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TRANSONIC COMPRESSORS

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THE DESIGN AND DEVELOPMENT OF TRANSONIC MULTISTAGE COMPRESSORS

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TRANSONIC MULTISTAGE COMPRESSORS

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ABSTRACT

The feasibility of operating multistage axial-flow compressors at transonic flow conditions was first demonstrated at the NACA Lewis Flight Propulsion Laboratory (now the NASA Lewis Research Center) during the early to mid fifties. This pioneering research provided a major breakthrough in technology and has contributed significantly to the high performance of present day turbine engines. Following the initial demonstrations, numerous organizations within governments, universities and industries worldwide have contributed to further enhancing the performance of the transonic compressors through their own research. NASA has and continues to play a key role in the development of technology in this critical area.

This paper presents a historical perspective on the development of the transonic multistage compressor. Changing trends in design and performance parameters are noted. These changes are related to advancements in compressor aerodynamics, computational fluid mechanics and other enabling technologies. The parameters normally given to the designer and those that need to be established during the design process are identified. Criteria and procedures used in the selection of these parameters are presented and discussed. Included are discussion on the selection of tip speed, aerodynamic loading, flowpath geometry, incidence and deviation angles, blade/vane geometry, blade/vane solidity, stage reaction, aerodynamic blockage, inlet flow per unit annulus area, stage/overall velocity ratio, and aerodynamic losses. Trends in these parameters both spanwise and axially through the machine are highlighted. The effects of flow mixing and methods for accounting for the mixing in the design process are discussed.

Brief discussions on the present day understanding and modeling capability of unsteady flow phenomena are presented which includes rotating stall, surge flutter and forced response. The influence of these unsteady phenomena on design parameter selections are highlighted.

The impact of the new and more demanding requirements being imposed on the propulsion system to meet advanced aircraft mission needs are noted. Examples of modern day high speed multistage compressors are presented. A case history of the design and development of a NASA Lewis Research Center high tip speed, high pressure ratio core compressor inlet stage group is highlighted.

OUTLINE

First, a historical perspective on the development of the transonic multistage compressor is provided. The impact of various technologies are highlighted. Present day design procedures are discussed. The parameters normally given to the designer and those that need to be established during the design process are identified. Criteria and procedures used in the selection of these parameters are presented. Ranges in the values for these parameters found in modern day transonic compressors are given. The affect of flow mixing and methods for accounting for it in the design process are discussed. Mechanical, aeroelastic and other considerations which must be taken into account during the design process are highlighted. And last, examples of modern day transonic compressors are presented along with lessons learned during their development.

OUTLINE

HISTORICAL PERSPECTIVE

TECHNOLOGY IMPACT

COMPRESSOR DESIGN PROCEDURE

DESIGN PARAMETERS

FLOW MIXING

MECHANICAL/AEROELASTIC CONSIDERATIO9NS

OTHER CONSIDERATIONS

EXAMPLES OF MODERN DAY TRANSONIC COMPRESSORS

HISTORICAL PERSPECTIVE

HISTORICAL PERSPECTIVE

A detailed review of the early research leading to the development of the transonic compressor is given in references 1-3. These papers summarize the transonic compressor research at the NASA Lewis and Langley Laboratories prior to 1960. Extensive reference lists are given with each paper.

As noted by Lieblein and Johnson in (ref. 1), the advent of supersonic flight in the late 1940's required that the turbojet engine be made lighter and more compact. For the compressor, this meant units of smaller diameter and fewer number of stages. Aerodynamically, the requirement was translated into a need for higher weight flow per unit frontal area and higher stage pressure ratio.

Compressor design theories at the time were generally based on the use of symmetrical velocity diagrams (50% reaction) and thick-nose blade profiles of approximately 10% maximum thickness ratio. These design practices were developed primarily from consideration of isolated-airfoil and two-dimensional low-speed concepts. As a result, serious limitations were imposed on the attainable specific weight flow and stage pressure ratio.

First, the use of thick blade profiles required that relative inlet Mach numbers be limited to about 0.70 to 0.75 to avoid serious loss penalties or possible choking of the flow. Second, the use of pre-rotation to maintain the rotor relative inlet mach number within the limiting values or to establish the symmetrical velocity diagrams restricted the maximum values of tip speed and inlet velocity (and consequently specific weight flow).
It was clear from the early research directed towards higher Mach number operation, that in order to achieve satisfactory performance goals, new blade shapes and improved design procedures needed to be developed. Highlights of the major accomplishments achieved prior to 1960 towards meeting these goals are summarized in the first three references. Included in these accomplishments was the development of a new blade loading parameter known as the D-factor (ref. 4), the development of a shock loss model (ref. 5), the demonstration of the double circular arc (DCA) blade shape, and the establishment of a data base involving single stage and multistage experiments which reflected on the desire to increase the reaction (reduce swirl leaving the stationary blade rows) and increase the blade solidity for the transmic stages.

The following figures present early subsonic compressors for comparison to early and modern day transonic multistage compressors to reflect on the progress to date for increasing the overall pressure ratio and reducing the required number of stages.

Shown here is the first axial-flow compressor used in a turbojet engine. It is a German engine. The compressor had eight stages. It was designed for subsonic flow in all blade rows. The design pressure ratio and tip speed are not known.
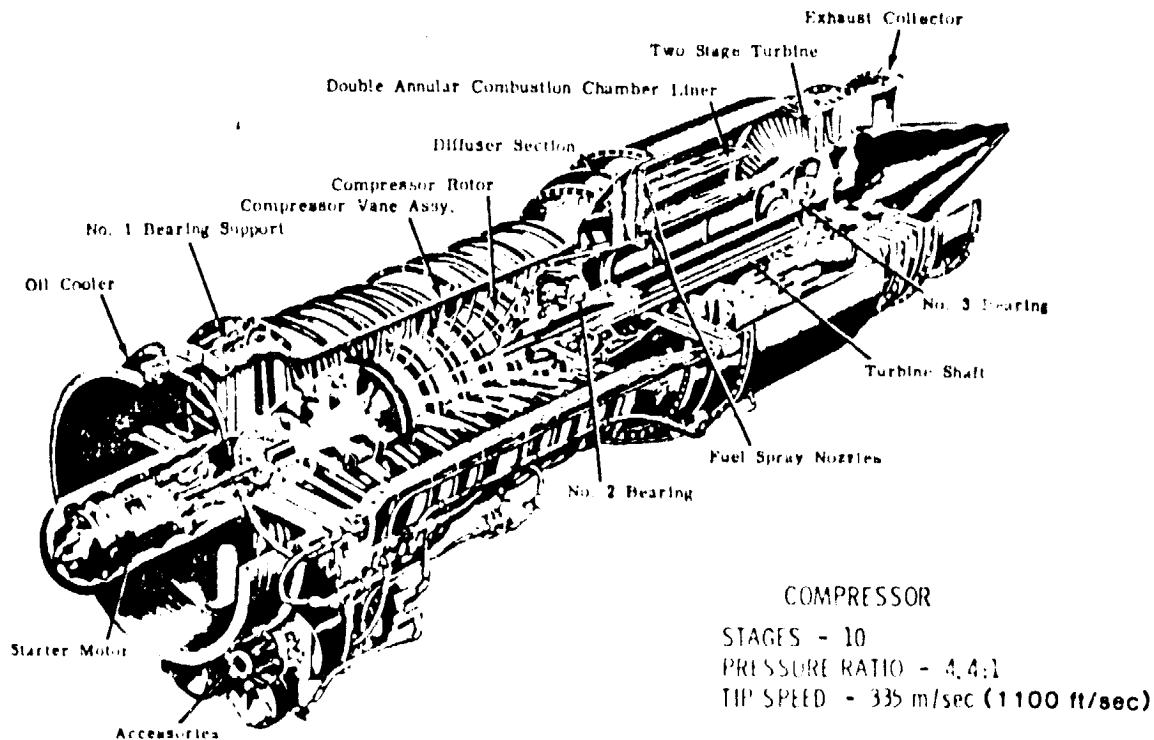
FIRST TURBOJET USING AXIAL-FLOW COMPRESSOR
GERMAN JUMO 004
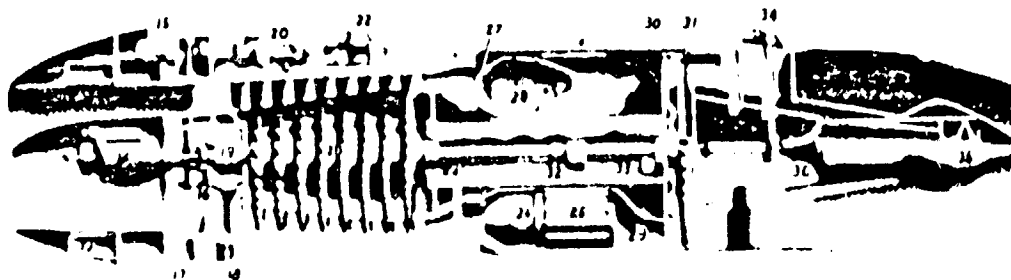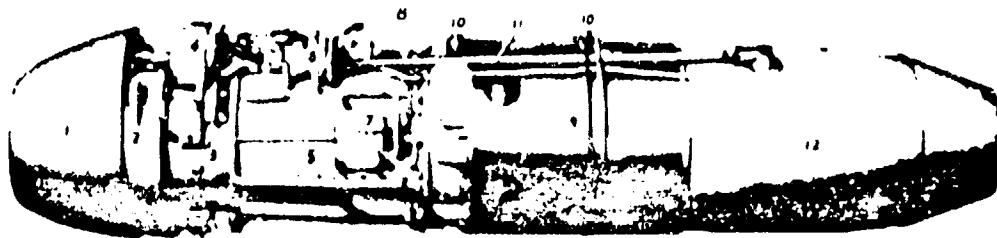
EARLY AMERICAN AXIAL TYPE TURBOJET

Shown here is an early American turbojet with a ten stage subsonic axial-flow compressor. This compressor was designed to produced a pressure ratio of 4.4 at a tip speed of 335 m/sec (1099 ft/sec). At this speed inlet guide vanes were needed to impart a swirl in the direction of rotation to keep the relative velocity to the rotor subsonic. It was designed by Westinghouse.
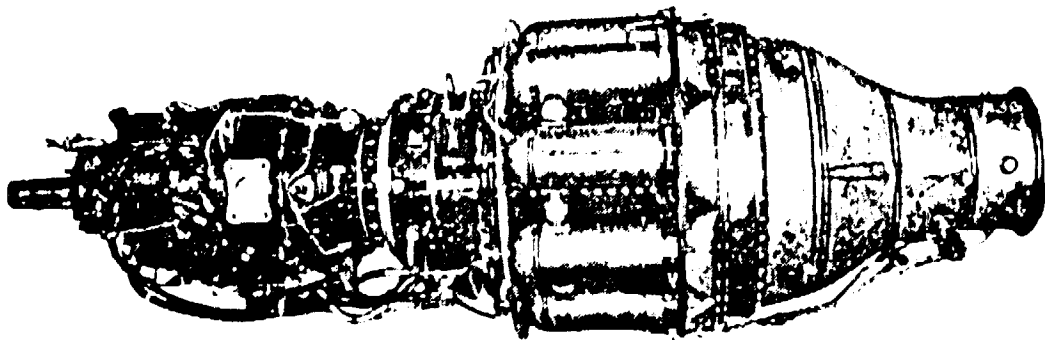
EARLY AMERICAN AXIAL TYPE TURBOJECT
WESTINGHOUSE J34

Exhaust Collector

Two Stage Turbine

Double Annular Combustion Chamber Liner

Diffuser Section

Compressor Rotor

Compressor Vane Assy.

No. 1 Bearing Support

Oil Cooler

No. 3 Bearing

Turbine Shaft

Fuel Spray Nozzles

No. 2 Bearing

Starter Motor

Accessories

COMPRESSOR

STAGES - 10
PRESSURE RATIO - 4.4:1
TIP SPEED - 335 m/sec (1100 ft/sec)

7

FIRST AMERICAN TURBOPROP

Shown here is the first American turboprop engine. It was designed and built
by General Electric. The burner cans are wrapped around the compressor as can
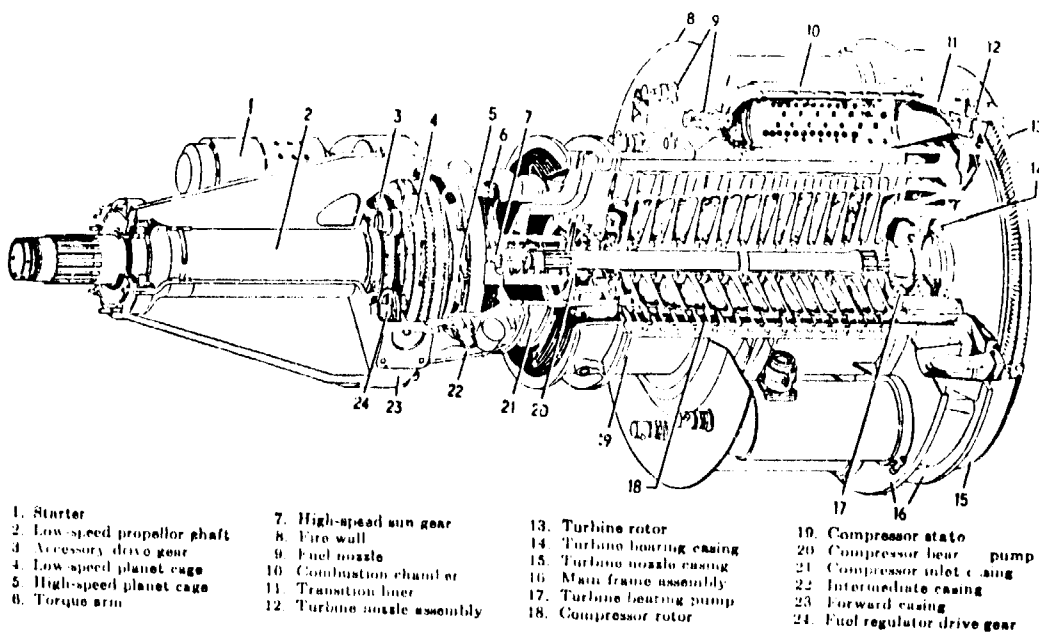be seen in the next figure.

FIRST AMERICAN TURBOPROP
GENERAL ELECTRIC TG-100

Shown here is a cross-section of the TG-100 turboprop engine. One can note that in the early engine, the compressor component component tended to dominate much of the engine volume and provided the impetus to reduce the compressor diameter and length. The next figure shows a photograph of the compressor and gives its design parameters.

## CROSS-SECTION OF GENERAL ELECTRIC TG-100 TURBOPROP

1. Starter	7. High-speed sun gear	13. Turbine rotor	19. Compressor stato
2. Low-speed propellor shaft	8. Fire wall	14. Turbine bearing casing	20. Compressor bear pump
3. Accessory drive gear	9. Fuel nozzle	15. Turbine nozzle casing	21. Compressor inlet casing
4. Low-speed planet cage	10. Combustion chamber	16. Main frame assembly	22. Intermediate casing
5. High-speed planet cage	11. Transition liner	17. Turbine bearing pump	23. Forward casing
6. Torque arm	12. Turbine nozzle assembly	18. Compressor rotor	24. Fuel regulator drive gear

9

COMPRESSOR ROTOR FOR GENERAL ELECTRIC TG-100

The TG-100 Compressor had 14 stages and produced a pressure ratio of 5.1 at at tip speed of 285 m/sec (935 ft/sec). As reflected in the design tip speed all compressor blade rows operated subsonically.

COMPRESSOR ROTOR FOR GENERAL ELECTRIC TG-100
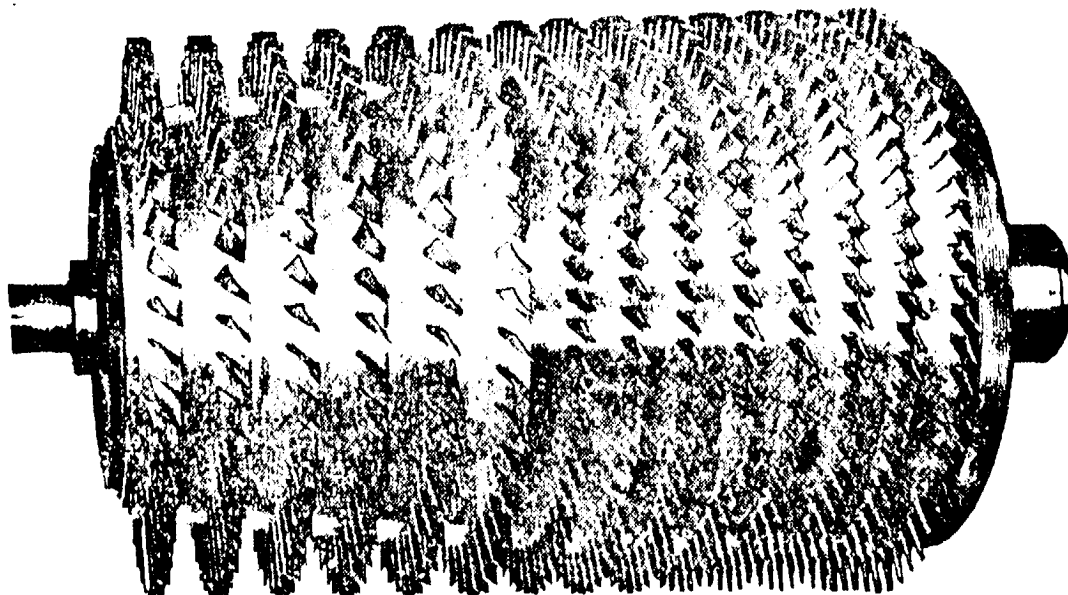
STAGES - 14
PRESSURE RATIO - 5.1:1
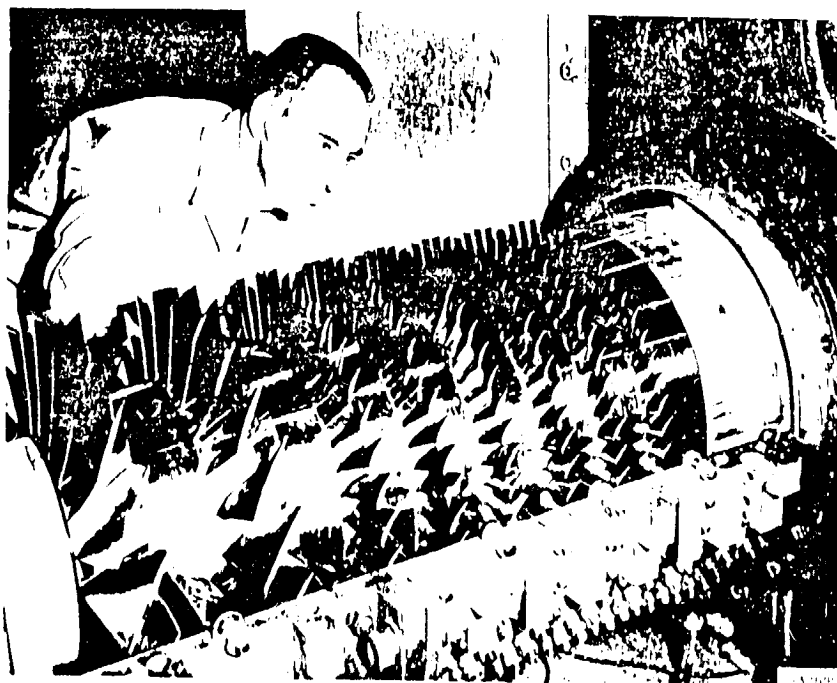TIP SPEED - 285 m/sec (935 ft/sec)

NACA EIGHT-STAGE AXIAL-FLOW TRANSONIC COMPRESSOR

Shown here is the NACA eight-stage axial-flow transonic compressor. It was designed for a pressure ratio of 10.3 and an inlet tip speed of 356 m/sec (1168 ft/sec) (ref. 6). It was this compressor that first demonstrated the feasibility of operating multistage axial-flow compressors at transonic conditions. The work was conducted at the NACA Lewis Flight Propulsion Laboratory (now NASA Lewis Research Center) during the early to mid fifties. This pioneering research provided a major breakthrough in technology for multistage compressor design and performance, and has contributed significantly to the high performance of present day turbine engines.

