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AN EXPERIMENTAL EVALUATION OF THE PERFORMANCE DEFICIT OF AN AIRCRAFT ENGINE STARTER TURBINE

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ABSTRACT

An experimental investigation was made at NASA-Lewis Research Center to determine the reasons for the low aerodynamic performance of a 13.5-centimeter-tip-diameter aircraft engine starter turbine. The investigation consisted of an evaluation of both the stator and the stage. An approximate ten percent improvement in turbine efficiency was obtained when the honeycomb shroud over the rotor blade tips was filled to obtain a solid shroud surface.

THE NASA LEWIS RESEARCH CENTER was requested by the Department of the Navy to conduct an experimental performance investigation of a 13.5-centimeter tip diameter aircraft engine starter turbine. Experimental evaluation of this turbine by the manufacturer indicated an efficiency that was approximately ten points lower than design. With this low performance it was believed that this starter turbine could not start the aircraft engine within the time period specified for the aircraft. The experimental program at Lewis Research Center was intended to determine the reasons for the turbine performance deficit.

A design review of the turbine at Lewis Research Center identified two features as potential contributors to the observed efficiency deficit. One of these features was the use of a manifold having a single inlet and designed to turn the flow 180° into the stator. In this kind of manifold design, the flow enters a single feed pipe, branches, and then flows down both sides of the manifold, resulting in large and variable stator incidence angles and possibly radial and circumferential gradients in mass flow. The second design feature was the use of a honeycomb shroud surface over the rotor blade tips. The honeycomb was used because it permitted the use of a lower rotor tip clearance and gave protection to the rotor in case of contact with the shroud.

The experimental evaluation consisted of two parts. The first part consisted of bellmouth inlet and stator exit surveys to determine whether the manifold significantly affected stator performance. The second part was a turbine stage evaluation. Three honeycomb shroud configurations were investigated in the stage evaluation: the original honeycomb shroud, a filled honeycomb shroud that produced a smooth continuous shroud surface, and a honeycomb shroud that had a cell depth that was half the original depth.

This paper summarizes the results of the experimental investigation (Ref.(1))* to evaluate the performance deficit of the starter turbine. Presented are survey results obtained

*Numbers in parentheses designate References at the end of paper.

at the bellmouth inlet and stator exit and stage performance results. The effect on turbine stage performance of varying the honeycomb depth is also discussed.

SYMBOLS

P	absolute pressure, N/cm^2
R_x	blade mean reaction, $\frac{P_{2m} - P_{3m}}{P_1' - P_{3m}}$
r	radius, m
T	absolute temperature, K
U	blade velocity, m/sec
V	absolute gas velocity, m/sec
ΔVu	change in absolute tangential velocity, m/sec
W	relative gas velocity, m/sec
w	mass flow, Kg/sec
α	absolute gas flow angle measured from axial direction, deg
β	relative gas flow angle measured from axial direction, deg
η_{1-3}'	total efficiency based on bellmouth inlet-to-rotor exit total pressure ratio
η_{1-3}	static efficiency based on bellmouth inlet-to-rotor exit static pressure ratio
$\Delta\eta_{1-3}'$	performance loss
μ	viscosity, (N) (sec)/m^2
ν_{1-3}	blade-jet speed ratio based on bellmouth inlet total-to-rotor exit static pressure ratio

SUBSCRIPTS-

cr	condition corresponding to Mach number of unity
m	mean section
0	station at manifold inlet (Fig. 6)
1	station at bellmouth inlet (Fig. 6)
2	station at stator exit (Fig. 6)
3	station at rotor exit (Fig. 6)

SUPERSCRIPTS-

'	absolute total condition

TURBINE DESIGN

The aircraft starter turbine tested in this investigation was a single-stage, axial-flow design having a tip diameter of 13.5 centimeters. A cross-section schematic of the turbine is shown in Fig. 1. Flow entered the manifold at a critical velocity ratio of about 0.25, turned 90°, split, and then turned another 90° into the stator. A short bellmouth inlet was used ahead of the stator to provide minimum inlet endwall boundary layers. The stator vanes were untapered and untwisted. The rotor blade design was also untwisted but was tapered. A honeycomb material was used over the rotor blades to protect the blades in case of a rotor rub. With this added protection, a smaller rotor tip clearance equal to about one percent of the rotor blade height could be used. The turbine exit section was designed to diffuse the flow.

