| UNCLASSIFIED |  |
| AD NUMBER |  |
| AD508838 |  |
| CLASSIFICATION CHANGES |  |
| TO: | UNCLASSIFIED |  |
| FROM: | CONFIDENTIAL |  |
| LIMITATION CHANGES |  |
| TO: |  |
| Approved for public release; distribution is unlimited. |  |
| FROM: |  |
| Distribution authorized to DoD only; Administrative/Operational Use; FEB 1970. Other requests shall be referred to Army Aviation Materiel Lab., Fort Eustis, VA. |  |
| AUTHORITY |  |
| USAAMRLD ltr 21 May 1973; USAAMRLD ltr 21 May 1973 |  |

THIS PAGE IS UNCLASSIFIED
SECURITY
MARKING

The classified or limited status of this report applies to each page, unless otherwise marked. Separate page printouts MUST be marked accordingly.

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C., SECTIONS 793 AND 794. THE TRANSMISSION OR THE REVELATION OF ITS CONTENTS IN ANY MANNER TO AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW.

NOTICE: When government or other drawings, specifications or other data are used for any purpose other than in connection with a definitely related government procurement operation, the U.S. Government thereby, inures no responsibility, nor any obligation whatsoever; and the fact that the Government may have formulated, furnished, or in any way have the said drawings, specifications, or other data is not to be construed by implication or otherwise as in any manner licensing the rights or any other person or corporation, or conveying any rights of production to manufacture, use or sell any patented invention that may in any way be related thereto.
DISCLAIMER NOTICE

THIS DOCUMENT IS BEST QUALITY AVAILABLE. THE COPY FURNISHED TO DTIC CONTAINED A SIGNIFICANT NUMBER OF PAGES WHICH DO NOT REPRODUCE LEGIBLY.
Disclaimers

The findings in this report are not to be construed as an official Department of the Army position unless so designated by other authorized documents.

When Government drawings, specifications, or other data are used for any purpose other than in connection with a definitely related Government procurement operation, the United States Government thereby incurs no responsibility nor any obligation whatsoever; and the fact that the Government may have formulated, furnished, or in any way supplied the said drawings, specifications, or other data is not to be regarded by implication or otherwise as in any manner licensing the holder or any other person or corporation, or conveying any rights or permission, to manufacture, use, or sell any patented invention that may in any way be related thereto.

Disposition Instructions

When this report is no longer needed, Department of the Army organizations will destroy it in accordance with the procedures given in AR 380-5.
The research described herein, which was conducted by Continental Aviation and Engineering Corporation, was performed under U.S. Army Contract DA-177-AMC-296(T). The work was performed under the technical management of Mr. David B. Cale, Propulsion Division, U.S. Army Aviation Materiel Laboratories.

This document is the classified addendum to USAAVLABS Technical Report 69-108.
ADVANCEMENT OF SMALL GAS TURBINE COMPONENT TECHNOLOGY (U)

ADVANCED SMALL AXIAL COMPRESSOR (U)

VOLUME II - ADDENDUM
AERODYNAMIC REDESIGN (U)

Continental Report 1033

By
James V. Davis
Edmund J. Dellert

Prepared By
Continental Aviation and Engineering Corporation
Detroit, Michigan

for
U. S. ARMY AVIATION MATERIEL LABORATORIES
Fort Eustis, Virginia

Downgraded at 3 year intervals;
declassified after 12 years.
DOD DIR 5200.10
SUMMARY

The two-stage axial compressor was redesigned using test data from the previous two stage tests and from an analysis of a family of high-pressure-ratio axial compressor rotors. The major changes made to the original design were reduced solidity in both rotors, increased first-stage rotor aspect ratio, and modified first-stage rotor blade profile.

The redesigned axial compressor had the following design point performance goals:

- Airflow = 5 lbs/sec
- Pressure ratio = 3.0:1
- Adiabatic efficiency = 82.6 percent
- Inlet hub tip ratio = 0.51
- First-stage rotor tip speed = 1448 ft/sec
This report presents the redesign analysis of the axial compressor program for the advancement of small gas turbine component technology. The redesign analysis is presented as an addendum to Volume II.