Following the initial demonstration, numerous organizations within governments, universities and industries worldwide have contributed to further enhancing the performance of the transonic compressors through their own research. NASA has and continues to play a key role in the development of technology in this critical area.

Only the first two stages of the NACA eight stage compressor operated with transonic flow. The last six stages operated subsonically. This was achieved through employing high reaction (non-symmetrical velocity diagrams) in the front two stages and 50% reaction (symmetrical diagrams) in the rear six stages.

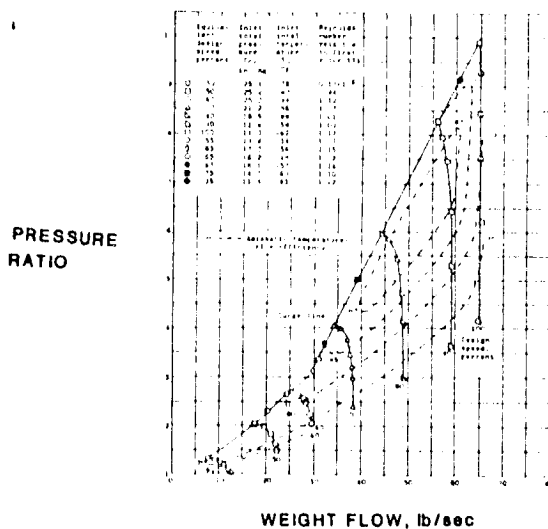## NACA EIGHT—STAGE AXIAL FLOW TRANSONIC COMPRESSOR

STAGES - 8
PRESSURE RATIO - 10.3:1
TIP SPEED - 356 m/sec (1168 ft/sec)

NACA EIGHT-STAGE AXIAL-FLOW TRANSONIC COMPRESSOR
OVERALL PERFORMANCE MAP

This figure shows the overall performance achieved with the NACA eight-stage
compressor. One can note that the demonstrated pressure ratio was shy of the
design value of 10.3 and the efficiency was on the order of 80% at maximum
pressure ratio. However, the part speed performance was excellent. The lower
than design pressure ratio at design speed was attributed to too large of
boundary layer blockage assumed in the design process for the rear stages
causing the rear stages to stall prematurely. Extensive research was
conducted on this compressor (ref.7-12). It was this extensive data base that
resulted in numerous transonic compressor designs for aircraft applications.

NACA EIGHT-STAGE AXIAL FLOW TRANSONIC COMPRESSOR
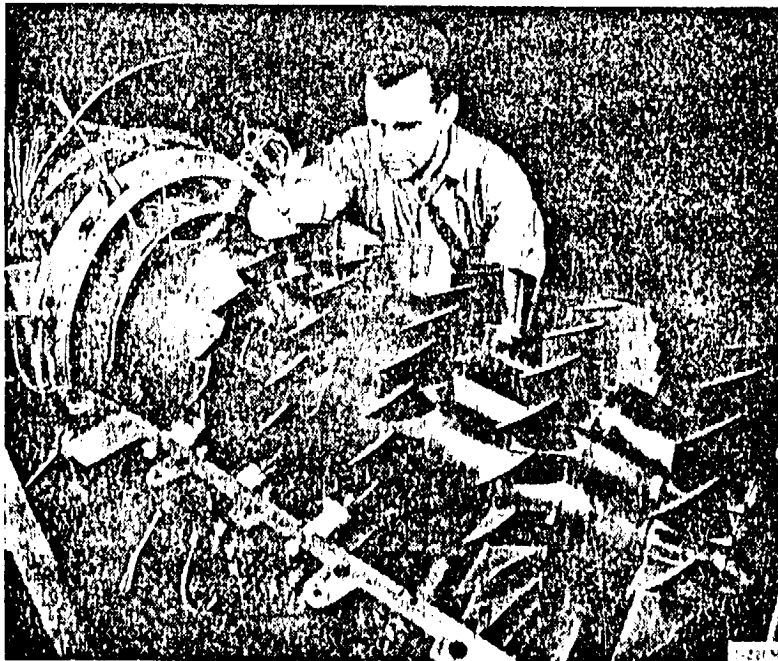
OVERALL PERFORMANCE MAP

PRESSURE
RATIO

WEIGHT FLOW, lb/sec

NACA FIVE-STAGE AXIAL-FLOW TRANSONIC COMPRESSOR

Shown here is a photograph of the NACA five-stage axial-flow transonic compressor. It was designed for a pressure ratio of 5 and its design tip speed was 335 m/sec (1100 ft/sec) (ref. 13). This design employed high reaction stages throughout. It did not employ an inlet-guide-vane and all stators were designed to turn the flow back to the axial direction. The design of this machine followed shortly the design of the NACA eight stage compressor.

## NACA FIVE—STAGE AXIAL FLOW TRANSONIC COMPRESSOR

STAGES – 5
PRESSURE RATIO – 5.0:1
TIP SPEED – 335 m/sec (1100 ft/sec)

13

NACA FIVE-STAGE AXIAL FLOW TRANSONIC COMPRESSOR OVERALL PERFORMANCE MAP

The performance of the NACA five-stage compressor is shown here. A peak pressure ratio of 5.0 was attained at design speed at an efficiency of approximately 0.81. Peak efficiency on the map was 0.87. The performance achieved, though lacking at design speed, did demonstrate the feasibility of achieving satisfactory performance with all rotor blades operating transonically. Like the NACA eight stage compressor, the part speed performance was excellent. The performance data obtained on this machine was reported in references 14-17.

NACA FIVE – STAGE AXIAL FLOW TRANSONIC COMPRESSOR

OVERALL PERFORMANCE MAP

WEIGHT FLOW, lb·sec

14
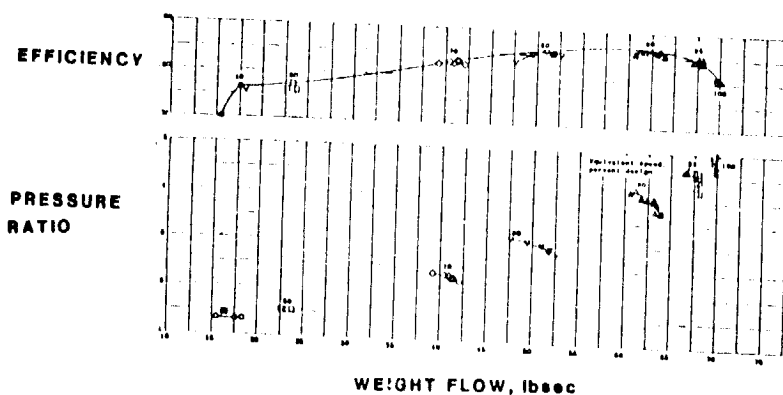
NASA ENERGY EFFICIENCY ENGINE CONFIGURATION
PRATT & WHITNEY

The Energy Efficient Engine (E[3]) program was initiated in the late 1970's and extended over a period of approximately five years. The objective was to demonstrate significant reductions in fuel consumption for advanced high bypass ratio turbofan engines. Two contracts were awarded by NASA, one to Pratt & Whitney and one to General Electric, to: (1) design, build and test the components separately and (2) to test an integrated core low spool (ICLS). Shown here is a cross-section of the Pratt & Whitney engine with the compressor overall design values. The high compressor for this engine was designed for ten stages to produce a pressure ratio of 14 at a tip speed of 379 m/sec (1243 ft/sec) (ref. 18).

NASA
C-79-2743

## NASA ENERGY EFFICIENT ENGINE CONFIGURATION
PRATT & WHITNEY
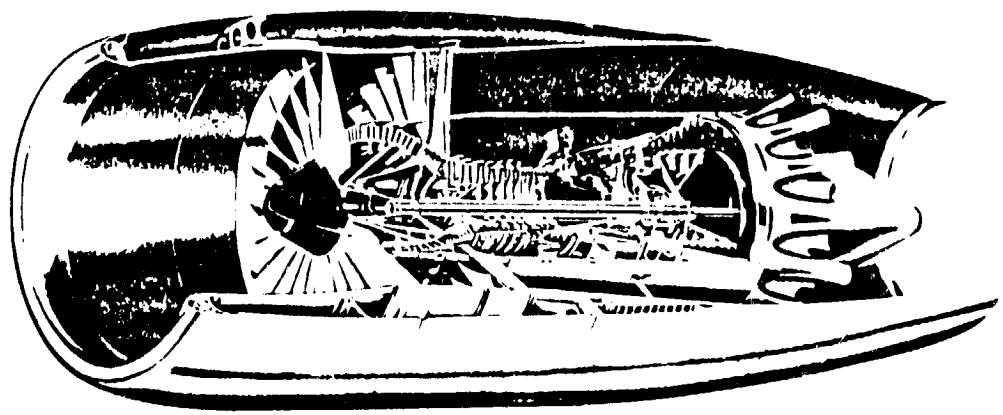
H. P. COMPRESSOR

STAGES - 10
PRESSURE RATIO - 14:1
TIP SPEED - 379 m/sec (1243 ft/sec)

15

NASA ENERGY EFFICIENT ENGINE CONFIGURATION
GENERAL ELECTRIC

Shown here is the cross-section of the General Electric engine. The high compressor for this engine was designed to achieve a pressure ratio of 23 in 10 stages at a design inlet tip speed of 456 m/sec (1495 ft/sec)(ref. 19). The Energy Efficient Engine compressors reflect modern day technology and a very significant advance over the first transonic compressors, in terms of overall pressure ratios, tip speeds, and in number of stages required to achieve a given pressure ratio.

NASA
C-79-2745

NASA ENERGY EFFICIENT ENGINE CONFIGURATION
### GENERAL ELECTRIC
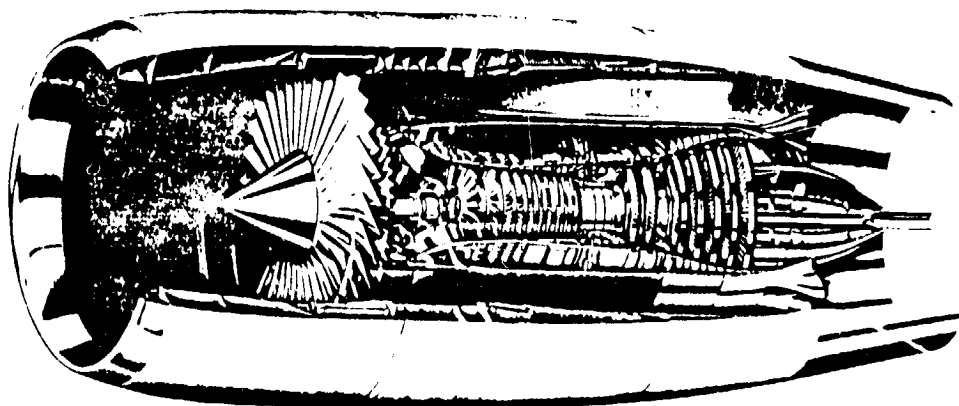
H. P. COMPRESSOR

STAGES - 10
PRESSURE RATIO - 23:1
TIP SPEED - 456 m/sec (1495 ft/sec)

16

NASA RESEARCH COMPRESSOR

Shown here is a NASA research compressor. It represents a modern day inlet stage group. It is designed to produce a pressure ratio of 4.45 in three stages at an inlet tip speed of 427 m/sec (1400 ft/sec) (ref. 20). To put the technology of this machine in perspective to other multistage compressors it would produce a pressure ratio of 9.27 in five stages and a pressure ratio of about 20 in eight stages maintaining stage loading levels.

The performance of the Energy Efficient Engine compressors and the NASA Research Compressor will be discussed in the section titled "Examples of Modern Day Transonic Compressors". Also, details about the design and development of these machines are given in the latter section.

NASA
C-79-3791

NASA RESEARCH COMPRESSOR

WEIGHT FLOW - 29.7 kg/sec
PRESSURE RATIO - 4.45:1
TIP SPEED - 427 m/sec (1400 ft/sec)

17
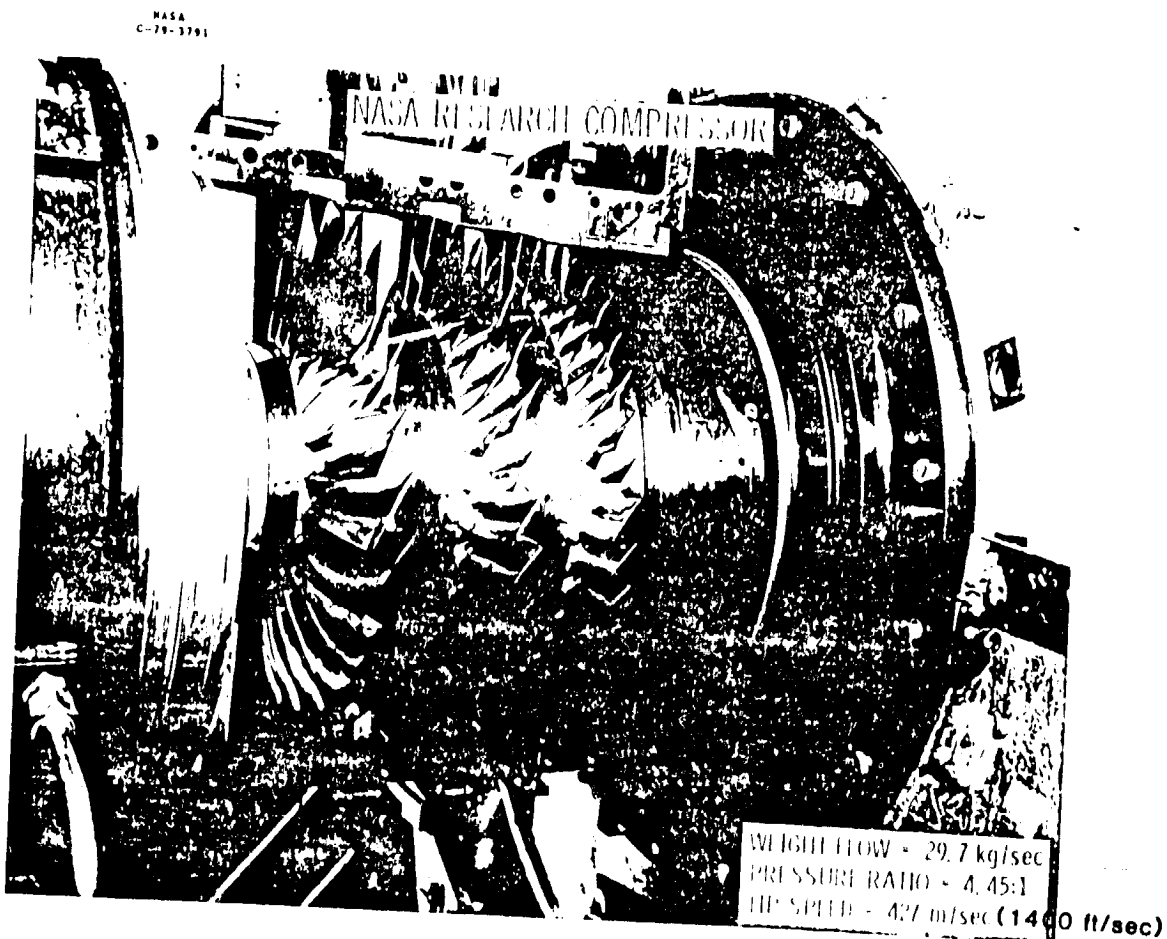
TECHNOLOGY IMPACT

TECHNOLOGY IMPACT

In each of the areas listed below, examples of the major impacts that have resulted in the present level of technology of transonic multistage compressors will be noted.

TECHNOLOGY IMPACT

- AERODYNAMIC

- COMPUTATIONAL FLUID MECHANICS

- INSTRUMENTATION AND CONTROLS

- STRUCTURAL MECHANICS

- MATERIALS

- MECHANICAL COMPONENTS

AERODYNAMICS

In the area of aerodynamics, advanced blade shapes is high on the list of items having a very significant impact on performance of modern day transonic compressors. The range of blade shapes presently being used are listed. The airfoil series type blade shapes are still used for blade rows operating with subsonic flows. The DCA blade shape is used for high subsonic to low supersonic Mach numbers. The MCA blade shape is used extensively at the higher supersonic relative Mach numbers. The polynomial shape (ref. 21) allows more flexibility than the MCA shape and in fact can duplicate the DCA and MCA shapes through proper selection of the polynomial coefficient. The polynomial shape can also be used to approximate the various airfoil series as well as the controlled diffusion (CD) shapes. The CD shape is being employed with a high degree of success for moderate to high subsonic blade elements (ref. 22-28). The words "controlled diffusion" were coined to reflect a class of blade shapes that employ the concept of shaping the blade behind the point of peak suction surface velocity such that the diffusion rate and associated suction surface boundary layer results in minimum loss for the airfoil section. The supercritical blade shape referred to in some of the references is considered a special class of the CD shape where, in addition to controlling the diffusion rate behind the location of peak suction surface velocity, the forward portion of the blade is shaped to permit supercritical operation (a shock free supersonic pocket along the suction surface) at the higher subsonic Mach numbers. The benefit of CD blading has come not only in lower losses but also improved range.

In addition to blade shapes, the extensive data base that has been established to guide the selection of the design parameters must also rate high on the impact scale. Without it, the level of technology for the transonic compressor would not be what it is today.

AERODYNAMIC

O BLADE SHAPES

- AIRFOIL SERIES

- DOUBLE CIRCULAR ARC (DCA)

- MULTIPLE CIRCULAR ARC (MCA)

- POLYNOMINAL (ARBITRARY)

- CONTROLLED DIFFUSION (CD)

O EMPIRICAL DATA BASE

The present design system, centered around the axisymmetric through-flow design code, still remains highly empirical in nature. Extensions of the empirical data base which supports the axisymmetric through-flow design codes have contributed the most to the present state of technology of transonic multistage compressors. However, this is rapidly changing. The computational methods, at the present time primarily used in an analysis mode, are now impacting the selection of design parameters in the design process. Potential flow and Euler codes, now being used routinely as analysis tools, are making their impact on the level of technology being demonstrated in the most recently designed machines. The application of viscous codes have been somewhat limited. However, they are now being employed in the design of the controlled diffusion (CD) blading.

COMPUTATIONAL FLUID MECHANICS

- AXISYMMETRIC DESIGN CODE

- POTENTIAL FLOW ANALYSIS CODES

- EULER ANALYSIS CODES

- VISCOUS ANALYSIS CODES

FANS AND COMPRESSORS CONTROLLED DIFFUSION BLADES

The controlled diffusion blade shape is shown here, along with the blade
surface Mach number distribution. A supersonic shock free pocket (super-
critical flow region) exists along the blade suction surface. The flow range
of the CD blade shape is compared to the conventional blade shape in the lower
right part of the figure. The design of the CD blade shape requires extensive
use of the internal flow codes. The benefits of this shape are indicated in
the lower left corner.