A list of the actual design conditions, the equivalent design conditions, and the test conditions for this turbine are shown in Table 1. The turbine was designed for an efficiency of 0.88 at a work factor of 1.47. Table 2 lists some of the physical parameters for this turbine.

The mean-section velocity diagram is shown in Fig. 2. The stator discharge angle was 73°, whereas the rotor was designed for zero exit swirl. The mean rotor reaction, R_x, was .21.

Figures 3 and 4 are photographs of the manifold and stator assembly and the rotor assembly, respectively. Three honeycomb shrouds were used in the stage evaluation. The original honeycomb shroud, denoted as the open honeycomb, had a cell diameter of 0.16 centimeters and a cell depth of 0.38 centimeters. The second honeycomb shroud, denoted as the half depth honeycomb, had the same cell diameter but a cell depth of only 0.19 centimeters. The open honeycomb shroud was filled to produce a solid shroud surface configuration, which was denoted as the filled honeycomb. The purpose of testing these three shroud configurations was to quantify the variation of turbine efficiency with honeycomb depth.

RESEARCH EQUIPMENT AND PROCEDURE

The apparatus used in this investigation consisted of the research turbine, an airbrake dynamometer used to control the speed and absorb and measure the power output of the turbine, an inlet and exhaust piping system including flow controls, and appropriate instrumentation. A schematic of the experimental equipment is shown in Fig. 5. The rotational speed of the turbine was measured with an electronic counter in conjunction with a magnetic pickup and a shaft-mounted gear. Mass flow was measured with a calibrated venturi. Turbine torque was determined by measuring the reaction torque of the airbrake which was mounted on air trunion bearings, and adding correction for tare losses. The torque load was measured with a commercial strain-gage load cell.

The turbine instrumentation stations are shown in Fig. 6. Instrumentation at the manifold inlet (station 0) measured wall static pressure and total temperature. Static pressures were obtained from two taps located 180° apart. Total temperature was measured with three thermocouple probes that were located at approximately 10, 50, and 90 percent of the pipe diameter.

At the bellmouth inlet (station 1), total pressure and flow angle were measured at four different circumferential locations using a survey probe. At each circumferential location, data were obtained at several radial positions ranging from about -7 to 115 percent of the stator blade height. At each position the survey probe remained fixed and the flow angle and total pressure were obtained from probe measurements and calibration curves.

At the stator exit (station 2), located less than half an axial chord length downstream of the stator, static pressures were measured with 16 taps with eight each on the inner and outer walls. The inner and outer wall taps were located opposite each other at different intervals around the circumference.

In order to obtain the blading performance, the original exit diffuser duct was replaced by a constant area exhaust duct (Fig. 6). Using a constant area exhaust duct allowed the rotor exit instrumentation to be located at a position where the rotor wakes were mixed out. For this turbine, this position as determined using a hot-wire anemometer survey probe was approximately one-and-a-half axial chord lengths downstream of the rotor.

At the rotor exit (station 3) static pressure, total pressure, total temperature, and flow angle were measured. The static pressure was measured with eight taps with four each on the inner and outer walls. These inner and outer wall taps were located opposite each other at 90° intervals around the circumference. Three self-aligning probes located at three positions around the circumference were used for measurement of total pressure, total temperature, and flow angle.

The experimental program consisted of an evaluation of both the stator and the stage. The stator evaluation consisted of both bellmouth inlet and stator exit surveys. The bellmouth inlet surveys were discussed earlier. For the stator exit surveys, the rotor was removed and a survey probe was used to obtain total pressure and flow angle over a range of circumferential and radial positions. Fig. 7 shows the bellmouth inlet and stator exit survey circumferential locations. Each of the four stator exit survey sectors consisted of two consecutive stator blade spacings (31.3^{o}). In each sector, data were obtained approximately every 1.2^{o} at radial positions equal to approximately 10, 30, 50, 70, and 90 percent of the stator blade height. For each data point the probe remained fixed and the total pressure and flow angle were obtained from probe readings and calibration curves. The survey data obtained at a given radial position were then arithmetically averaged. Both the bellmouth inlet and stator exit surveys were conducted at the design manifold inlet-to-stator exit mean static pressure ratio of 2.2.