This program was sponsored by the United States Army Aviation Materiel Laboratories under Contract DA 44-177-AMC-296(T), Task 1G162203D14413.

The compressor program was divided into three phases. Phase I presents a study of a family of advanced axial compressors. It is reported as Volume I, with an addendum under a separate cover reporting the analysis and design.

Phases II and III are presented in Volume II. Phase II presents the axial compressor fabrication and test. Phase III presents the axial compressor redesign, fabrication and test.
# TABLE OF CONTENTS

<table>
<thead>
<tr>
<th>Section</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>SUMMARY</td>
<td>iii</td>
</tr>
<tr>
<td>FOREWORD</td>
<td>v</td>
</tr>
<tr>
<td>LIST OF ILLUSTRATIONS</td>
<td>viii</td>
</tr>
<tr>
<td>LIST OF TABLES</td>
<td>viii</td>
</tr>
<tr>
<td>DETAILED AERODYNAMIC REDESIGN</td>
<td>1</td>
</tr>
<tr>
<td>DISTRIBUTION</td>
<td>15</td>
</tr>
</tbody>
</table>
(U) LIST OF ILLUSTRATIONS

Figure
1  Axial Compressor Rotor .......................... 2
2  Comparison of USAAVLABS First-Stage Flow Paths .......................... 3
3  First Stage Rotor, Loss Coefficient Comparison .......................... 6
4  First Stage Rotor, Diffusion Factor Comparison .................................. 7
5  First Stage Rotor, Relative Velocity Comparison .................................. 7
6  First Stage Rotor, Absolute Flow Angle Comparison .................................. 8
7  First Stage Rotor, Performance Comparison .................................. 9
8  First Stage Stator, Loss Coefficient Comparison .................................. 10
9  Comparison of Compressor Exit Axial Velocity Profile .................................. 10
10 First Stage Rotor, Velocity Triangles .................................. 12

(U) LIST OF TABLES

Table
1 Redesigned Compressor Performance Comparison 4
2 Flow Blockage Factors .................................. 11
3 USAAVLABS First-Stage Rotor Redesign Geometry, Double Circular Arc Blading 14
The compressor rotor under study in this program is shown in Figure 1. The redesign of the compressor rotor, first-stage was initiated based on the analysis of the family of high-pressure-ratio axial compressor rotors using the following ground rules:

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Range</th>
</tr>
</thead>
<tbody>
<tr>
<td>Tip Solidity</td>
<td>1.3 to 1.45</td>
</tr>
<tr>
<td>Aspect Ratio</td>
<td>Greater than 0.8</td>
</tr>
<tr>
<td>Free Vortex Diffusion Factor</td>
<td>Less than 0.5</td>
</tr>
</tbody>
</table>

Five redesign flowpaths were generated. Each flowpath was constructed to cover the range of hub and tip contours capable of matching with the existing second-stage flowpath contour. The aerodynamic parameters resulting from computer runs for each flowpath were analyzed and compared to provide direction to the final redesign flowpath. The optimum flowpath, Figure 2, was selected on the basis of rotor tip relative Mach number, tip turning, tip loading, and stator hub absolute Mach number. The rotor losses used in this study were obtained from the high-pressure-ratio axial compressor analysis.

Using the optimum flowpath, the first-stage rotor exit pressure profile was varied to determine the effect of this parameter on the second-stage rotor incidence angles and the compressor exit velocity gradient. From this analysis, it was determined that a flat first-stage rotor exit stagnation pressure profile provided the best second-stage incidence match and compressor exit velocity profile.

Throughout the redesign analysis, the Continental axial compressor aerodynamic design computer program was used to generate the static pressures and the corresponding velocities. This program included the effects of streamline curvatures and calculated the boundary layer growths based on flat-plate theory.