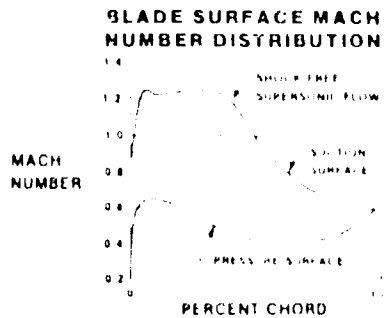NASA
C-82-5489

Lewis Research Center   NASA

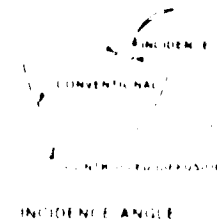# FANS AND COMPRESSORS
CONTROLLED-DIFFUSION BLADES

**BLADE SURFACE MACH
NUMBER DISTRIBUTION**

PRESSURE SURFACE   SUCTION SURFACE

MACH NUMBER

SHOCK FREE SUPERSONIC FLOW
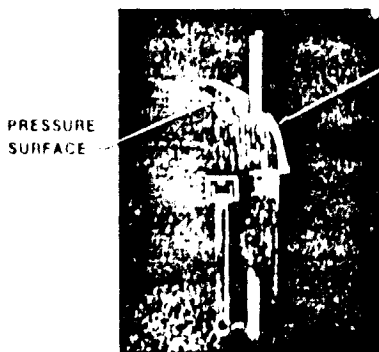
SUCTION SURFACE

PRESSURE SURFACE

PERCENT CHORD

FLOW RANGE

- DESIGNED UTILIZING INTERNAL FLOW CODES
- RUGGED EROSION RESISTANT BLADES
- SHOCK FREE FLOW DESIGN
- IMPROVED FLOW RANGE

LOSS COEFFICIENT

INCIDENCE

CONVENTIONAL

CONTROLLED DIFFUSION

INCIDENCE ANGLE

The non-intrusive laser anemometry instrumentation has had a very significant technological impact in that the flow fields of transonic machines have now been well documented (ref. 29-32). Improved models have been developed and computational methods have been verified. The design of higher Mach number blading is now being guided by the verified Euler codes.

Also, the digital electronic controls have permitted more precise control of variable geometry for maximum performance over the speed range.

INSTRUMENTATION AND CONTROLS
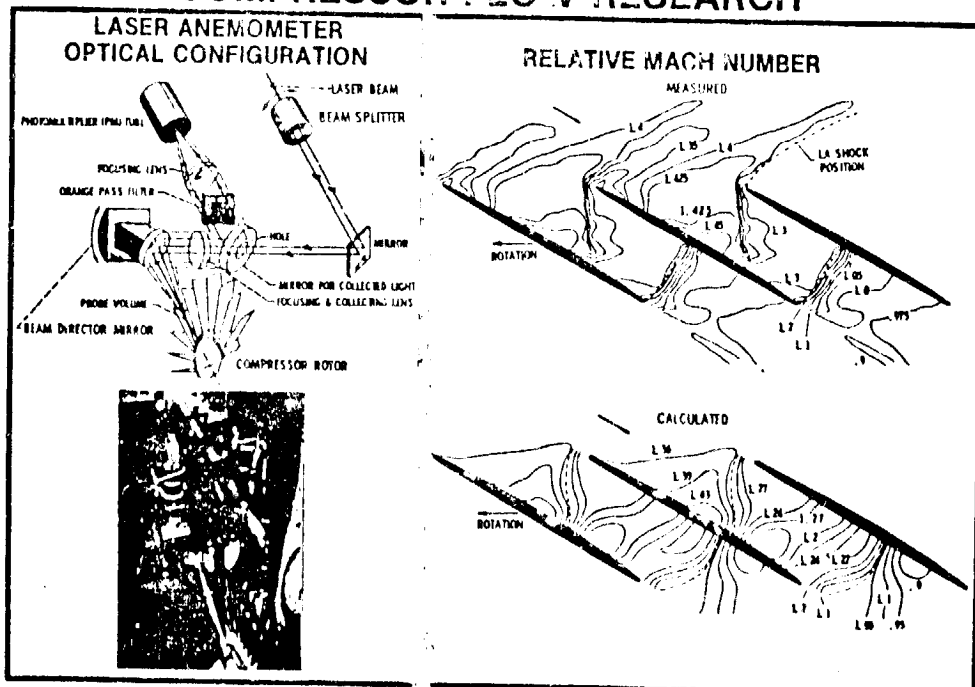
NONINTRUSIVE LASER ANEMOMETRY

HOLOGRAPHY

DIGITIAL ELECTRONIC CONTROLS

ADVANCED METHODS FOR COMPRESSOR FLOW RESEARCH

Shown here is a laser system used at NASA Lewis along with a comparison between measured and calculated Mach number contours. The three dimensionality of the shock structure has been well documented (ref. 31). Even the unsteadiness has been assessed (ref. 32).

NASA
C-81-3305

ADVANCED METHODS FOR COMPRESSOR FLOW RESEARCH

LASER ANEMOMETER OPTICAL CONFIGURATION

RELATIVE MACH NUMBER

MEASURED

CALCULATED

CD-81-12604

24

As the tip speeds have increased the need for more reliable structural analysis codes were needed. The NASA Structural Analysis Code (NASTRAN) is now used extensively to not only calculate untwist and blade natural frequencies, but also to calculated uncamber. The blade coordinates are being adjusted to account for the uncamber in the design process. Significant improvements in the aeroelastic analysis codes have also been made. However, adequate modeling of stall flutter and of forced responses remains inadequate.

STRUCTURAL MECHANICS

NASA STRUCTURAL ANALYSIS CODE (NASTRAN)

AEROELASTIC CODES

MATERIALS

Without the advancement in high temperature high strength steels and the high strength lightweight titanium alloys desired high blade speeds could not be achieved. However, the rim and blade stresses can still impose limits on the aerodynamic design. This is particularly true for highly supersonic aircraft where the incoming air is at elevated temperatures.

MATERIALS

HIGH TEMPERATURE – HIGH STRENGTH STEELS

TITANIUM ALLOYS

Improvements in the technology of bearings and seals has also permitted going to the higher rotative speeds. Without these improvements transonic operation would be limited.

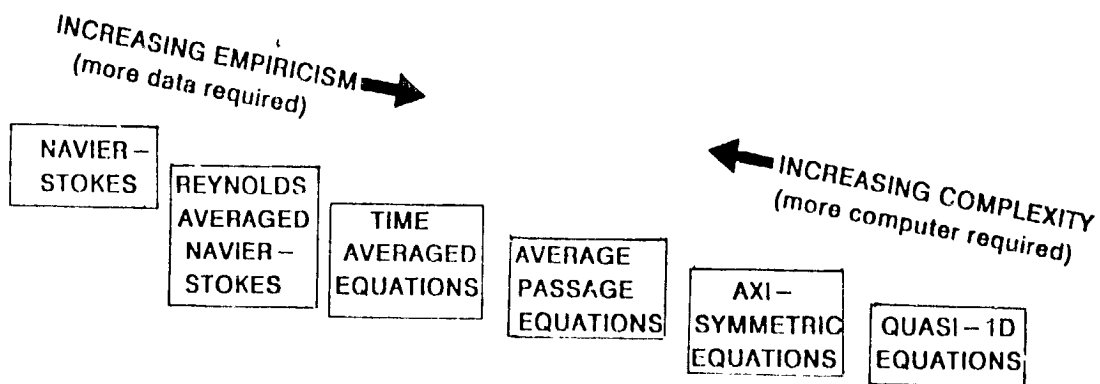# MECHANICAL COMPONENTS

BEARINGS

SEALS

COMPRESSOR DESIGN PROCEDURE

The hierarchy of design and analysis codes are shown here. Moving from right to left results in increased complexity and more computer power required. Moving from left to right results in increasing empiricism and more extensive data bases. The present systems in general have as their basis the axisymmetric through-flow equations. A model for closing the inviscid form of the average-passage equation system is given in reference 33 and would bring the compressor design system a step closer to a fully viscous design system.

DESIGN/ANALYSIS CODE HIERARCHY

INCREASING EMPIRICISM
(more data required) ➡

⬅ INCREASING COMPLEXITY
(more computer required)

| NAVIER-STOKES | REYNOLDS AVERAGED NAVIER-STOKES | TIME AVERAGED EQUATIONS | AVERAGE PASSAGE EQUATIONS | AXI-SYMMETRIC EQUATIONS | QUASI-1D EQUATIONS |

COMPRESSOR DESIGN PROCEDURE

As noted in the discussion of the previous figure, present design procedures tend to remain centered around the axisymmetric through-flow design type codes to provide the initial geometry and performance estimates. These codes require empirical input to establish such parameters as minimum loss incidence angle, flow deviation angle, viscous and shock losses and thus efficiency, and flow blockage allowances to name a few. An example of a axisymmetric design codes is given in reference 21. The output from the axisymmetric code is input into the analysis codes as indicated in the block diagram reflecting the overall design procedure. Iterations are made between the two types of codes until a satisfactory aerodynamic design is established. The analysis codes range from two dimensional potential flow (ref. 34 and 35) and Euler codes (ref. 36) to three dimensional viscous codes (ref. 37) as appropriate and available. With convergence on the aerodynamic design iteration, iterations with the structural analysis codes are pursued along with the aeroelastic codes. With an acceptable overall design, input is provided to blade coordinate and off-design analysis codes.

As the technology advances in computational fluid mechanics and computers, the design system will move from the highly empirical system to a first principle system. Recent years have shown significant advancements towards meeting this goal.

COMPRESSOR DESIGN PROCEDURE
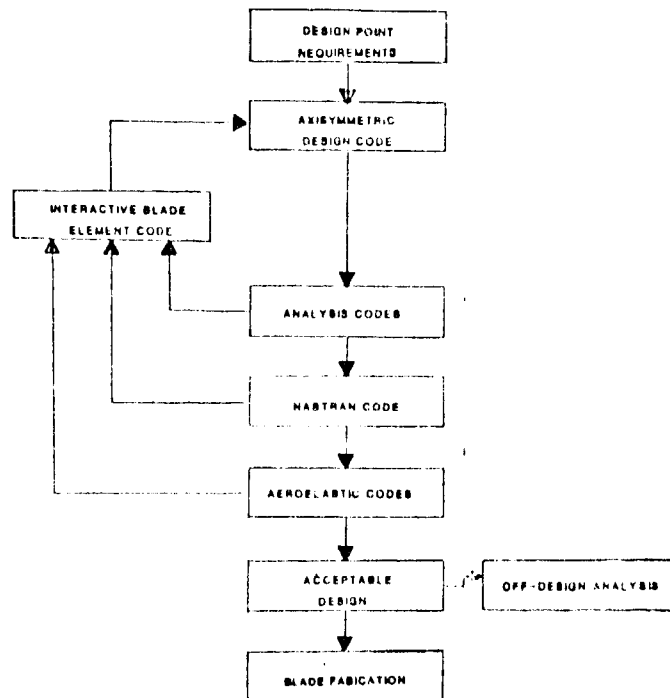
DESIGN PARAMETERS

DESIGN PARAMETERS

Parameters normally given to the designer and those select during the design process are listed below. Many factors influence the values selected for these parameters such as engine type, flow size, application/mission, and the inter dependency between the various parameters for achieving satisfactory performance.Therefore it is difficult to reflect on typical ranges for the design parameters of modern day transonic compressors. However, it is felt some generalized comments may be helpful for some of the parameters, but caution is given in how they might be used in executing a design.

Overall pressure ratios in a single spool range from 14 to more than 20. Tip speeds range up to 457 m/sec (1500 ft/sec). The axial velocity is normally decreased somewhat through the machine to meet the low Mach number required at the burner inlet. Inlet hub/tip ratios go as low as .5. The flowpath is typically between a constant mean diameter and a constant tip diameter. The various blade/vane parameters are listed in the next figure.

DESIGN PARAMETERS

- GIVEN
  - WEIGHT FLOW
  - PRESSURE RATIO
  - ROTATIVE SPEED

- SELECTED DURING DESIGN PROCESS
  - TIP SPEED
  - WEIGHT FLOW PER UNIT ANNULUS AREA
  - VELOCITY RATIO
  - AERODYNAMIC LOADING (NUMBER OF STAGES)
  - INLET HUB/TIP RATIO
  - FLOWPATH GEOMETRY
  - BLADE/VANE GEOMETRY

BLADE/VANE GEOMETRY

Listed here are parameters which are directly related to the blade/vane geometry along with those parameters which influence the blade/vane geometry. Again, caution is given on the use of the generalized comments on the typical ranges for these parameters.

The blade/vane shapes used range from the airfoil series designated shapes to DCA for low transonic conditions and to MCA or tailored airfoils for blade elements operating above 1.2 relative inlet Mach numbers. In the highly transonic front stages the rotor tip solidity ranges from 1.2 to 1.5 and is reduced to about 1.0 in the subsonic rear stages. Aspect ratio for the rotors range from 2 to less than one in the rear stages. Stage reaction runs from moderate to high. Flow blockages allowances range from on the order of 4% in the front to 10 to 15% in the rear stages.

More insight into typical values for the design parameters can be gained by studying the example compressors presented at the end of this lecture.

Reference 38, even though it was initially published during 1956 and revised in 1965, remains an excellent reference today on the design of axial-flow compressors. The empirical data base has changed with technological advancement, but the basic design methods and considerations remain very much the same.

BLADE/VANE GEOMETRY

AIRFOIL SHAPE

SOLIDITY

ASPECT RATIO (CHORD)

THICKNESS DISTRIBUTION

STAGE REACTION

FLOW BLOCKAGE
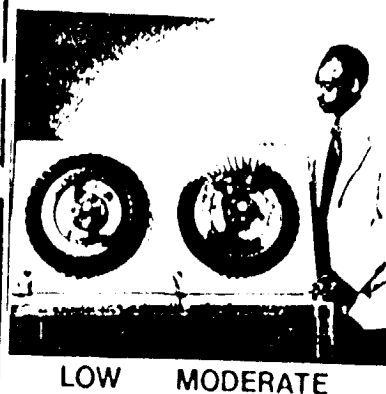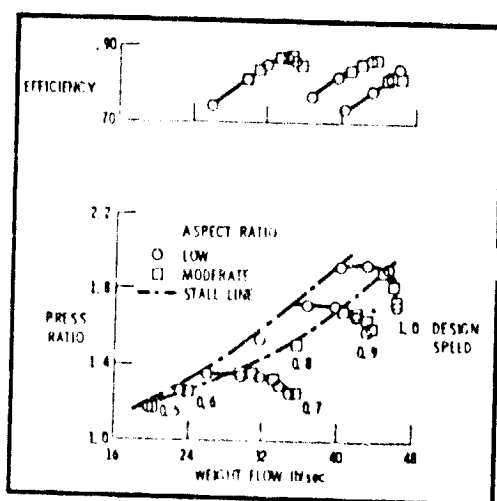
AERODYNAMIC LOSSES

INCIDENCE ANGLE

DEVIATION ANGLE

This figure reflects on the benefit of low aspect ratio type blading for achieving improved flow range in highly loaded high tip speed transonic compressor stages (ref. 39).

NASA
C-81-4187

EFFECT OF ASPECT RATIO ON PERFORMANCE OF HIGH TIP SPEED CORE INLET STAGE
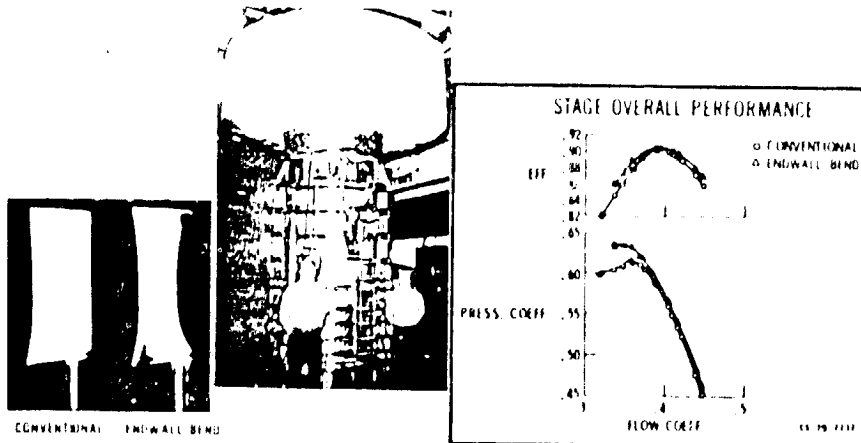

LOW   MODERATE

CD-81-12601

LARGE LOW SPEED COMPRESSOR RESEARCH ENDWALL FLOW FUNDAMENTAL EXPERIMENT

This figure shows the benefits that can be realized by shaping the blade in the endwall region to accommodate the endwall boundary layer. (ref. 40).

LARGE LOW SPEED COMPRESSOR RESEARCH
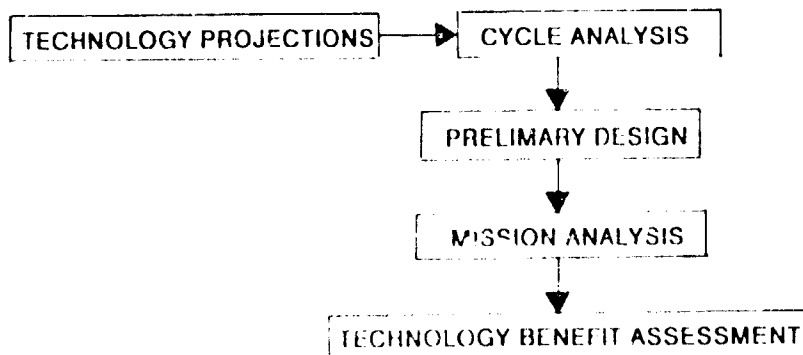### ENDWALL FLOW FUNDAMENTAL EXPERIMENTS

- PROVIDED UNDERSTANDING OF ENDWALL FLOWS
- IMPROVED BLADE SHAPE IN ENDWALL REGION
  TO ACCOMMODATE BOUNDARY LAYER
- RESULTED IN IMPROVED FLOW RANGE AND
  EFFICIENCY

CD 81-12590

36

As part of the selection process trade studies are often made to maximize the benefits of a new design in terms of some figure of merit such as return on investment, ROI or direct operating cost, DOC. The block diagram shown here reflects on this process. Examples of these types of studies are given in reference 41 and 42.

SELECTION PROCESS

```
┌──────────────────────────┐      ┌──────────────────┐
│ TECHNOLOGY PROJECTIONS    │──────▶│  CYCLE ANALYSIS  │
└──────────────────────────┘      └──────────────────┘
                                           │
                                           ▼
                                  ┌──────────────────┐
                                  │ PRELIMARY DESIGN │
                                  └──────────────────┘
                                           │
                                           ▼
                                  ┌──────────────────┐
                                  │ MISSION ANALYSIS │
                                  └──────────────────┘
                                           │
                                           ▼
                    ┌──────────────────────────────────────┐
                    │ TECHNOLOGY BENEFIT ASSESSMENT          │
                    └──────────────────────────────────────┘
```

FLOW MIXING

FLOW MIXING

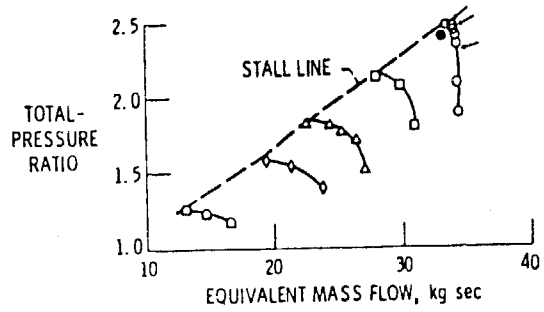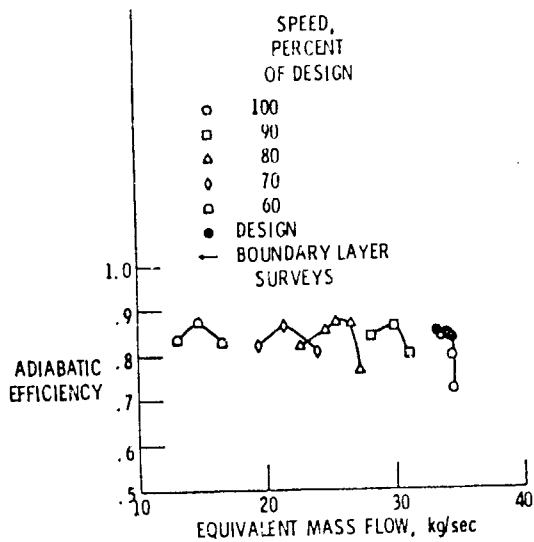The success of multistage compressor designs depends in part on suitable modeling of the effects of radial mixing of the flow through the machine which tends to temper the gradient in flow parameters near the endwalls. Numerous reports and papers have treated various aspects of this phenomenon over the years. References 43-45 specifically address the mixing issue. NASA conducted a study in a two-stage fan where the endwall boundary layers were measured along with the radial distributions of other flow parameters (ref. 46). A photograph of the fan is shown below. The data obtained is presented in the next six figures. The first figure presents the overall performance of the fan and the remaining five figures show the radial distribution in flow parameters near the outer wall. In examining the data one can see that the gradients in flow parameters exiting the rotors are tempered on passing through the stators. This is an indication of the magnitude of the mixing that is taking place in this machine. A feature build into NASA axisymmetric through-flow design code (ref. 21) is the ability to smooth the temperature gradient coming out of the rotor on passing through the stator. Tempering the actual endwall loss gradients used in the design process is also employed.