For the stage evaluation, data were obtained at nominal inlet total flow conditions of 322K and 13.8 newtons per square centimeter. The turbine Reynolds number at these conditions was about 3.7×10^5 which was considered high enough to avoid any effect of Reynolds number on performance. Data were obtained over a range of bellmouth inlet total-to-rotor exit static pressure ratio from 1.55 to 4.50 and over a range of design equivalent speed from 30 to 110 percent.

The turbine was rated on the basis of total efficiency. The actual work was calculated from torque, speed, and mass flow measurements. The rotor exit total pressure used in determining the efficiency was calculated from mass flow, static pressure, total temperature, and flow angle. The bellmouth inlet total pressure was based on an arithmetic average of the bellmouth inlet survey results.

RESULTS AND DISCUSSION

STATOR SURVEY RESULTS - As shown by the bellmouth inlet survey results in Fig. 8(a), the bellmouth-inlet-to-manifold inlet total pressure ratio was about 0.98 around the circumference. Because the flow entering the manifold impacted the elbow section before being turned 180° into the stator (Fig. 1), about 50 percent of the velocity heat was lost. In addition, the total pressure ratio decreased from the hub to the tip at each circumferential position. This trend was attributed to a probe pitch angle effect as the probe was moved radially towards the tip into a region where high radial components of flow existed. Since the survey probe was not calibrated for a pitch angle greater than 10°, the total pressure obtained from the calibration curve would have been less than the true total pressure. However, even if it were assumed that the total pressure ratio had remained constant near the tip instead of decreasing, this would have had only a small effect on the average value. Thus, the probe pitch effect was not considered detrimental.

Also indicated by Fig. 8(a) is a circumferential variation in the level of total pressure. The total pressure ratio was about 0.5 percent higher near the bottom of the manifold (locations B and C) than at the sides (locations A and D). Although this trend was unexpected, the amount of variation was considered small and no further investigation of this trend was made.

An arithmetically averaged total pressure ratio was calculated at each of the four locations surveyed and then those four averages were combined into a single overall average. The overall average was .983 which was also equal to the design value. This value of 0.983 was subsequently used in the stage evaluation to determine the bellmouth inlet total pressure from the manifold inlet total pressure.

Figure 8(b) shows a wide radial and circumferential variation in the bellmouth inlet flow angle. Since the design bellmouth inlet flow angle was 0°, it is evident that the stator experienced incidence angles ranging from -40° to 40°. The effect of this incidence angle variation would be expected to be small since the critical velocity ratio at the stator inlet was only about 0.2.

The stator exit survey results are shown in Fig. 9. Test data obtained by the manufacturer indicated a high loss area near sector E (Fig. 7). The stator exit surveys were made to determine whether this or any other major loss area existed.

As with the bellmouth inlet survey results, the stator exit surveys indicated radial and circumferential variations in total pressure and flow angle. The highest total pressure was near the bottom of the manifold (sector F), which was consistent with the bellmouth inlet results. Arithmetically averaging the total pressure ratio and flow angle data resulted in values of 0.950 and 71°. This compared to design values of 0.936 and 73°. The survey results at each circumferential position were used to calculate stator exit tangential velocity ratios. The results indicated that the radial variations in tangential velocity ratio at the four survey sectors were close to the design intent. Based on the results shown in Fig. 9, it was determined that even though there were radial and circumferential gradients in flow, no large loss areas existed in the stator.

STAGE RESULTS - Figure 10 shows the stage performance results for the three honeycomb shrouds investigated. Total efficiency, based on the bellmouth inlet-to-rotor exit total pressure ratio, is plotted against blade-jet speed ratio.

Figure 10(a) indicates that at design equivalent conditions of speed and blade-jet speed ratio, the total efficiency with the original open honeycomb shroud was 0.786, which was 9.4 points lower than the design value of 0.88.

Figure 10(b) shows the dramatic improvement in turbine efficiency that occurred when the honeycomb shroud was filled. At design equivalent conditions of speed and blade-jet speed ratio, the total efficiency was 0.868 which was 8.2 points higher than with the open honeycomb shroud. It was apparent that the effect of the open honeycomb shroud was to cause a massive loss mechanism to occur. As shown in Fig. 11, the performance loss between the open and filled honeycomb shroud configurations was not constant with rotative speed, but actually increased in a linear manner with increasing speed.