Table I compares the redesign compressor performance with the original design values. A 1-percent transition duct loss, which was verified in Phase I testing, was used to determine the overall performance at the end of the transition duct. The differences in first-stage and overall efficiency between the original design and the redesign are directly related to the reevaluation of first-stage losses. In addition to redesign of the first stage, the aerodynamic implications of second-stage solidity were investigated in great depth. Significant
### Table 1 (U)

<table>
<thead>
<tr>
<th>Description</th>
<th>Original Design</th>
<th>First Stage Redesign</th>
<th>First Stage Redesign With Lower Solidity</th>
<th>Second Stage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Compressor Rotational Speed, rpm</td>
<td>60,000</td>
<td>60,000</td>
<td>60,000</td>
<td></td>
</tr>
<tr>
<td>Compressor Flow Rate, lb/sec</td>
<td>5.0</td>
<td>5.0</td>
<td>5.0</td>
<td></td>
</tr>
<tr>
<td>Overall pressure ratio</td>
<td>4.0:1</td>
<td>4.0:1</td>
<td>4.0:1</td>
<td></td>
</tr>
<tr>
<td>Overall efficiency, %</td>
<td>82.3</td>
<td>81.1</td>
<td>82.6</td>
<td></td>
</tr>
<tr>
<td>Pressure Ratio - first stage</td>
<td>1.825:1</td>
<td>1.825:1</td>
<td>1.825:1</td>
<td></td>
</tr>
<tr>
<td>Pressure Ratio - second stage</td>
<td>1.660:1</td>
<td>1.660:1</td>
<td>1.660:1</td>
<td></td>
</tr>
<tr>
<td>Efficiency - first stage, %</td>
<td>85.5</td>
<td>83.9</td>
<td>83.9</td>
<td></td>
</tr>
<tr>
<td>Efficiency - second stage, %</td>
<td>82.9</td>
<td>82.9</td>
<td>86.1</td>
<td></td>
</tr>
<tr>
<td>Tip Speed - first rotor, ft/sec</td>
<td>1415</td>
<td>1451</td>
<td>1451</td>
<td></td>
</tr>
</tbody>
</table>

Performance at transition duct exit

Performance increases were anticipated with a reduced solidity second stage (Table 1) due to reductions in loss coefficient and increases in efficient flow range.

The second-stage solidity analysis was conducted using the Continental double circular arc blade loss system. This loss system, which was recently placed in operation, predicts losses based mainly on diffusion and critical Mach number criteria. This method has successfully predicted performance of other compressors in the Mach number and diffusion range of the USAAVLABS second stage.
(U) As a consequence of the second-stage solidity analysis, the number of blades in the second stage was reduced from 41 to 35, a solidity reduction of 19.5 percent. The blade contour of the original second stage was maintained.

(U) The losses used for the redesigned first-stage rotor, Figure 3, as compared to the original values, show a significant increase at the tip of the rotor and a slight decrease from the hub to about 80-percent span. This trend was observed in the test data for most of the rotors investigated in the high-pressure-ratio axial compressor study.

(C) The loss variation along the blade height significantly changes the local tip loading, as shown in a comparison with diffusion factors and relative velocity ratio, Figures 4 and 5, respectively. The value of tip diffusion factor for the redesigned rotor (0.56) appears to be high but has been demonstrated and exceeded on previous Continental high-pressure ratio rotors. For example, the first-stage rotor of Continental's 8:1 pressure ratio compressor has demonstrated diffusion factors in excess of 0.65 at comparable tip pressure ratio levels.

(C) The free vortex diffusion factor (the diffusion factor resulting from an assumption of constant spanwise pressure ratio and constant spanwise energy addition) is 0.423 as compared to 0.39 for the original design.

(C) Because of the increased redesign tip loss, the tip absolute flow angle into the stator increases from the original design value of 42 degrees to about 50 degrees (Figure 6). To match the tip of the stator to this higher angle, the stator was closed 8.5 degrees at the tip to zero degrees at a 2.53-inch radius (80-percent length).