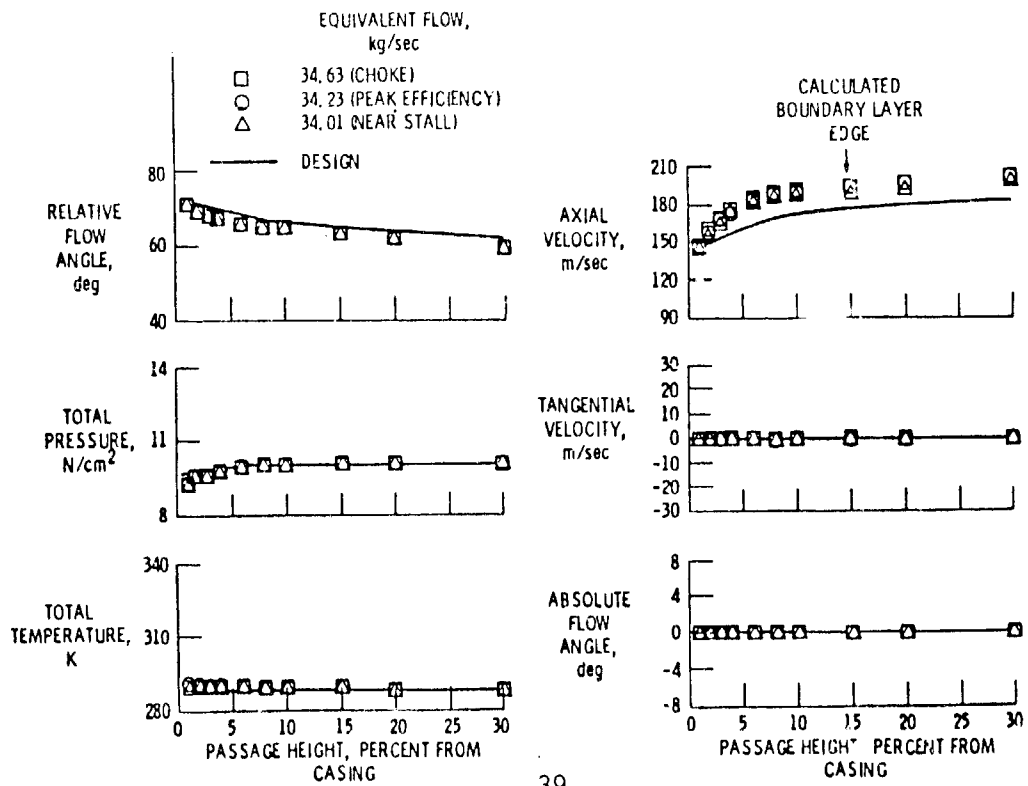TWO-STAGE FAN WITH LOW ASPECT RATIO FIRST STAGE ROTOR
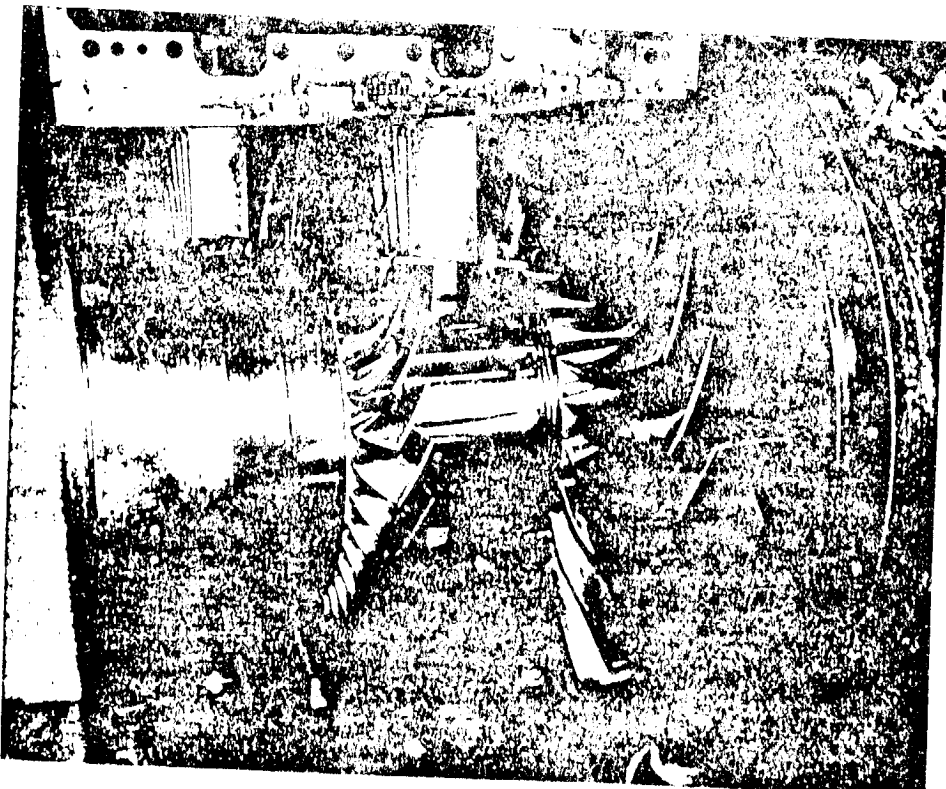
OVERALL PERFORMANCE
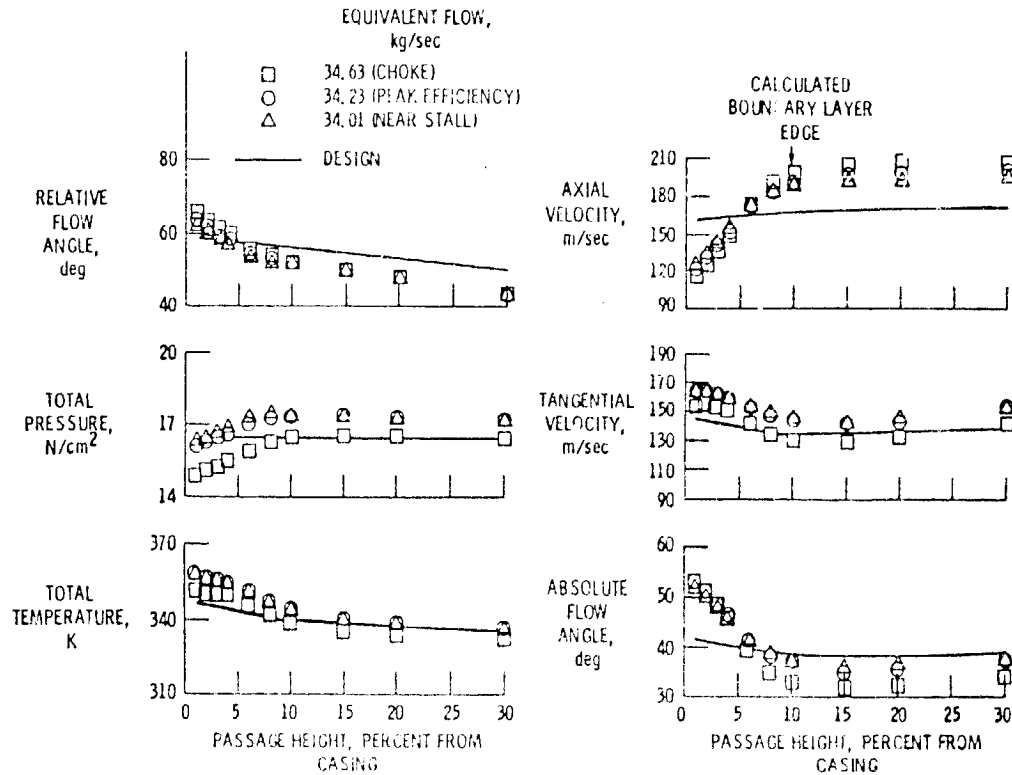
TWO-STAGE, LOW-ASPECT-RATIO FAN

SPEED,
PERCENT
OF DESIGN

o	100
□	90
△	80
◇	70
□	60
●	DESIGN
←	BOUNDARY LAYER SURVEYS

RADIAL DISTRIBUTIONS AT STATION 1

FIRST-STAGE ROTOR INLET

EQUIVALENT FLOW,
kg/sec

□	34.63 (CHOKE)
O	34.23 (PEAK EFFICIENCY)
△	34.01 (NEAR STALL)
—	DESIGN

39

RADIAL DISTRIBUTIONS AT STATION 2

FIRST-STAGE ROTOR EXIT

EQUIVALENT FLOW,
kg/sec

- □   34.63 (CHOKE)
- ○   34.23 (PEAK EFFICIENCY)
- △   34.01 (NEAR STALL)
- —— DESIGN

RADIAL DISTRIBUTIONS AT STATION 3

FIRST-STAGE STATOR EXIT

EQUIVALENT FLOW,
kg/sec

- □   34.63 (CHOKE)
- ○   34.23 (PEAK EFFICIENCY)
- △   34.01 (NEAR STALL)
- —— DESIGN

40
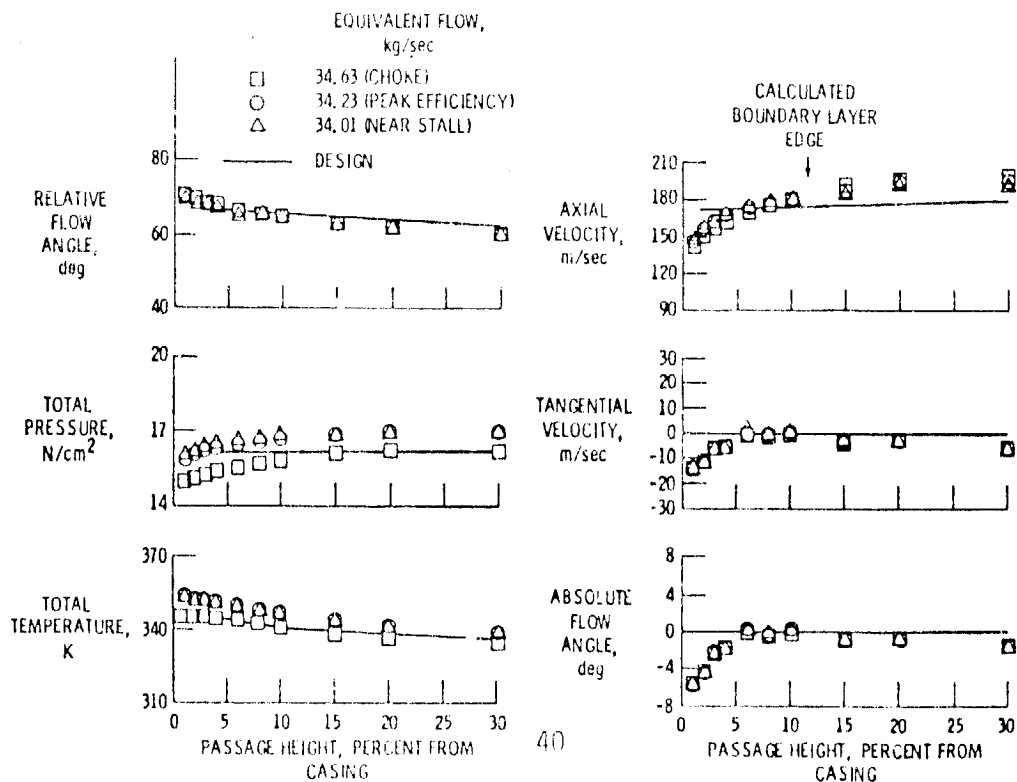
RADIAL DISTRIBUTIONS AT STATION 4

SECOND-STAGE ROTOR EXIT

EQUIVALENT FLOW,
kg/sec

□ 34.63 (CHOKE)
○ 34.23 (PEAK EFFICIENCY)
△ 34.01 (NEAR STALL)
—— DESIGN
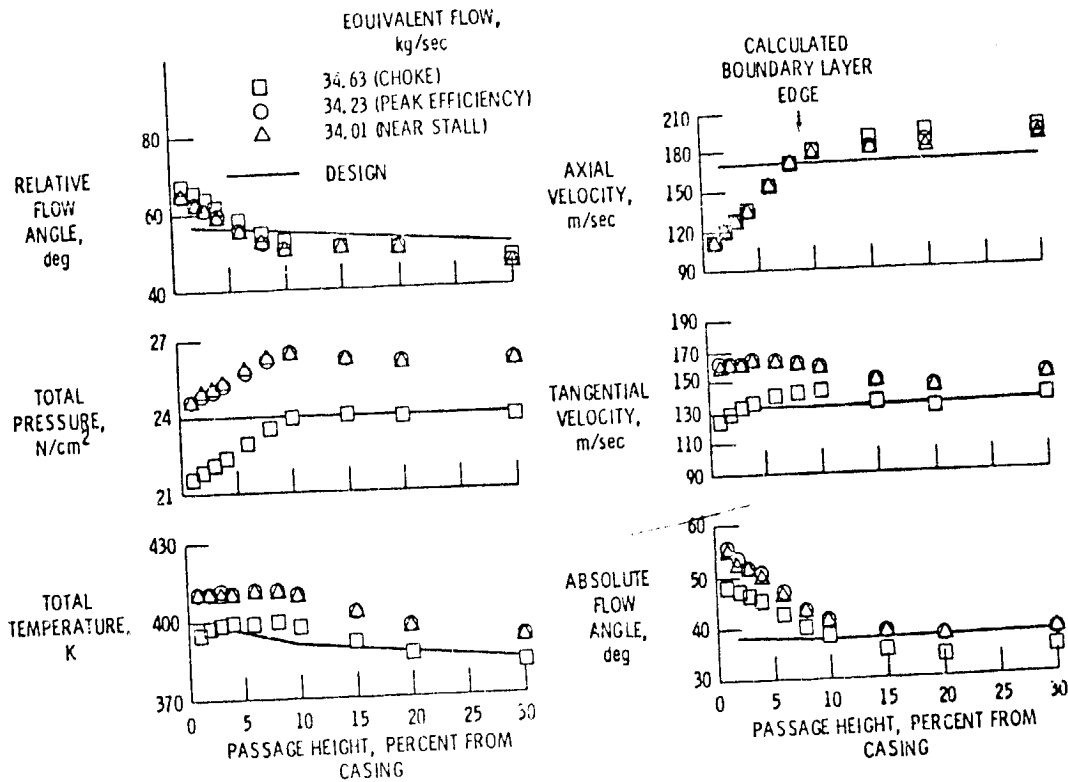
RADIAL DISTRIBUTIONS AT STATION 5

- SECOND-STAGE STATOR EXIT -

EQUIVALENT FLOW,
kg/sec

□ 34.63 (CHOKE)
○ 34.23 (PEAK EFFICIENCY)
△ 34.01 (NEAR STALL)
—— DESIGN

41
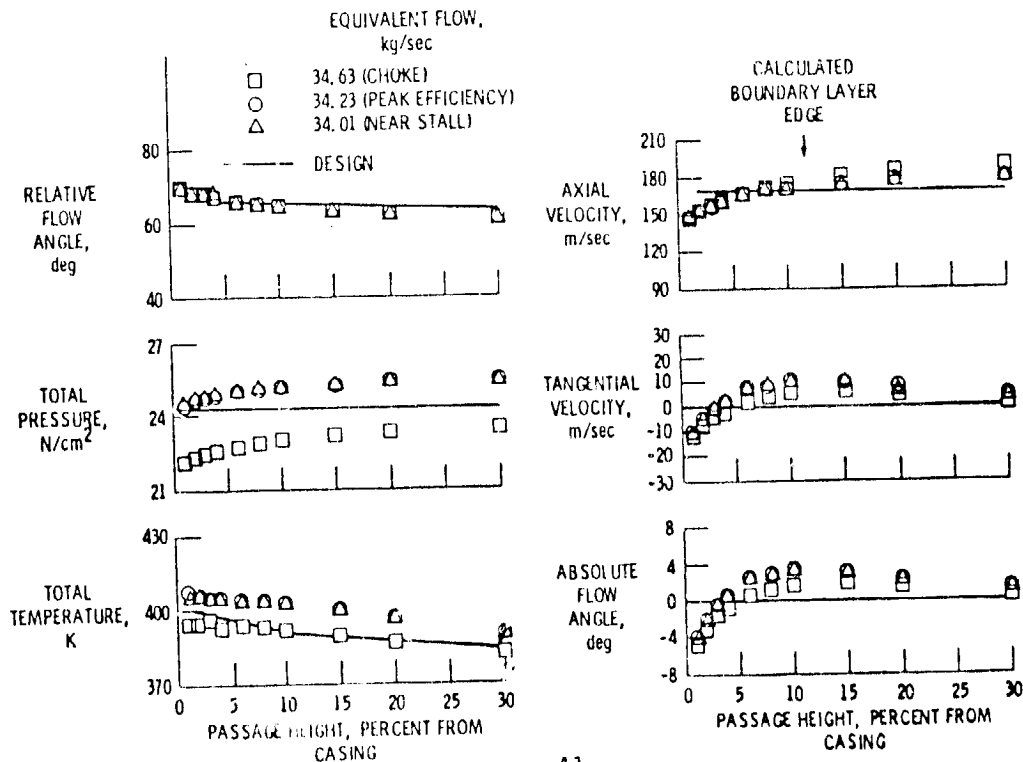
MECHANICAL/AEROELASTIC CONSIDERATIONS

MECHANICAL/AEROELASTIC CONSIDERATIONS

A number of mechanical and aeroelastic consideration must be addressed in the design process. These are listed below. Blade untwist has been taken into account for many years by resetting blade elements so they untwist to the design setting at design speed. Only recently has uncambering been considered in the design process. The increased blade speeds coupled with low aspect ratio blades makes this an important consideration. The structural analysis codes like NASTRAN are employed in this process. Flutter is not a common problem encountered in advance relatively low aspect ratio blading, however, it must be considered in the design process to assure avoidance. Forced response can be a problem. Exit rim stresses may limit the exit diameter of the machine. Variable geometry must be incorporated in high pressure ratio machines to improve part speed performance. The trend is to use shrouded stators in at least the front variable rows. Methods for maintain tight clearances are of particular importance to achieving good efficiencies, particularly for the high hub/tip ratio rear stages.