In order to obtain additional data in regard to the performance loss due to the honeycomb shroud, the half-depth shroud configuration was used. The performance results obtained with this shroud are shown in Fig. 10(c). At design equivalent conditions of blade-jet speed ratio and speed the total efficiency was 0.850.

This efficiency was 6.4 points higher than with the original honeycomb shroud and 1.8 points less than with the filled honeycomb shroud. Compared to the original honeycomb shroud, the percentage increase in efficiency with the half depth honeycomb shroud was greater than the percentage change in the honeycomb cell depth.

Rotor exit radial surveys of total temperature, total pressure, and flow angle were conducted at design equivalent values of speed and blade-jet speed ratio to determine variations in blade element performance. Three survey probes were located downstream of the rotor at three different circumferential positions. The results from the probes were then arithmetically averaged. Figure 12 shows significant differences in the radial variations of the flow parameters among the three shroud configurations. The absolute levels of turbine efficiency calculated from the survey data (Fig. 12(d)) are not as accurate as those based on the measured turbine torque, primarily because of difficulties encountered in obtaining accurate total temperature measurements due to conduction and local Mach number effects. However, the radial variations and relative levels of turbine efficiency calculated from the survey data are considered adequate to reveal significant differences in the blade element performance. These results show that the honeycomb shrouds did not just affect the flow locally near the tip, but affected the flow across the entire blade span. For the filled honeycomb shroud, Figs. 12(b) and 12(c) show that the radial variations in total temperature and total pressure were more nearly constant than for the two open honeycomb shrouds. The most dramatic efficiency improvement occurred with the filled honeycomb shroud as shown in Fig. 12(d). This figure shows that the efficiency for the filled honeycomb was higher across the entire blade span compared to the open honeycomb. The biggest difference was near the midspan location, where a large loss area existed for the open honeycomb. For the half depth honeycomb this midspan region loss area was still present, although not as severe.

The reason for the large performance loss with the open honeycomb shroud was not completely understood. It appeared that a major loss contribution was due to gas being transported into and out of the honeycomb cells as a result of the blade-to-blade pressure differential at the tip section. Because the transported gas lost kinetic energy in entering and leaving the honeycomb cells, a major loss resulted. It was believed, however, that the entire loss mechanism was very complicated and could not be understood without further experimental and analytical research.

CONCLUDING REMARKS

The results obtained in this experimental investigation showed that the use of an open honeycomb shroud caused the large performance deficit for the aircraft engine starter turbine. Although the loss mechanism associated with the honeycomb shroud was not fully understood, there was no doubt that the use of this shroud in this turbine application was not desired. It is possible that a honeycomb shroud configuration exists in which the performance loss would be minimized. However, the results of this program could not be used to determine what this optimum configuration would be.

The efficiency that was achieved when the honeycomb shroud was filled (.868) was considered excellent for a turbine of this size and work factor. Although no stator inlet boundary layer measurements were made, it was believed that the use of the short, bellmouth inlet with its resultant thin boundary layers was an important contributor to this efficiency level.

REFERENCES

1. J. E. Haas, et. al, "An Experimental Investigation to Determine the Reasons for the Low Aerodynamic Performance of a 13.5-Centimeter Tip Diameter Aircraft Engine Starter Turbine." NASA TP. To be published.

Table 1 - Turbine Design Parameters

Parameter	Actual	Equivalent	Test Conditions
Turbine inlet temperature, K	501.7	288.2	322
Turbine inlet pressure, N/cm^2	31.8	10.1	13.8
Mass flow rate, Kg/sec	0.828	0.348	0.448
Rotative speed, rpm	43,893	33,267	35,178
Specific work, J/g	114.2	65.6	73.3
Torque, N-M	20.6	6.5	8.8
Power, KW	94.5	22.8	32.8
Total to total pressure ratio, P_1'/P_3'	2.84	2.84	2.84
Total to static pressure ratio, P_1'/P_3	3.10	3.10	3.10
Total efficiency, η_{1-3}'	0.880	0.880	0.880
Static efficiency, η_{1-3}	0.819	0.819	0.819
Work Factor $\frac{\Delta V_u}{U_m}$	1.47	1.47	1.47
Reynolds number, $\frac{w}{\mu r_m}$	5.1×10^5	3.2×10^5	3.7×10^5
Blade jet speed ratio, ν_{1-3}	0.524	0.524	0.524