(U) The redesigned first-stage pressure ratio and efficiency profiles are presented in Figure 7. The reduced efficiency at the tip is a result of the additional tip losses shown in Figure 3. The slight variation in stage pressure ratio is due to a reevaluation of first-stage stator losses (Figure 8), which was necessary because of the change in stator absolute angles.

(U) A comparison of the redesigned and original design exit velocity gradient, Figure 9, shows only minor change. The redesigned gradient should provide a satisfactory transition duct exit velocity profile.
Figure 3. (U) First-Stage Rotor, Loss Coefficient Comparison.
Figure 4. (U) First-Stage Rotor, Diffusion Factor Comparison.

Figure 5. (U) First-Stage Rotor, Relative Velocity Comparison.
The redesigned flow path, Figure 2, embodies a higher hub and higher tip radius at the inlet to the first stage. No change in flow path contour was made downstream of the second rotor inlet. Increasing the aspect ratio from 0.646 to 0.963 required a higher first-stage inlet tip radius to provide a smooth flow path transition between the first and second stage and to hold the tip relative inlet air angle between 65 and 70 degrees. Higher angles tend to close the rotor and narrow the rotor choke margin (the ratio of actual blade throat to the flow limiting throat, that is, sonic conditions), so that excessive incidence angles are required to pass the desired flow. Lower angles tend to raise the tip relative Mach number and in turn increase tip losses.

As shown in Figure 2, an accelerating tip flow path turn between the first-stage rotor and stator has been provided to reduce the tip loading. The accelerating turn tends to locally increase meridional velocities and to offset some of the velocity deceleration caused by high tip losses.

The flow blockage factors (aerodynamic flow area divided by actual area) calculated for the redesign are listed in Table II and are compared to the original design values.
Figure 7. (U) First-Stage Rotor, Performance Comparison.
Figure 8. (U) First-Stage Stator, Loss Coefficient Comparison.

Figure 9. (U) Comparison of Compressor Exit Axial Velocity Profile.
(C) TABLE II (U)

FLOW BLOCKAGE FACTORS (U)

<table>
<thead>
<tr>
<th>Axial Station</th>
<th>Original Design Blockage</th>
<th>Redesign Blockage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inlet to Rotor 1</td>
<td>0.99</td>
<td>0.991</td>
</tr>
<tr>
<td>Exit of Rotor 1</td>
<td>0.98</td>
<td>0.973</td>
</tr>
<tr>
<td>Exit of Stator 1</td>
<td>0.97</td>
<td>0.972</td>
</tr>
<tr>
<td>Exit of Rotor 2</td>
<td>0.97</td>
<td>0.954</td>
</tr>
<tr>
<td>Exit of Stator 2</td>
<td>0.97</td>
<td>0.953</td>
</tr>
</tbody>
</table>

Good agreement is shown (maximum deviation of 1.7 percent) between the blockage values of the original design and the redesign.

The velocity triangles, which define the aerodynamics of the redesigned first-stage rotor, are presented in Figure 10. The seven velocity triangles are located at selected streamlines that enclose the percent total flow values from the hub as defined below:

<table>
<thead>
<tr>
<th>Streamline</th>
<th>Total Flow Value From Hub (Percent)</th>
</tr>
</thead>
<tbody>
<tr>
<td>7</td>
<td>100.0</td>
</tr>
<tr>
<td>6</td>
<td>83.3</td>
</tr>
<tr>
<td>5</td>
<td>66.7</td>
</tr>
<tr>
<td>4</td>
<td>50.0</td>
</tr>
<tr>
<td>3</td>
<td>33.3</td>
</tr>
<tr>
<td>2</td>
<td>16.7</td>
</tr>
<tr>
<td>1</td>
<td>0.0</td>
</tr>
</tbody>
</table>

The blade geometry summary is presented in Table III. Double circular arc blading was selected on the basis of flow range and successful experience with other Continental high-pressure-ratio axial compressors.
Figure 10. (C) First-Stage Rotor, Velocity Triangles. (U)
Figure 10. (C) Continued. (U)
### USAAVLABS FIRST STAGE ROTOR REDESIGN GEOMETRY:
**DOUBLE CIRCULAR ARC BLADING**