MECHANICAL/AEROELASTIC CONSIDERATIONS

- BLADE UNTWIST/UNCAMBER
- FLUTTER
- FORCED RESPONCE
- RIM STRESS
- VARIABLE GEOMETRY
- CANTILEVERED/SHROUDED STATORS
- TIP CLEARANCE CONTROL

REGIONS OF COMPRESSOR FLUTTER

The performance map shown here reflects regions where flutter may be encountered. Subsonic/transonic flutter is the type most encountered in modern multistage designs and is the least understood.

REGIONS OF COMPRESSOR FLUTTER

SUPERSONIC
STALL

STALL LINE

SUPERSONIC UNSTALLED
HIGH BACKPRESSURE

SUB/TRANSONIC
STALL

PRESSURE
RATIO

SUPERSONIC
UNSTALLED LOW
TO MODERATE
BACKPRESSURE

50%

CHOKE

75%

100%

WEIGHT FLOW

44

Shown here is a NASA Lewis research facility devoted to gaining an understanding of subsonic/transonic stall flutter.
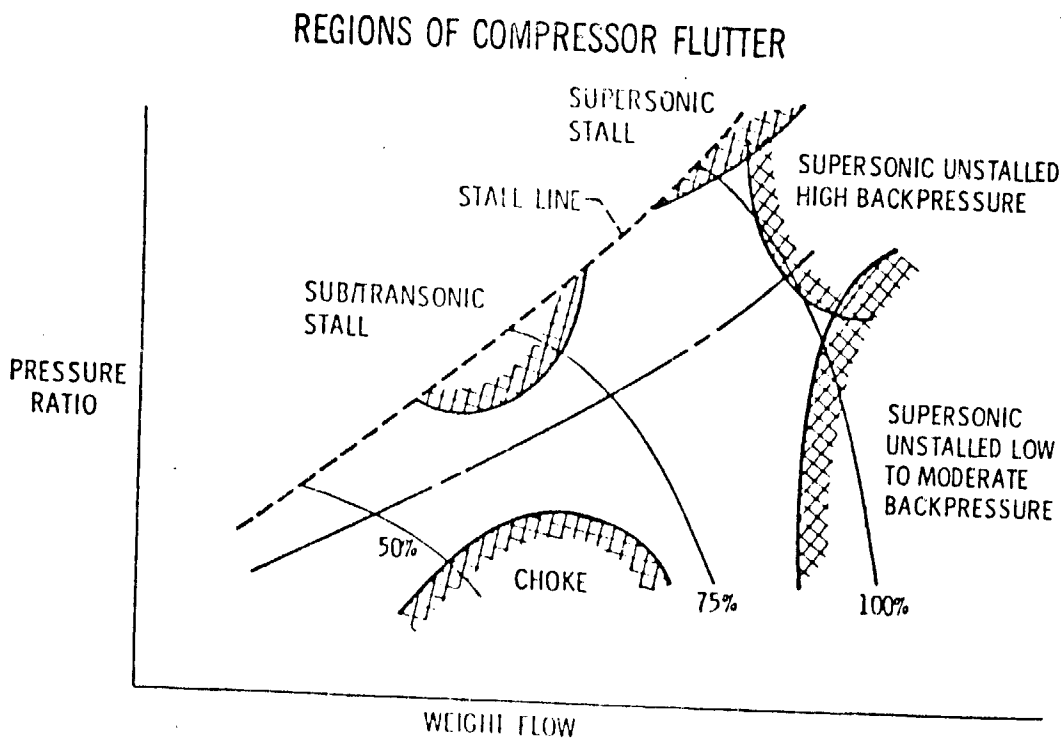
NASA
C-81-3324

FAN AND COMPRESSOR RESEARCH
TRANSONIC FLUTTER
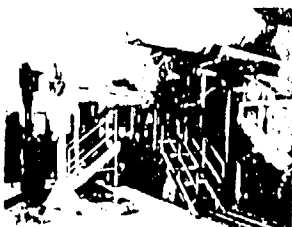
FLUTTER CASCADE FACILITY

NINE-AIRFOIL TEST SECTION

MACH 0.8 STEADY FLOW
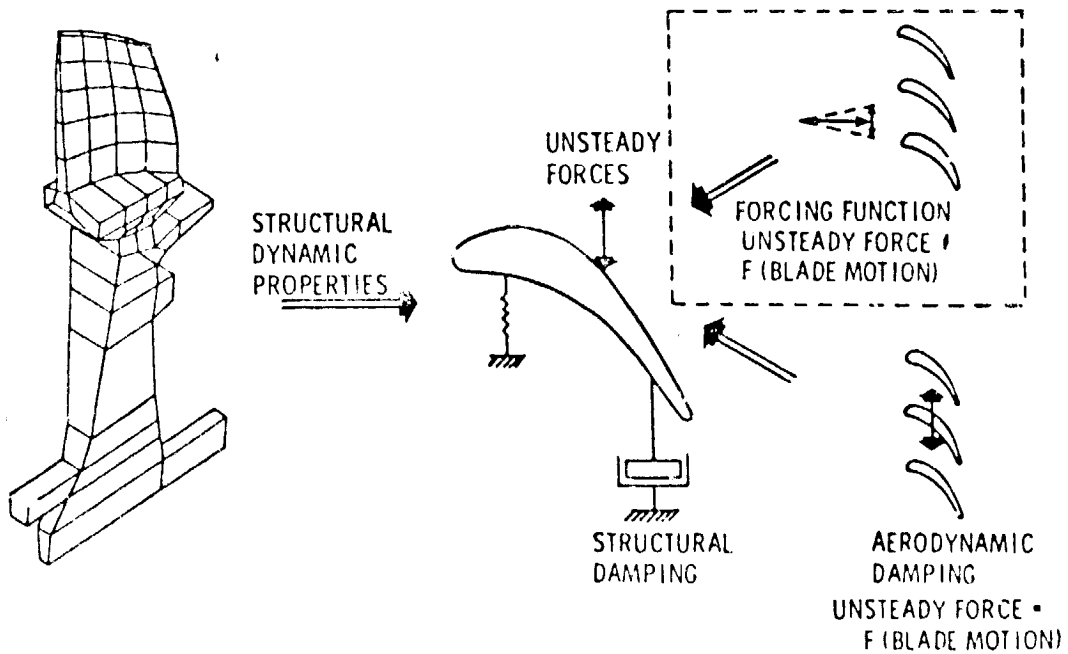
1.2
Δ 0 0
-1.2

TIME

TIME
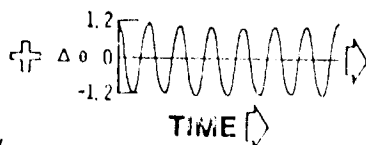
AIRFOIL DISPLACEMENT
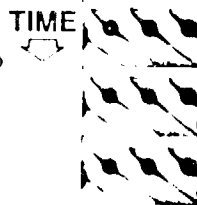
FLOW OVER AIR AIRFOILS
OSCILLATING AT 400 HZ

CD-81-12599

PRIMARY ELEMENTS OF FORCED RESPONSE

This figure shows the primary elements of forced response. The forced response problem is triggered by nonaxisymmetric flows. The following figure list sources of these flows.

PRIMARY ELEMENTS CF FORCED VIBRATION

STRUCTURAL
DYNAMIC
PROPERTIES

UNSTEADY
FORCES

FORCING FUNCTION
UNSTEADY FORCE ·
F (BLADE MOTION)

STRUCTURAL
DAMPING

AERODYNAMIC
DAMPING

UNSTEADY FORCE ·
F (BLADE MOTION)

46

Of the various sources of non-axisymmetric flows listed the ones due to struts and blade/vane wakes are the most common ones considered in the design process.

SOURCES OF NON-AXISYMMETRIC FLOWS

- INLET DISTORTED FLOW FIELDS
  - PRESSURE
  - TEMPERATURE
- WAKE GENERATION AND TRANSPORT
  - STRUTS
  - BLADE/VANE WAKES
  - TIP VORTICES
  - ENDWALL BOUNDARY LAYERS
  - INSTRUMENTATION PROBES
- POTENTIAL FLOW FIELD DISTURBANCES
- ROTATING STALL CELLS
- SURGE
- BLEED PORTS

CAMPBELL DIAGRAM SHOWING FORCED EXCITATION POTENTIAL

This figure shows the commonly used diagram for identifying the potential for forced response. The lower order excitation are normally due to struts and avoidance of crossing one and two order exitations is common practice. The higher order excitations are due normally to rotor/stator interactions. However, in reality excitation can be noted throughout the map. This can be seen from the next figure.

CAMPBELL DIAGRAM SHOWING FORCED EXCITATION POTENTIALS
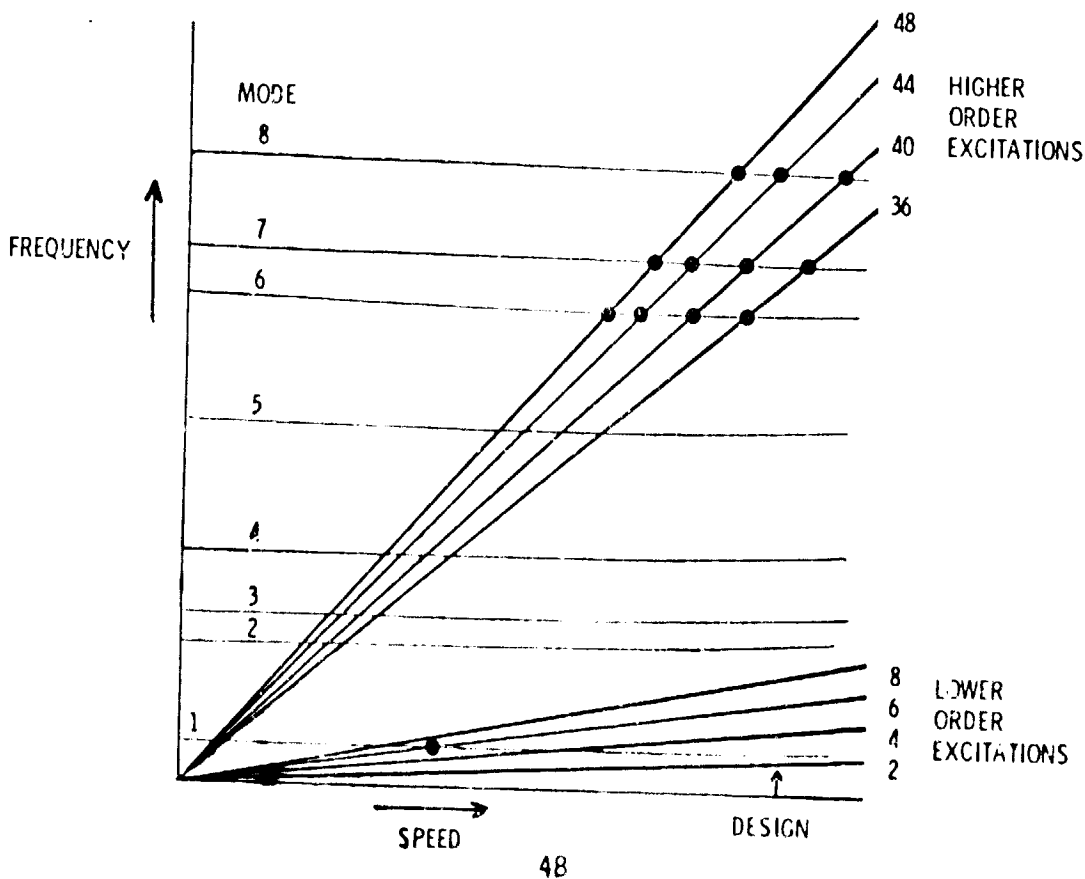
Tnis data was recently taken by Fred Newman on the NASA three-stage research compressor shown earlier. The data indicated by the symbols reflect the range of excitations encountered. The data is now being analyzed to determine the aerodynamic damping involved in the various modes and to develop models and codes for predicting these modes during the design process. Present modeling is inadequate to establish the magnitude of these forced responses.

NASA THREE–STAGE RESEARCH COMPRESSOR
THIRD STAGE ROTOR
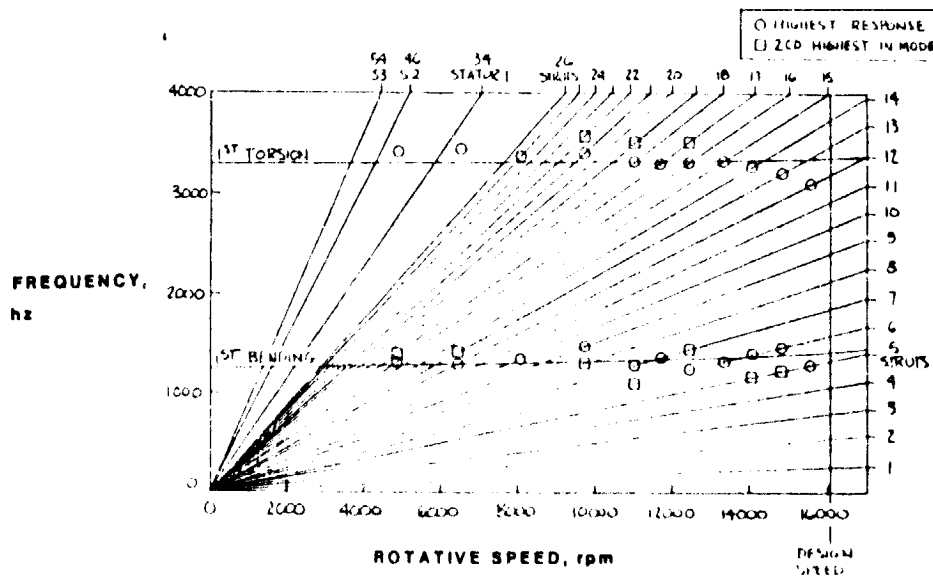
OTHER CONSIDERATIONS

Other considerations which the designer must address are listed below. Of all the items listed, the one which has received most attention recently is stall recoverability. Some of the basic modeling treating this area are presented in references 47-55. Structural loads encountered due to surge is treated in references 56 and 57.

OTHER CONSIDERATIONS

STALL MARGIN
INLET DISTORTION TOLERANCE
LOAD LINE
STALL RECOVERABILITY

Significant progress has been made in the modeling of this phenomena. However, additional work is needed to further address some of the issues listed below.

STALL RECOVERY ISSUES/CONSIDERATIONS

INLET/EXIT VOLUME/GEOMETRY

COMPRESSOR/BURNER INTERACTIONS

LUMPED VS STAGE – BY - STAGE COMPRESSOR MODEL

REVERSED FLOW CHARACTERISTICS

COMPRESSOR INERTANCE

HYSTERESIS

TRIGGERING MECHANISMS – INLET DISTORTIONS

FAN/LOW COMPRESSOR VS HIGH COMPRESSOR

SINGLE CELL VS MULTIPLE CELL/PART SPAN VS FULL SPAN

TURBOFAN VS TURBOJET VS TURBOSHAFT

SMALL ENGINES – AXIAL/CENT AND CENT/CENT

EXAMPLES OF MODERN DAY
TRANSONIC COMPRESSORS

NASA ENERGY EFFICIENT ENGINE
CORE COMPRESSORS

Shown here is a photograph of the P&W core compressor along with the overall design values given earlier.

**NASA/P&W ENERGY EFFICIENT ENGINE
CORE COMPRESSOR**
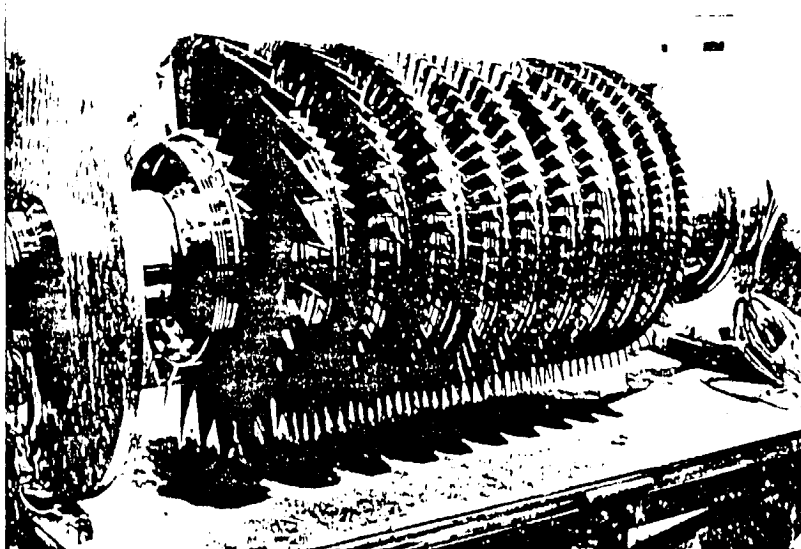
STAGES – 10
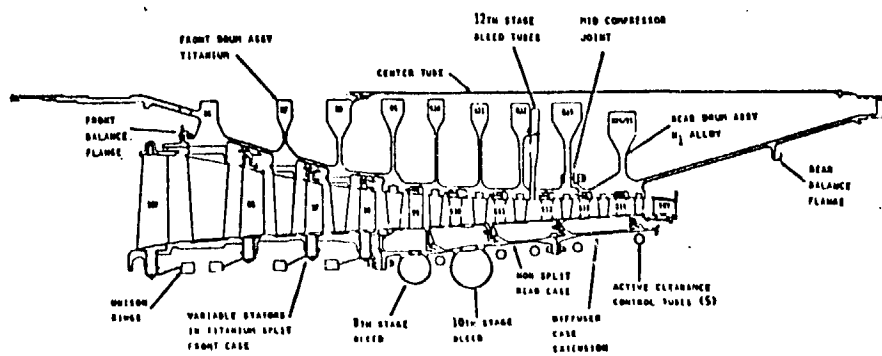PRESSURE RATIO – 14:1
TIP SPEED – 379 m/sec (1243 ft/sec)

Shown here is a cross-section of the P&W compressor. The compressor achieved its design pressure and an efficiency of 85.5% at design point conditions (ref. 18). This is considered excellent performance.

NASA/P&W ENERGY EFFICIENT ENGINE CORE COMPRESSOR

CROSS – SECTION

ADIABATIC EFFICIENCY OF 85.5%
DEMONSTRATED AT DESIGN POINT
CONDITIONS

56

Shown here is a photograph of the GE core compressor along with its design goals shown earlier.
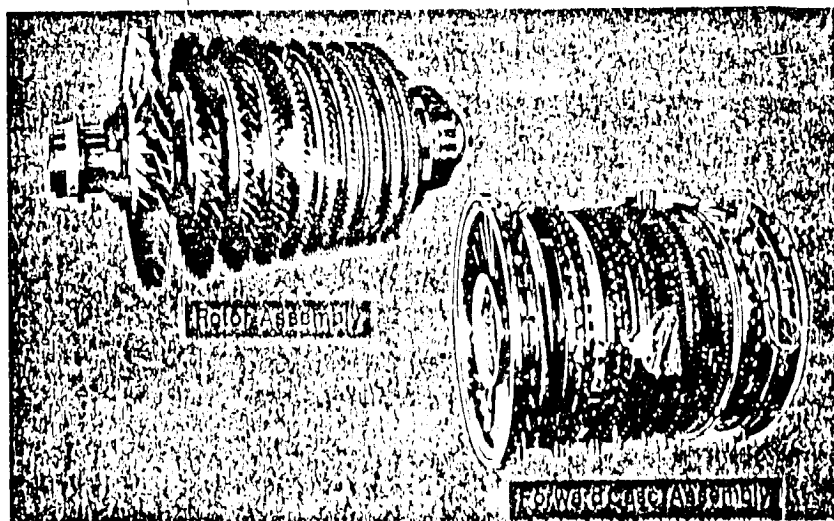
NASA/GE ENERGY EFFICIENT ENGINE CORE COMPRESSOR

STAGES – 10
PRESSURE RATIO – 23:1
TIP SPEED – 456 m/sec (1495 ft/sec)

Shown here is a cross-section of the GE core compressor. The next several figures reflect on various design features of this compressor. Detailed discussions of the design is given in reference 58. One can see the flow path geometry is between a constant mean diameter and a constant tip. Shrouded stators are employed in all stages. Active clearance control is incorporated within the casing over the rear stages. Also note that the compressor incorporates a variable IGV. It serves two purposes, one to maximize performance at off design conditions and two, to impart a swirl to reduce the rotor tip Mach number at the high design tip speed. The swirl imparted is not of the magnitude to reduce the reaction to 50% in the front stage.

