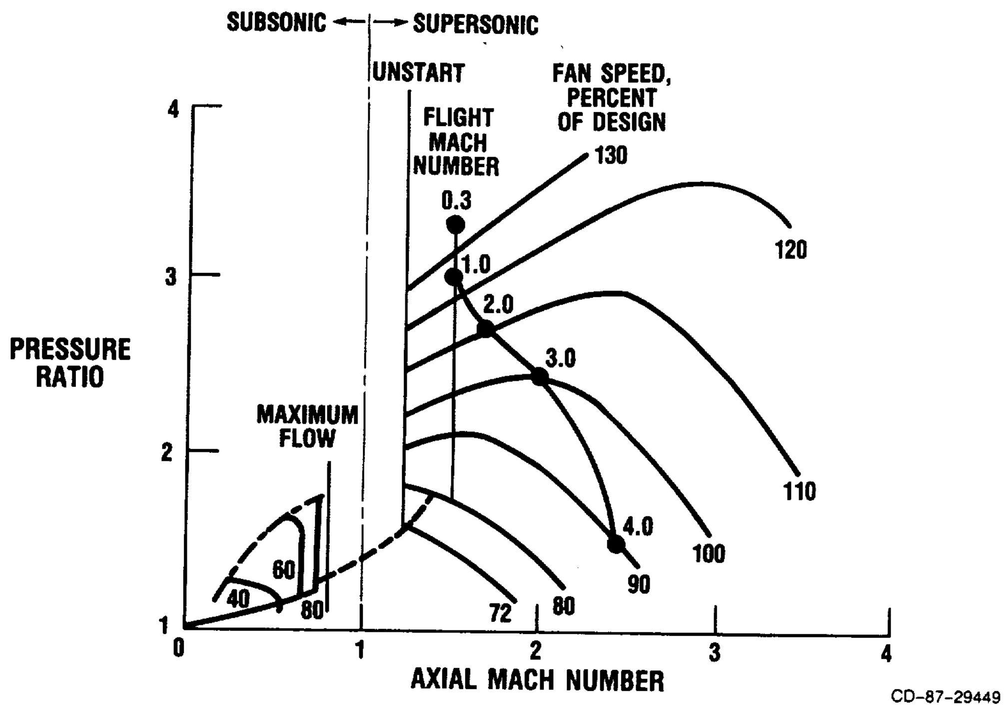


Figure 15. - Supersonic throughflow turbofan fan operating line.

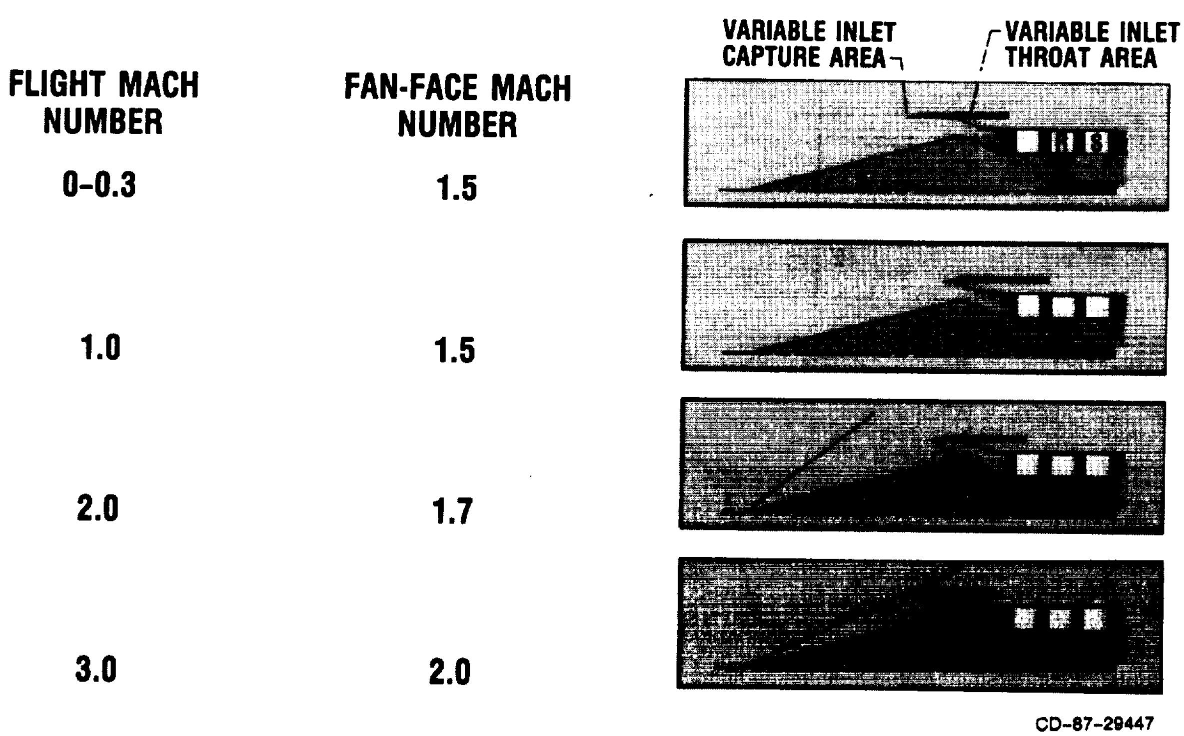


Figure 16. - Inlet geometry configurations for supersonic throughflow turbofan.