<table>
<thead>
<tr>
<th>Radius</th>
<th>Chord</th>
<th>Solidity</th>
<th>Thickness Suction to Chord Surface Ratio</th>
<th>Pressure Surface Radius (In.)</th>
<th>Axial Length (In.)</th>
</tr>
</thead>
<tbody>
<tr>
<td>2.79</td>
<td>1.23</td>
<td>1.403</td>
<td>0.03</td>
<td>6.357</td>
<td>339.71</td>
</tr>
<tr>
<td>2.50</td>
<td>1.23</td>
<td>1.566</td>
<td>0.0367</td>
<td>3.898</td>
<td>16.89</td>
</tr>
<tr>
<td>2.20</td>
<td>1.23</td>
<td>1.779</td>
<td>0.0445</td>
<td>2.412</td>
<td>5.85</td>
</tr>
<tr>
<td>1.90</td>
<td>1.23</td>
<td>2.060</td>
<td>0.054</td>
<td>1.608</td>
<td>3.03</td>
</tr>
<tr>
<td>1.65</td>
<td>1.23</td>
<td>2.373</td>
<td>0.065</td>
<td>1.161</td>
<td>1.884</td>
</tr>
</tbody>
</table>

**L. E. R. and T. E. R.**

<table>
<thead>
<tr>
<th>To Chord Ratio</th>
<th>Inlet Metal Angle (Deg)</th>
<th>Exit Metal Angle (Deg)</th>
<th>Camber Angle (Deg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.00326</td>
<td>62.76</td>
<td>57.14</td>
<td>5.62</td>
</tr>
<tr>
<td>0.00326</td>
<td>59.52</td>
<td>48.42</td>
<td>11.09</td>
</tr>
<tr>
<td>0.00326</td>
<td>56.18</td>
<td>35.49</td>
<td>20.68</td>
</tr>
<tr>
<td>0.00326</td>
<td>52.80</td>
<td>18.77</td>
<td>34.02</td>
</tr>
<tr>
<td>0.00326</td>
<td>49.23</td>
<td>-1.594</td>
<td>50.82</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Turning Angle (Deg)</th>
<th>Deviation Angle (Deg)</th>
<th>Incidence Angle (Deg)</th>
<th>Setting Angle (Deg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>5.35</td>
<td>3.21</td>
<td>2.94</td>
<td>59.95</td>
</tr>
<tr>
<td>9.80</td>
<td>4.67</td>
<td>3.38</td>
<td>53.97</td>
</tr>
<tr>
<td>18.01</td>
<td>6.51</td>
<td>3.83</td>
<td>45.83</td>
</tr>
<tr>
<td>30.35</td>
<td>7.97</td>
<td>4.30</td>
<td>35.79</td>
</tr>
<tr>
<td>47.50</td>
<td>7.99</td>
<td>4.67</td>
<td>23.82</td>
</tr>
</tbody>
</table>

*L. E. R. - Leading edge radius
T. E. R. - Trailing edge radius*
10 ABSTRACT
This report presents the redesign analysis of the axial compressor program for the advancement of small gas turbine component technology. (U)

A two-stage axial compressor was redesigned using test data from the previous two stage tests together with an analysis of a family of high-pressure-ratio compressor rotors. The major changes made to the original design were reduced solidity in both rotors, increased first-stage rotor aspect ratio, and modified first-stage rotor blade profile. (U)

The redesigned axial compressor had the following design performance goals:

- Airflow = 5 lbs/sec
- Pressure ratio = 3.0:1
- Adiabatic efficiency = 82.6 pct
- Inlet hub tip ratio = 0.51
- First-stage rotor tip speed = 1448 ft/sec (U)
<table>
<thead>
<tr>
<th>KEY WORDS</th>
<th>LINE A</th>
<th>LINE B</th>
<th>LINE C</th>
<th>LINE D</th>
<th>LINE E</th>
</tr>
</thead>
<tbody>
<tr>
<td>Axial Compressor Design</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Component Technology</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Small Gas Turbine</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>