NASA/GE ENERGY EFFICIENT ENGINE CORE COMPRESSOR

CROSS – SECTION
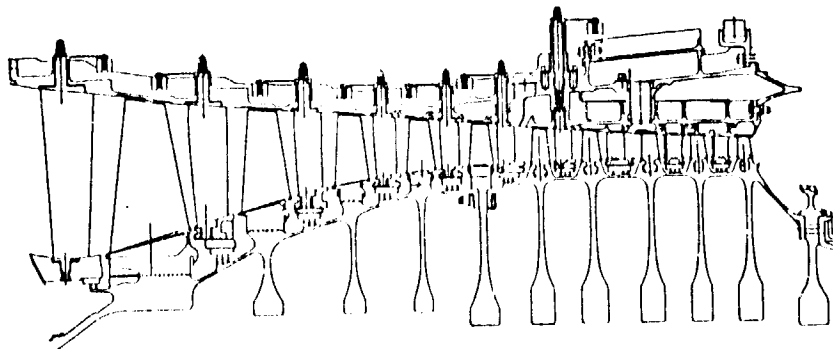
NASA/GE ENERGY EFFICIENT ENGINE CORE COMPRESSOR ASPECT RATIO

Shown here is the aspect ratio of the rotor and stator blading. Note the trends from front to rear. The aspect ratio of the rear stages is reduced, in part, to maintain acceptable blade and vane cords. Also, note the higher aspect ratio in the front stage stators. Stator aspect ratio can be higher in these stages compared to the rotor and maintain good efficiency with acceptable operating range at the highly transonic flow condition.

NASA/GE ENERGY EFFICIENT ENGINE CORE COMPRESSOR

ASPECT RATIO

STAGE NUMBER
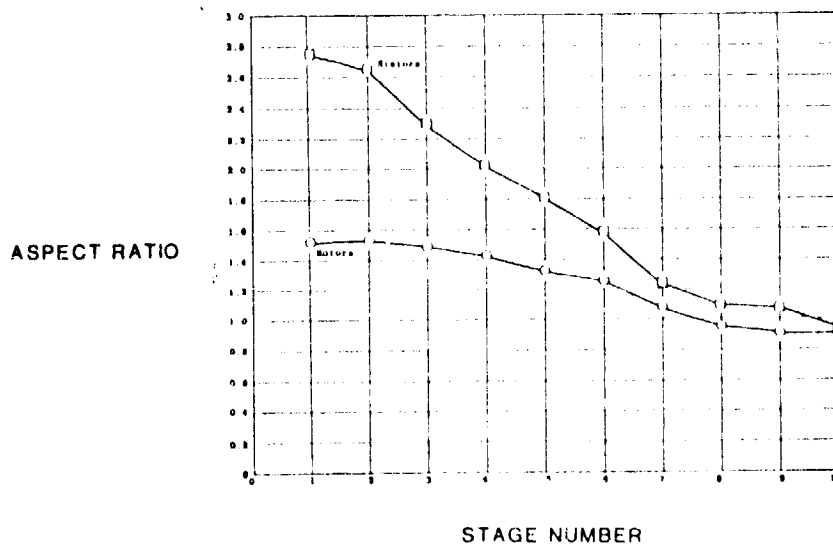
NASA/GE ENERGY EFFICIENT ENGINE CORE COMPRESSOR
PITCH-LINE SOLIDITY

Shown here is the pitch-line solidities for the rotors and stators. Note the
higher solidity for the front stage rotors as compared to the rear stages.
The higher solidities are required on the front rotors because of the highly
transonic flow conditions. The stator solidity is more or less constant
except for the last stator where the solidity was increased significantly to
reduce loading in this vane row.

NASA/GE ENERGY EFFICIENT ENGINE
CORE COMPRESSOR

PITCH–LINE SOLIDITY

PITCH-LINE SOLIDITY

STAGE NUMBER

60
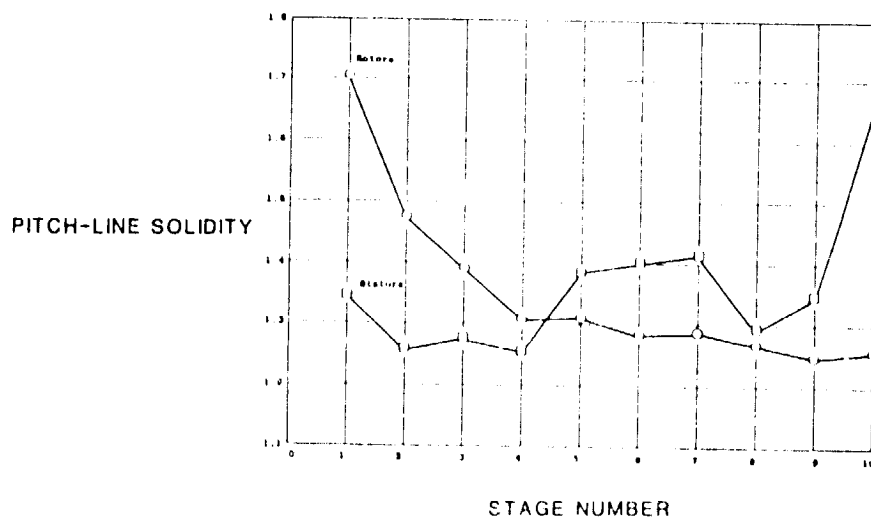
NASA/GE ENERGY EFFICIENT ENGINE CORE COMPRESSOR
PITCH-LINE MERIDINAL MACH NUMBER

Shown here is the pitch-line meridinal Mach number. Note the decrease from approximately 0.6 at the inlet to approximately 0.28 at the OGV exit. The high Mach number at the inlet reduces frontal area and enhances transonic operation. The lower Mach number at the exit reduces the amount of diffusion that has to take place between the compressor exit and the combustor inlet.

NASA/GE ENERGY EFFICIENT ENGINE CORE COMPRESSOR

PITCH-LINE MERIDIONAL MACH NUMBER

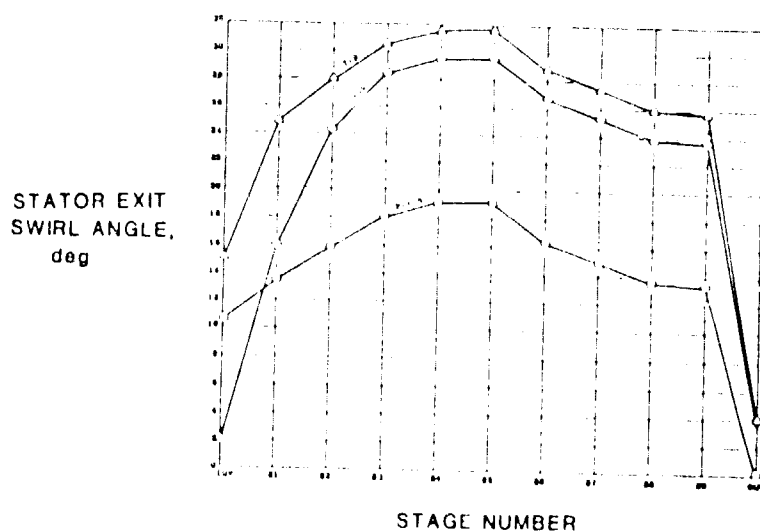PITCH-LINE MERIDIONAL MACH NUMBER

STAGE NUMBER
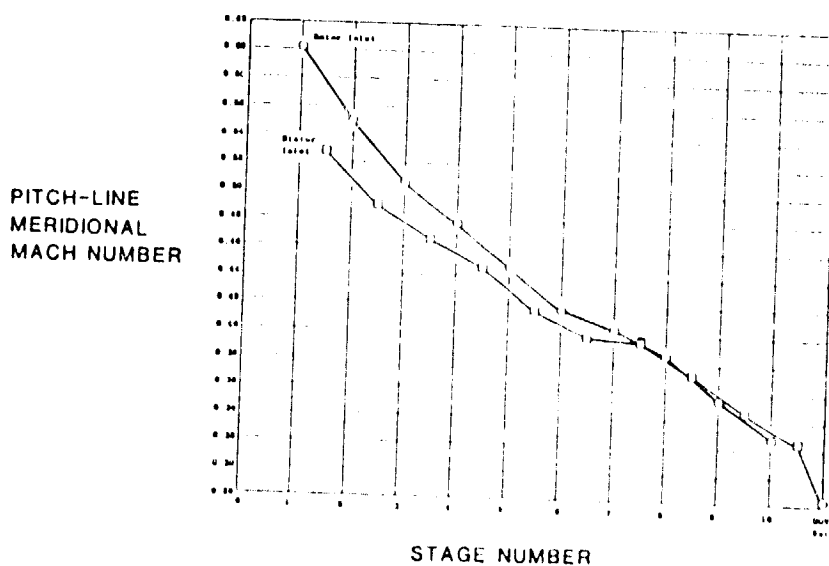
NASA/GE ENERGY EFFICIENT ENGINE CORE COMPRESSOR STATOR EXIT SWIRL ANGLE

Shown here is the stator exit swirl angle for the tip, pitch and hub sections. Note the high swirl leaving the tip section of the IGV and the low value at the hub. This reduced the rotor tip Mach number and limits the stator one inlet hub Mach number to approximated .85, considered a practical upper limit for achieving good stator performance. In general, the swirl is increased in the front stages with the limiting stator hub Mach number determining the rate of increase. The high swirl in the endwall regions compared to pitch values for the middle stages is incorporated due to considerations of the endwall boundary layer. Note a OGV is used to remove the remaining swirl leaving the last stator.

NASA/GE ENERGY EFFICIENT ENGINE CORE COMPRESSOR

STATOR EXIT SWIRL ANGLE

STATOR EXIT
SWIRL ANGLE,
deg

STAGE NUMBER

Shown here is the flowpath from the axisymmetric through-flow design code
(CAFD). Note the calculation station placed within the first four rotor
blades. This was done to help in establishing the blade shape for these high
Mach number blade rows. The modeling of the inlet and exit is also shown.
This is done to assure the flow parameters entering and leaving the compressor
are properly defined.

NASA/GE ENERGY EFFICIENT ENGINE
CORE COMPRESSOR

CAFD FLOWPATH

RADIUS,
in

AXIAL LOCATION, in

Shown here is the pitch-line D-factor. Note the increase in stator loading in the rear stages. This is due in part to the removal of a portion of the total swirl existing at the exit of stator five as the flow passes through stators six to nine. This is required to keep the swirl that needs to be removed on the tenth stator and OGV within practical limits. The lower diffusion factors in stator 6 and 7 is due in part to higher solidities in those stators. This resulted from consideration of compressor bleed requirements.

NASA/GE ENERGY EFFICIENT ENGINE
CORE COMPRESSOR

PITCH-LINE DIFFUSION FACTOR
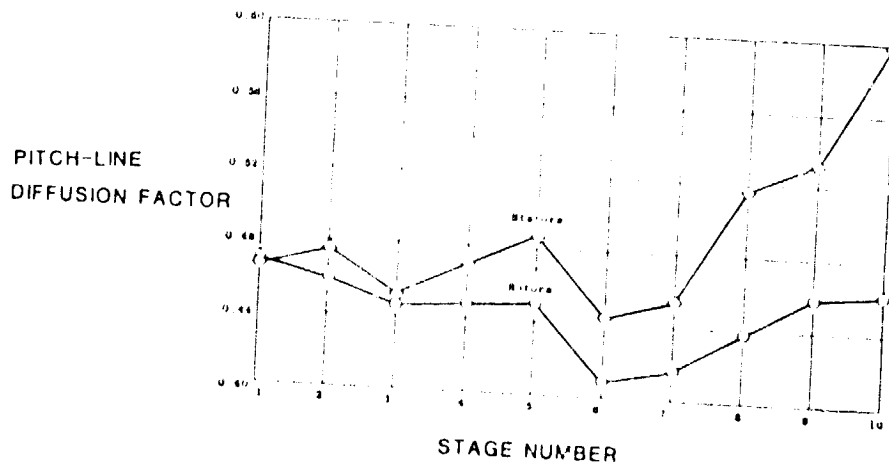
STAGE NUMBER

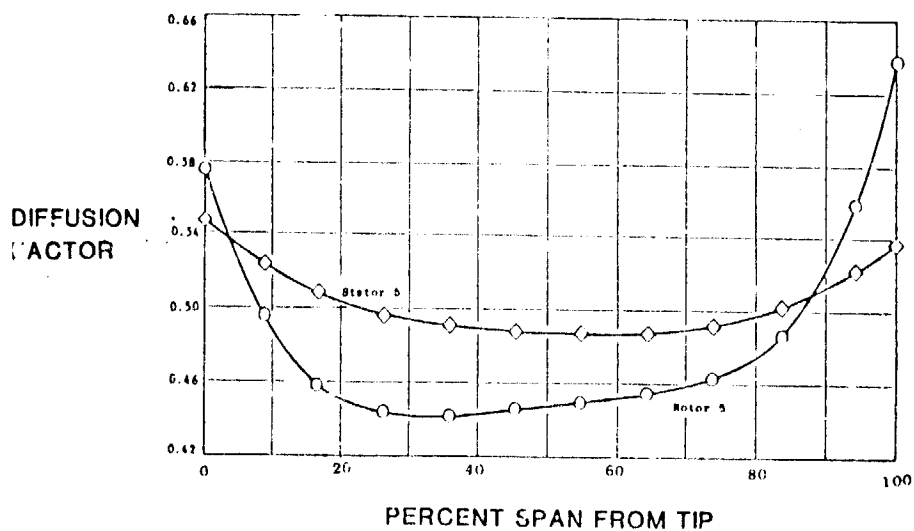## NASA/GE ENERGY EFFICIENT EINGINE CORE COMPRESSOR
RADIAL DISTRIBUTION OF DIFFUSION FACTOR

Shown here is the radial distribution of diffusion factor for rotor 5 and
stator 5. Note the increases in the endwall regions. This was req ired to
obtain the desired radial pressure gradients.

NASA/GE ENERGY EFFICIENT ENGINE CORE COMPRESSOR

RADIAL DISTRIBUTION OF DIFFUSION FACTOR

PERCENT SPAN FROM TIP

Note the tip Mach number for the first rotor. This Mach number would be
higher if inlet swirl was not employed. Note the hub Mach number for stator
one of approximately .85, considered the practical limit. The Mach number
decreases through the machine due in part to increasing temperature and
somewhat to decreasing axial velocity.

NASA/GE ENERGY EFFICIENT ENGINE
CORE COMPRESSOR

HUB INLET MACH NUMBER

HUB INLET
MACH NUMBER

Max. Rotor Mach Number

Rotor Tip

Stator Hub

STAGE NUMBER

66
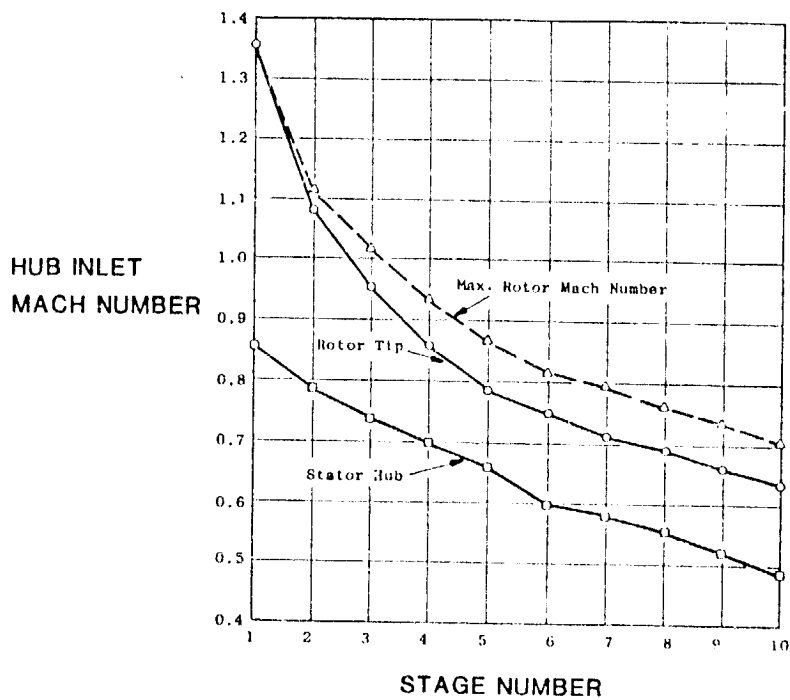
NASA/GE ENERGY EFFICIENT ENGINE
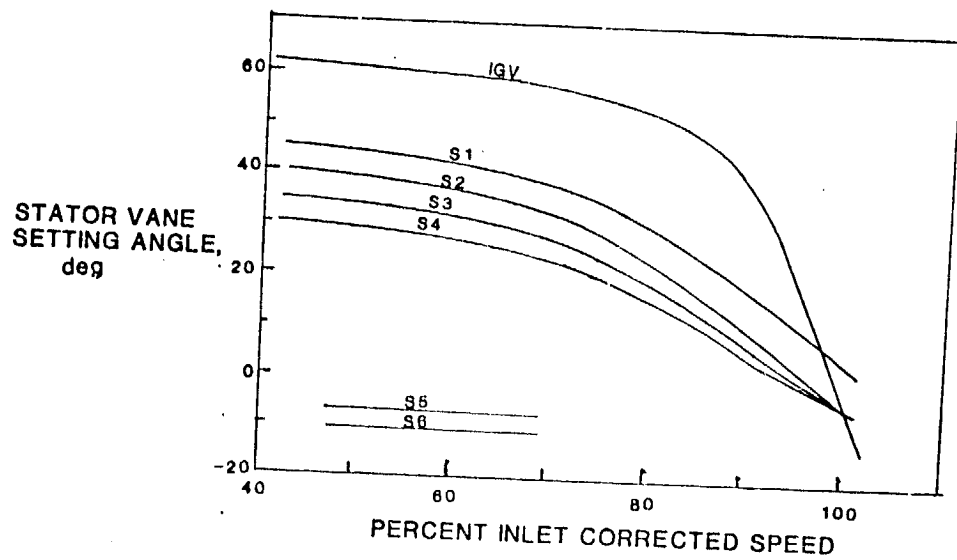CORE COMPRESSOR

STATOR SCHEDULE

Shown here is the schedule for the IGV and stators. Note the high degree of
reset required at the part speed condition particularly in the IGV and stator
one.

NASA/GE ENERGY EFFICIENT ENGINE
CORE COMPRESSOR

STATOR SCHEDULE

STATOR VANE SETTING ANGLE, deg vs **PERCENT INLET CORRECTED SPEED**

Curves labeled IGV, S1, S2, S3, S4, S5, S6

NASA/GE ENERGY EFFICIENT ENGINE CORE COMPRESSOR
OVERALL PERFORMANCE MAP

Shown here is the overall performance of the GE compressor (ref. 58). Design weight flow and pressure ratio were exceeded at design speed with a adiabatic efficiency of approximately 83%. This is considered excellent for such a high overall pressure ratio. Part speed efficiencies approach 85%. The program was a complete success and sets the standard for high pressure ratio cores.

NASA/GE ENERGY EFFICIENT ENGINE CORE COMPRESSOR

OVERALL PERFORMANCE MAP
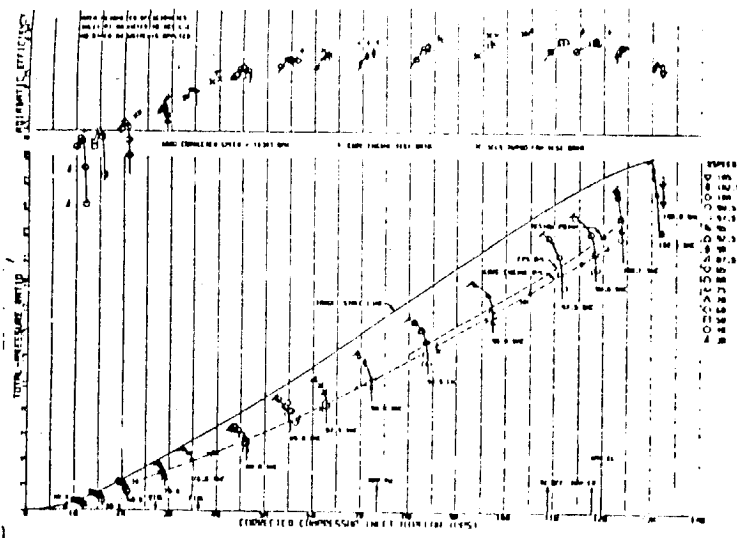
NASA FIVE-STAGE RESEARCH COMPRESSOR

NASA FIVE-STAGE RESEARCH COMPRESSOR

In the late sixties/early seventies, NASA embarked on a program to further
enhance the performance of transonic multistage compressors from that
demonstrated in the NACA eight-stage and NACA five-stage compressors. Our
first design was a seven-stage compressor having an overall pressure ratio of
9.27 and an inlet tip speed of 1200 ft./sec. A mechanical failure occurred
early in the testing of this design. The decision was made to not rebuild
which would have required complete replacement of the blades and vanes but to
concentrate on a second more aggressive design. The second design was a
five-stage compressor having the same overall pressure ratio of the
seven-stage (9.27) but with an inlet tip speed of 1400 ft./sec. To put this
compressor in perspective technology wise, the NACA five-stage had a design
pressure ratio of 5.0 and an inlet tip speed of 1100 ft./sec. The NASA
five-stage compressor was a very aggressive design for its time period and
remains so today. A low aspect ratio version of the original design has
demonstrated excellent performance (ref. 20). Many lessons were learned to
get to this level of performance. The following figures tell the story.

NASA FIVE-STAGE RESEARCH COMPRESSOR
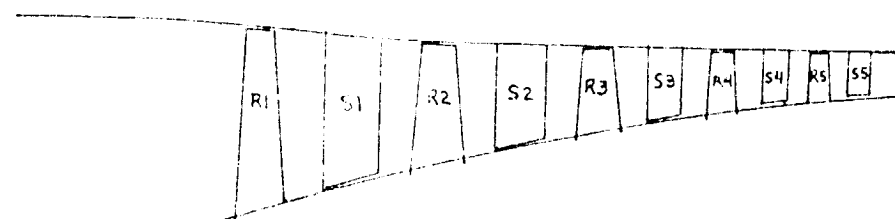## ORIGINAL DESIGN

Shown here is the flowpath and some of the design parameters for the original NASA five-stage design. Note the high overall design efficiency. The design efficiencies for the rear stages were particularly high. The efficiencies were based on a very limited amount of data considering the aerodynamic loading levels and blade/vane aspect ratios. However, the excellent part speed performance of the NACA five-stage and even more so the eight-stage compressor gave encouragement in being able to achieve this level of performance. The lower performance of these machines at their design speed was attributed much to stage matching problems. There was also a philosophy at the time to achieve good efficiency one must design optimistically. In retrospect, the design was too aggressive considering the data base that existed for selecting the design parameters consistent with the blade loading and highly transonic conditions. The direction at the time was to increase stage pressure ratio through increased aerodynamic blade/vane loading and blade speed, and at the same time increase the blade and vane aspect ratios to further reduce overall length. This turned out to be an unacceptable design philosophy for achieving good overall efficiencies and flow range. One might note the relatively low aspect ratios of the NACA eight and five-stage machines from the photographs shown earlier.

An IGV was incorporated into the design but did not impart a swirl at design setting to reduce the relative flow mach number entering the first rotor, and all stators turned the flow back to the axial direction, thus a high stage reaction design throughout like the NACA five-stage.

NASA FIVE—STAGE RESEARCH COMPRESSOR

ORIGINAL DESIGN

STAGE					
PRESSURE RATIO	1.60	1.73	1.67	1.51	1.32
EFFICIENCY	.86	.89	.90	.90	.92

OVERALL
PRESSURE RATIO 9.27
EFFICIENCY .87
WEIGHT FLOW 65.5 lb/sec

NASA FIVE-STAGE RESEARCH COMPRESSOR
ORIGINAL DESIGN

The aerodynamic loading (stage pressure ratio) distribution was chosen purposely to unload the inlet and outlet stages in an attempt to enhance off-design operation. The idea was that the middle stages could sustain the higher loadings because they tend to remain near their design point at part speed (off-design) operation. The blade/vane aspect ratios chosen were considerably higher than those of the NACA eight and five-stage machines as noted earlier. Because of the high aspect ratio of the first rotor, a damper (part span shroud) was used to eliminate potential aeroelastic problems. Little data base existed for stages incorporating dampers. A blockage allowance of two percent at the inlet of the compressor and five percent at the exit was assumed in the design, and like that measured in the NACA five-stage. The NACA eight-stage incorporated a 20 percent blockage at the exit. A large overall axial velocity decrease of 0.7 was taken across the machine (nearly linear distribution) to reduce the amount of velocity diffusion between the compressor exit and combustor inlet. This further increased aerodynamic loading for a given overall pressure ratio. The aerodynamic loadings were much higher than for the NACA five-stage.

NASA FIVE-STAGE RESEARCH COMPRESSOR

ORIGINAL

| ROTOR					
TIP RELATIVE MACH NO	1.35	1.23	1.09	1.00	0.93
TIP DIFFUSION FACTOR	.42	.52	.55	.51	.38
ASPECT RATIO	3.08	2.11	1.66	1.64	1.68
STATOR					
HUB MACH NO	0.69	0.66	0.59	0.50	0.39
HUB DIFFUSION FACTOR	.45	.62	.68	.66	.50
ASPECT RATIO	2.70	2.04	2.03	2.30	2.27

PHOTOGRAPH

Shown here is a photograph of the original NASA five-stage research compressor
with one half of the casing removed. One can note the relatively high
blade/vane aspect ratio and the damper incorporated into the first rotor
design.

NASA FIVE-STAGE RESEARCH COMPRESSOR
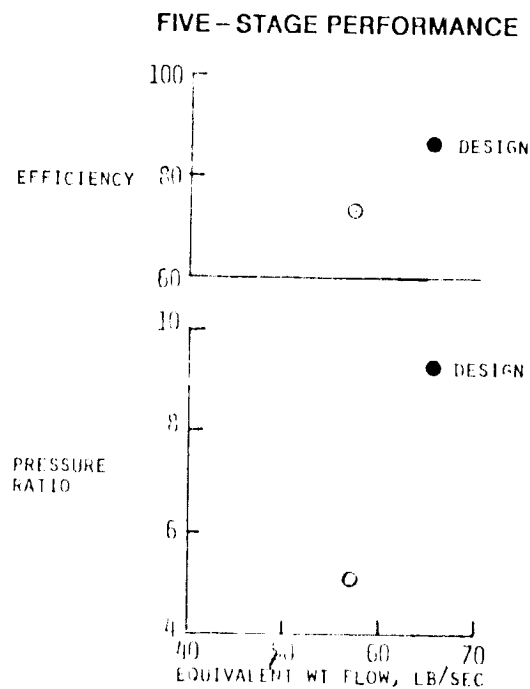## ORIGINAL

FIVE-STAGE HARDWARE
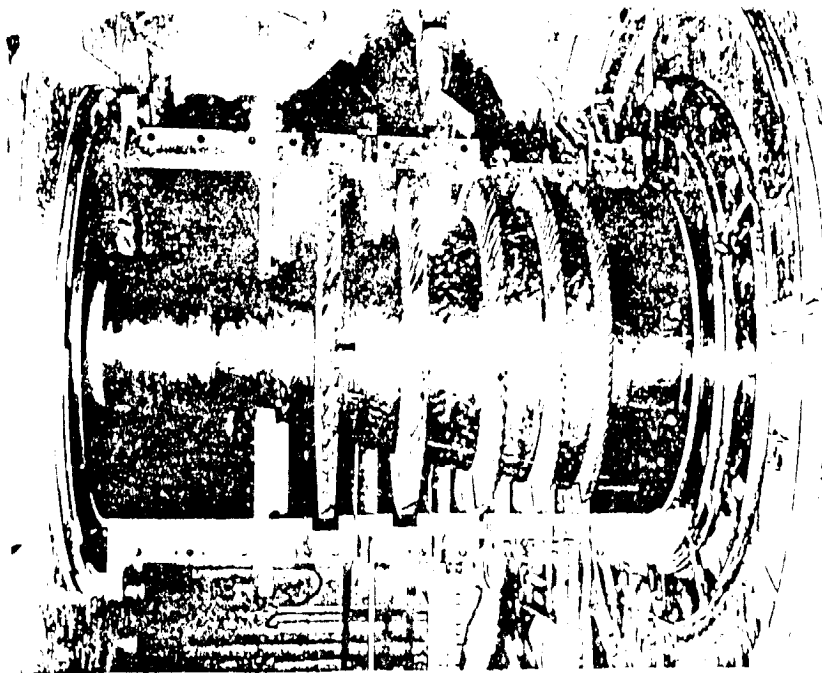
NASA FIVE-STAGE RESEARCH COMPRESSOR
ORIGINAL DESIGN

FIVE-STAGE PERFORMANCE

Shown here is the overall performance obtained at design speed for the NASA
five-stage compressor as originally designed. As can be seen, a pressure
ratio of only five was demonstrated and the flow was well below design. The
data indicated that the front stages remained in stall and the back stages in
choked at the design speed. The use of the variable IGV and stators along
with interstage bleed at the exit of rotors two, three and four was not
sufficient to bring the front stage out of stall and the back out of choke.
Insufficient flow range at all speeds made it difficult to characterize the
individual stages. The decision was made to remove the back two stages and
try to obtain more complete data on the front three.

NASA FIVE — STAGE RESEARCH COMPRESSOR
ORIGINAL
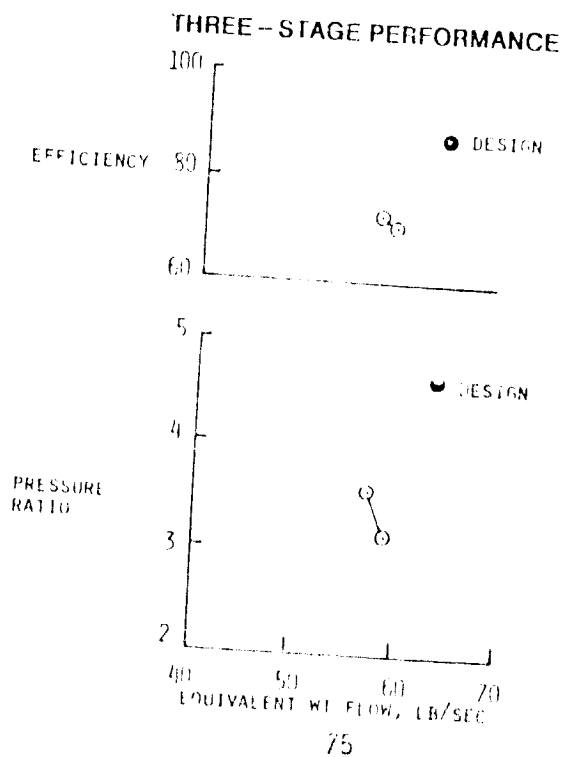
FIVE — STAGE PERFORMANCE

74

NASA FIVE-STAGE RESEARCH COMPRESSOR
ORIGINAL DESIGN

THREE-STAGE PERFORMANCE

Shown here is the performance obtained for the front three stages of the
original NASA five-stage compressor. Even though more flow range was obtained
than for the full five stages, the performance remained well below design.
The design pressure ratio for the front three stages was 4.6. The decision
was then made to test the front stage alone (ref. 59).

NASA FIVE – STAGE RESEARCH COMPRESSOR
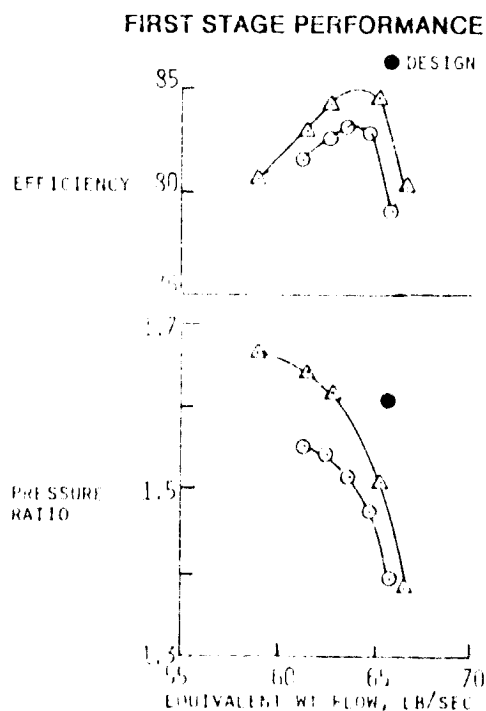## ORIGINAL

THREE – STAGE PERFORMANCE

NASA FIVE-STAGE RESEARCH COMPRESSOR
ORIGINAL DESIGN

FIRST-STAGE PERFORMANCE

Shown here is the performance of the first stage of the original NASA
five-stage compressor. Data is shown for a solid casing (circles) and with
casing treatment (triangles). Significant improvements in range and
efficiency were obtained with casing treatment. The stage was tested over a
range of IGV and stator setting angles to include establishing its install
characteristics for off-design stage matching studies (ref. 59). The next
step was to establish the second and third stage performance.

NASA FIVE-STAGE RESEARCH COMPRESSOR
ORIGINAL

### FIRST STAGE PERFORMANCE

NASA FIVE-STAGE RESEARCH COMPRESSOR
ORIGINAL DESIGN

SECOND AND THIRD STAGE PERFORMANCE

Shown here is the performance obtained from the second and third stage of the
original NASA five-stage compressor. Both pressure ratio and efficiency
remained well below design.

NASA FIVE – STAGE RESEARCH COMPRESSOR
ORIGINAL

SECOND AND THIRD STAGE PERFORMANCE
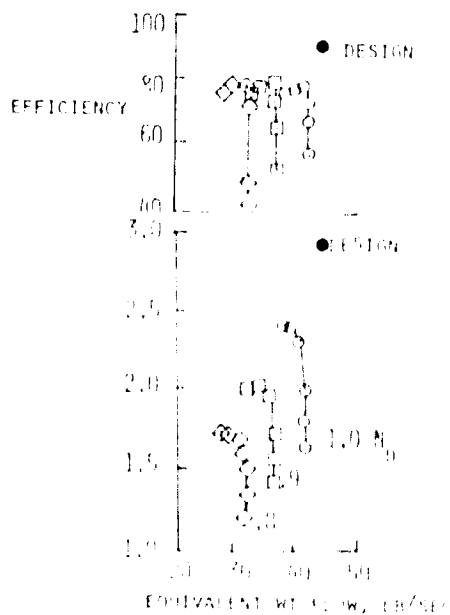
NASA FIVE-STAGE RESEARCH COMPRESSOR
ORIGINAL DESIGN

In examining all of the data obtained on the original NASA five-stage design along with the data that had been obtained from single-stage studies, it was concluded that the original five-stage design was deficient in several respects. First, the aspect ratio of the blading tended to be too high for the aerodynamic loading. This was aggravated by the unloading of the front and rear stages which resulted in very highly loaded middle stages. New incidence angle, deviation angle and loss data obtained from the single stage program indicated a strong need for adjustments in these parameters. The blockage allowances for the rear stages were considered to be too low. The velocity decrease through the machine further aggravated the blade loading problem. With these and other factors taken into consideration, the decision was made to accept the performance of the first stage with casing treatment and the slightly lower flow associated with its peak efficiency point, and to redesign the second through the fifth stage to include changing the flowpath. Blade and vane chords were maintained. This was done to permit resetting and recambering the existing blading by the coining process. This was a compromised position since the blade/vane aspect ratio was maintained at the relative high level. However, to ease the loading problem, a new loading distribution was incorporated and the axial velocity was not reduced through the compressor. New incidence and deviation angles were incorporated into the modified design. The overall pressure ratio was reduced to compensate for the lower first stage pressure ratio. The overall efficiency was reduced through increased losses and flow blockage was increased.
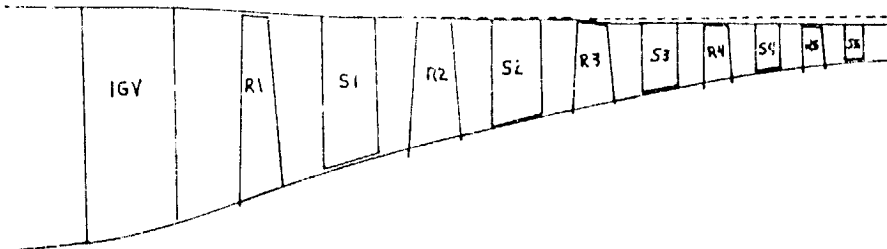
NASA FIVE-STAGE RESEARCH COMPRESSOR
MODIFIED DESIGN

Shown here is the flowpath for the modified design along with overall and stage design parameters. One can note the adjustments in the outer wall flowpath. More contraction was taken across the rear rotors than for the stators to help in unloading the rotors. The demonstrated pressure ratio of the front stage was accepted as noted earlier, and the stage pressure distribution was changed in the other stages. Stage efficiencies and thus overall efficiency was reduced.

NASA FIVE – STAGE RESEARCH COMPRESSOR

MODIFIED

STAGE					
PRESSURE RATIO	1.52	1.64	1.60	1.51	1.43
EFFICIENCY	.85	.84	.85	86	.87

OVERALL
PRESSURE RATIO 8.60
EFFICIENCY .82
WEIGHT FLOW 64.2 lb/sec

NASA FIVE-STAGE RESEARCH COMPRESSOR
MODIFIED DESIGN

Shown here are more details on the modified NASA five-stage compressor design. One can note the changes in loading in terms of D-factor compared to the original design.

NASA FIVE–STAGE RESEARCH COMPRESSOR

MODIFIED

ROTOR					
TIP RELATIVE MACH NO	1.42	1.27	1.14	1.05	0.99
TIP DIFFUSION FACTOR	.38	.47	.50	.48	.46
ASPECT RATIO	3.61	1.93	1.44	1.33	1.25
STATOR					
HUB MACH NO	0.68	0.69	0.66	0.61	0.58
HUB DIFFUSION FACTOR	.35	.47	.47	.44	.41
ASPECT RATIO	2.54	1.82	1.71	1.80	1.62

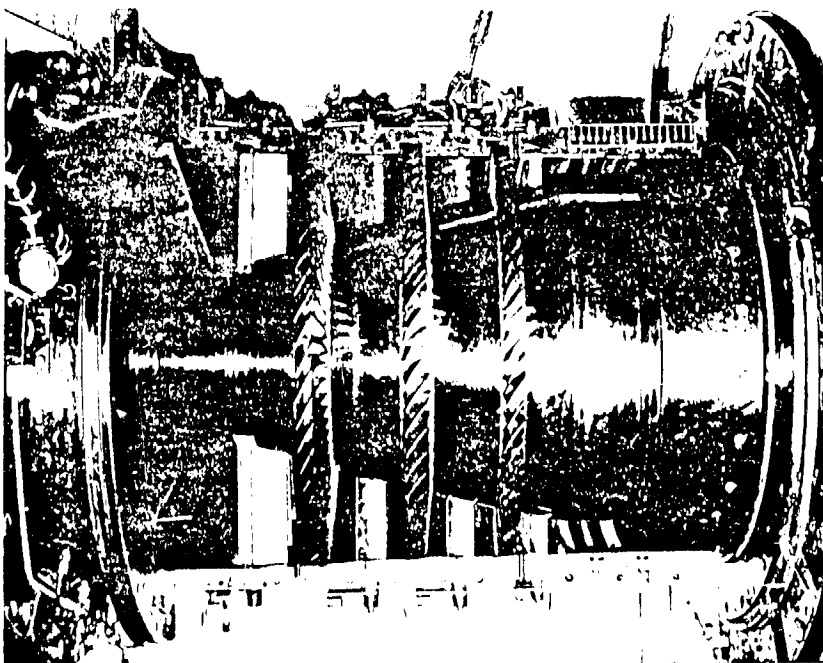### NASA FIVE-STAGE RESEARCH COMPRESSOR
MODIFIED DESIGN

Shown here is a photograph of the modified NASA Five-Stage Research Compressor
with one half of the casing removed. One can note the inserts in the outer
wall (casing) used to modify the flowpath geometry.

NASA FIVE-STAGE RESEARCH COMPRESSOR
## MODIFIED

THREE-STAGE HARDWARE

NASA FIVE-STAGE RESEARCH COMPRESSOR
MODIFIED DESIGN

THREE-STAGE PERFORMANCE

Shown here is the performance obtained on the front three stages of the
modified NASA five-stage compressor. As can be seen, the design pressure
ratio and weight flow was obtained at design speed. However, the overall
efficiency was lower than design. This, in part, was attributed to the limits
imposed by the recoining process, particularly near the blade/vane base
regions. Also, there was evidence from other programs that the extended
spacing between the blade rows for survey instrumentation was contributing to
the lower efficiencies. In addition, the relatively high blade and vane
aspect ratios even at the somewhat lower aerodynamic loadings of the modified
design were likely contributing to the low efficiency and limited flow range.

With the performance of the first three stages of the modified five-stage
compressor in hand along with even more single stage data available, a
completely new design was pursued. Its design and performance is presented in
the next several figures.

NASA FIVE-STAGE RESEARCH COMPRESSOR
## MODIFIED
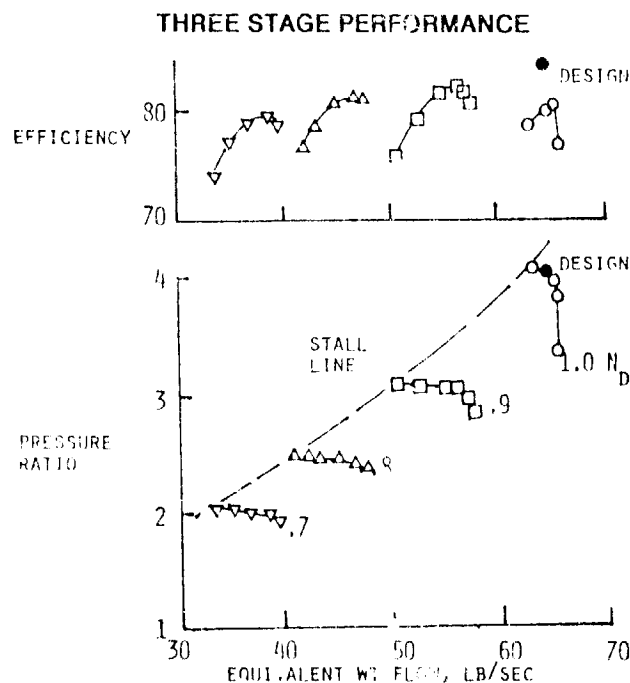
THREE STAGE PERFORMANCE

NASA FIVE-STAGE RESEARCH COMPRESSOR
LOW ASPECT RATIO DESIGN

Shown here is the flowpath for the low aspect ratio NASA Five-Stage Compressor along with overall and stage design values. Note the contraction across each rotor and little contraction across the stators. This was done to keep the velocity ratio up across the rotors. Losses were assumed on the high side in favor of stage matching and performance at part speed.

NASA FIVE – STAGE RESEARCH COMPRESSOR

LOW ASPECT RATIO

STAGE					
PRESSURE RATIO	1.74	1.65	1.58	1.48	1.40
EFFICIENCY	.82	.85	.86	.87	.88

IGV   RI   SI   R2   S2   R3   S3   R4   S4   R5   S5
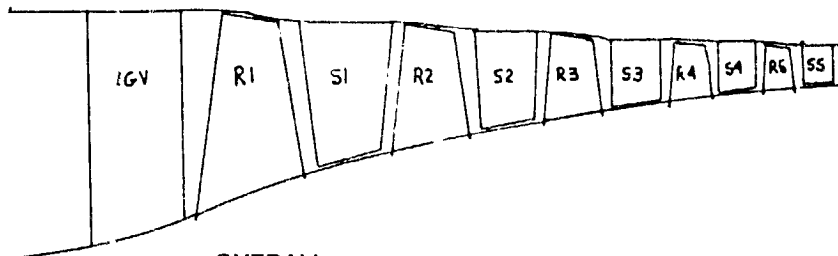
OVERALL

PRESSURE RATIO 9.27

EFFICIENCY .80

WEIGHT FLOW 65.5 lb/sec

Shown here are additional rotor and stator design parameters. Note the lower blade/vane aspect ratios.

NASA FIVE – STAGE RESEARCH COMPRESSOR

LOW ASPECT RATIO

ROTOR					
TIP RELATIVE MACH NO	1.41	1.22	1.07	0.99	0.93
TIP DIFFUSION FACTOR	.47	.52	.54	.53	.52
ASPECT RATIO	1.45	1.17	1.04	1.01	1.02
STATOR					
HUB MACH NO	0.84	0.72	0.66	0.59	0.54
HUB DIFFUSION FACTOR	.56	.54	.54	.53	.54
ASPECT RATIO	1.53	1.43	1.19	1.15	1.11

NASA FIVE-STAGE RESEARCH COMPRESSOR
LOW ASPECT RATIO DESIGN

Shown here are the blockages and energy addition assumed for the low aspect
ratio design. Note the blockage was increased from approximately four percent
at the inlet to approximately 14 percent at the exit. The energy addition was
slightly higher for the middle stages.

NASA FIVE-STAGE RESEARCH COMPRESSOR
LOW ASPECT RATIO

DESIGN FACTORS

BLOCKAGE

ENERGY SPLIT
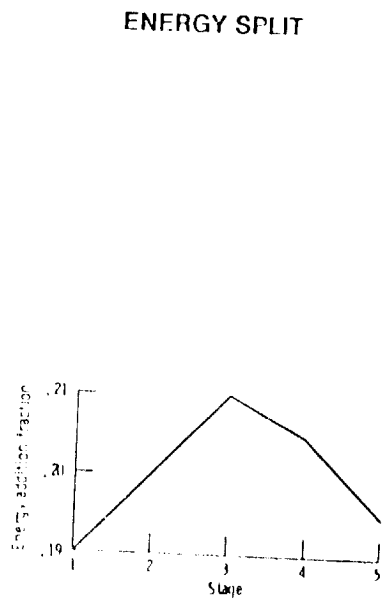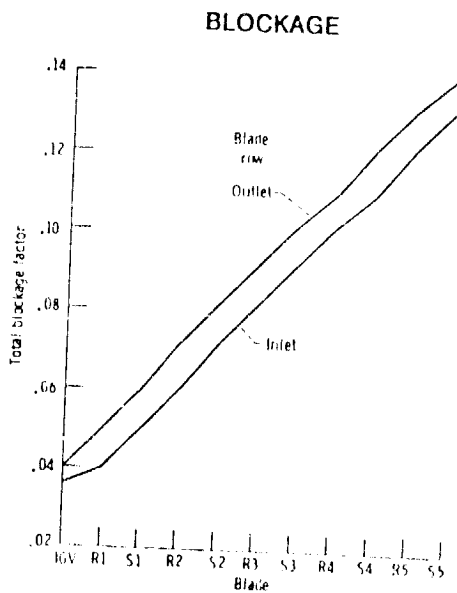
NASA FIVE-STAGE RESEARCH COMPRESSOR
LOW ASPECT RATIO DESIGN

Shown here are the loss correlations used in the design. The shock loss used
a simple correlation as a function of inlet relative Mach number. The profile
losses were based on single stage results. The measured endwall losses were
tempered somewhat to partly account for mixing effects.

NASA FIVE – STAGE RESEARCH COMPRESSOR
LOW ASPECT RATIO

LOSS PARAMETER CORRELATIONS

86
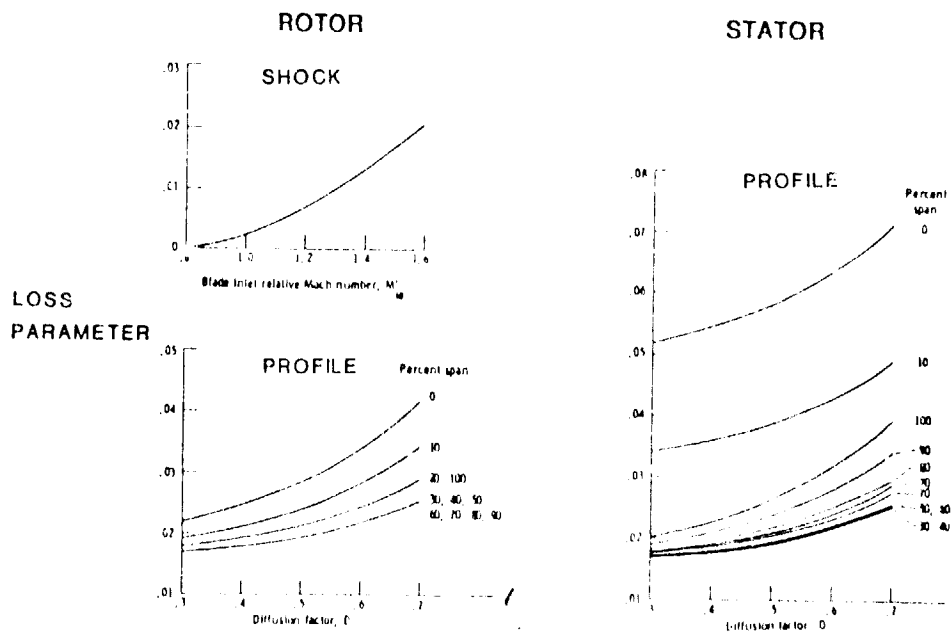
NASA FIVE-STAGE RESEARCH COMPRESSOR
LOW ASPECT RATIO DESIGN

Shown here are the deviation angle adjustments to Carter's Rule used in the design to reflect observed endwall effects on this parameter.

NASA FIVE-STAGE RESEARCH COMPRESSOR
LOW ASPECT RATIO

DEVIATION ADJUSTMENTS TO CARTER'S RULE
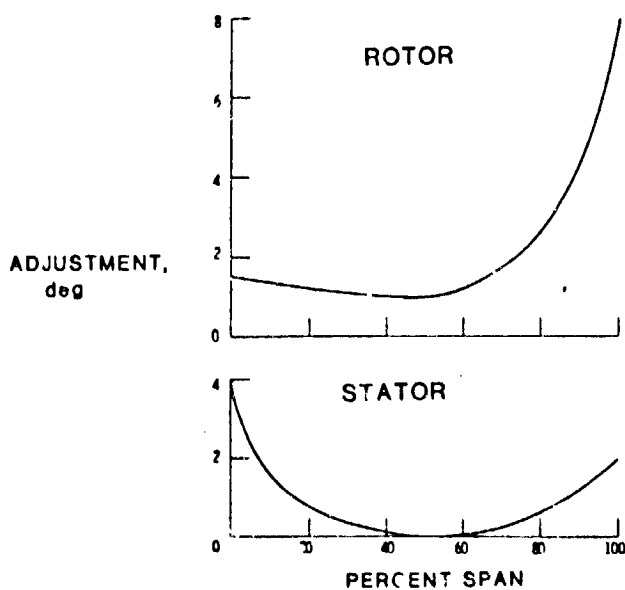
87

Shown here is a photograph of the first three-stages of NASA Five-Stage
Research Compressor incorporating low aspect ratio rotors and stators. Note
the blade rows are close coupled. Dampers are not required on the first rotor
because of the low aspect ratio.

NASA FIVE-STAGE RESEARCH COMPRESSOR
LOW ASPECT RATIO
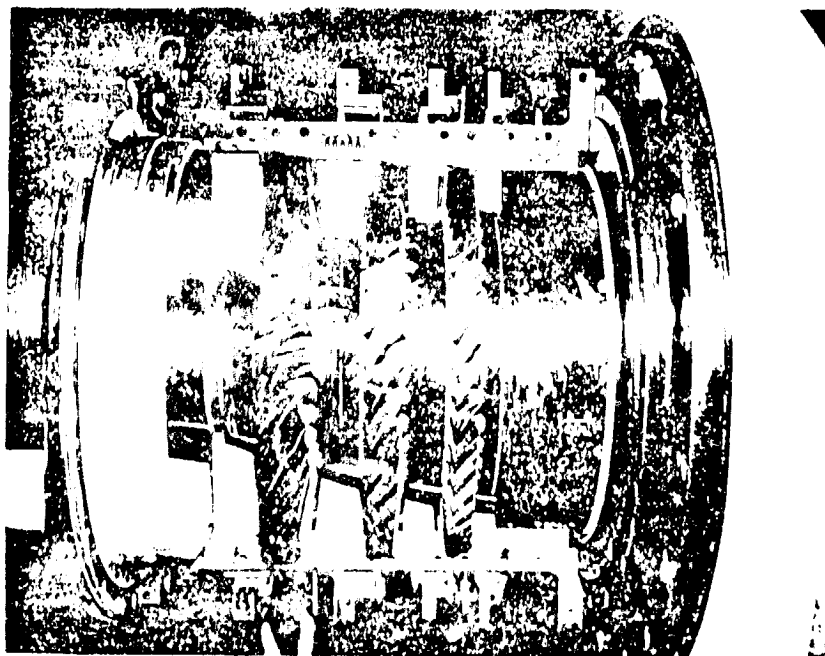
FIRST STAGE HARDWARE

NASA FIVE-STAGE RESEARCH COMPRESSOR
LOW ASPECT RATIO DESIGN

THREE STAGE PERFORMANCE

Shown here is the performance of the first three stages of the low aspect ratio NASA Five-Stage Compressor. The data was obtained at design IGV and stator setting angles. Excellent part speed performance was obtained. Some falloff in efficiency occurred at design speed and was expected. This is because the design efficiency was established as mentioned earlier to favor part speed performance. It was anticipated that improvements in design speed performance could be achieved through rematching of the stages using variable geometry.

NASA FIVE – STAGE RESEARCH COMPRESSOR
LOW ASPECT RATIO
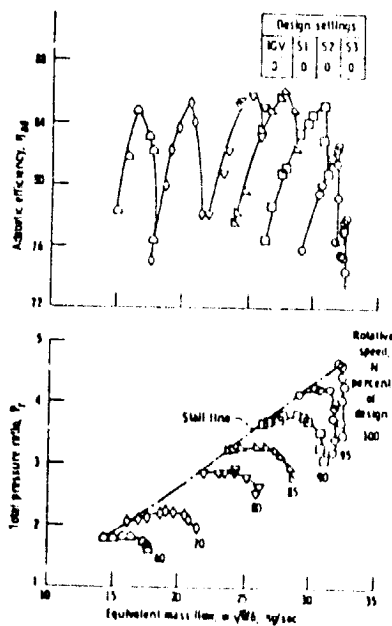
THREE STAGE PERFORMANCE

89

NASA FIVE-STAGE RESEARCH COMPRESSOR
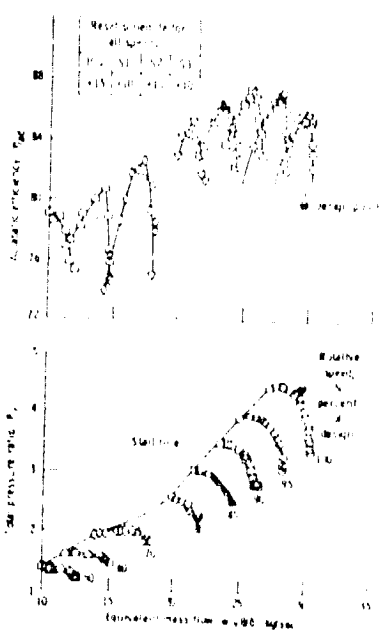## LOW ASPECT RATIO DESIGN

THREE STAGE PERFORMANCE

Shown here is the performance for the first three stages of the low aspect ratio NASA five-stage compressor: (1) with vanes scheduled to maximize performance at design speed and maintained at that setting at part speed and (2) for the vanes set for optimum performance (maximum efficiency at each speed). Note the severe falloff in efficiency at part speed when the vanes were maintained at the design speed optimum setting. Also note the improved performance at design speed for optimum setting compared to design. Performance with the vanes optimized for each speed is considered excellent. This compressor is now being used to study stage matching, install performance of highly loaded, high tip speed compressors and aeroelastic responses due to rotor stator interactions.

Obviously, the data base was not in place when we started such an aggressive design. However, we learned as much from the failures as we did from the successes along the way. Many lessons were learned in the process.

NASA FIVE–STAGE RESEARCH COMPRESSOR
LOW ASPECT RATIO

THREE STAGE PERFORMANCE
VANE SCHEDULE EFFECTS
DESIGN SPEED SCHEDULE   OPTIMUM SCHEDULE
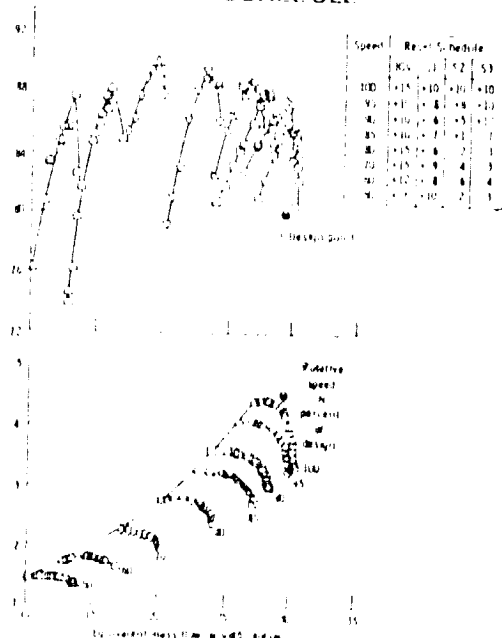
SUMMARY

The design of modern day transonic multistage compressors is highly
sophisticated and depends on an extensive empirical data base as well as
the judicious judgment of the design engineer. As computational methods
are developed and validated, the move towards a first principle system will
occur. Advanced computational codes are now being used to analyze and guide
transonic multistage designs.

Performance in terms of efficiency is considered to be approaching a physical
maximum for large flow size machines. However, further increases in overall
pressure ratio and a reduction in the number of stages required to achieve a
given pressure ratio are anticipated.

Future advancements in the design of multistage transonic compressors will
depend heavily on advanced computer technology, computational methods and more
detailed internal flow measurements.

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