Table 2 - Turbine Design Physical Parameters

Parameter	Stator	Rotor
Actual Chord, cm (a)	2.490	1.592
Axial chord, cm (a)	1.693	1.405
Leading edge radius, cm	.166	.104
Trailing edge radius, cm	.021	.015
Radius, cm		
Hub	5.32	5.13 (b)
Mean	6.02	6.02 (b)
Tip	6.72	6.87 (b)
Blade height, cm	1.40	1.56
Solidity, (a)	1.51	1.56
Aspect ratio	.56	.98
Number of blades	23	37
Radius ratio	.79	.77
Blade Pitch, cm (a)	1.64	1.02

a Values at mean section
b Values at rotor exit

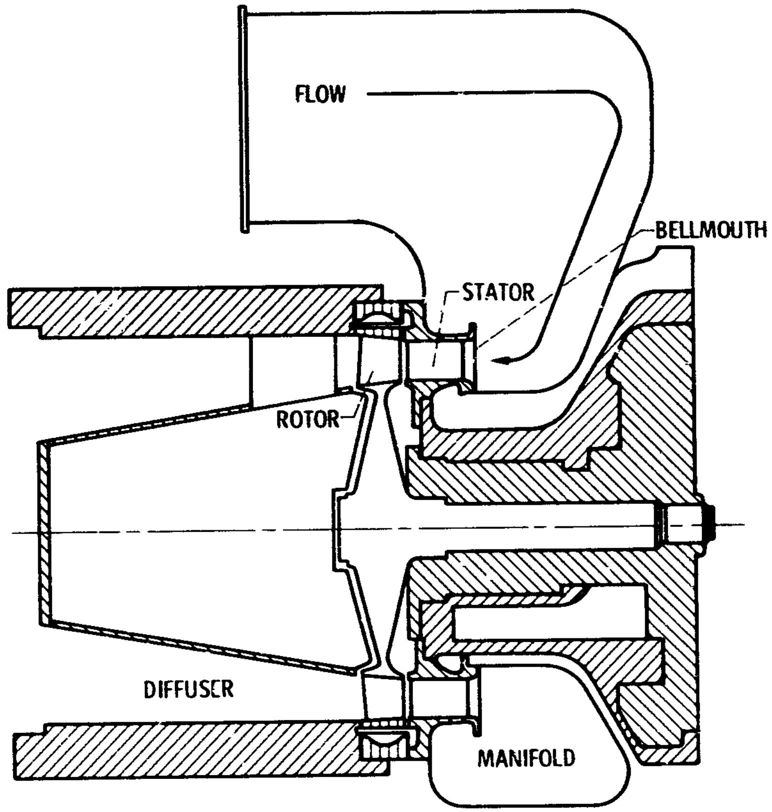
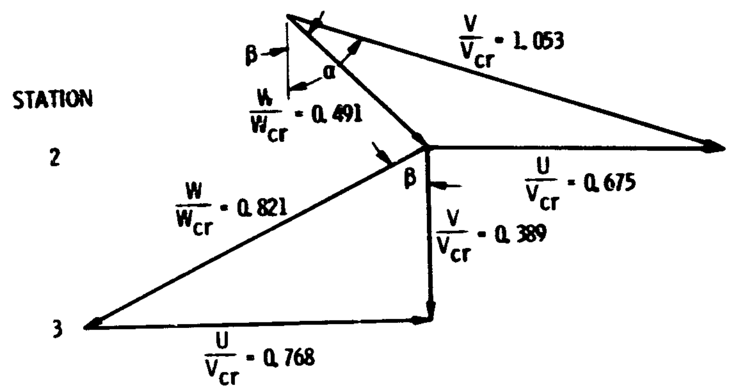


Figure 1. - Turbine cross-sectional schematic.



	α	β
STATION 2	73°	47.2°
STATION 3	0°	-63.1°

Figure 2. - Mean-section velocity diagram.

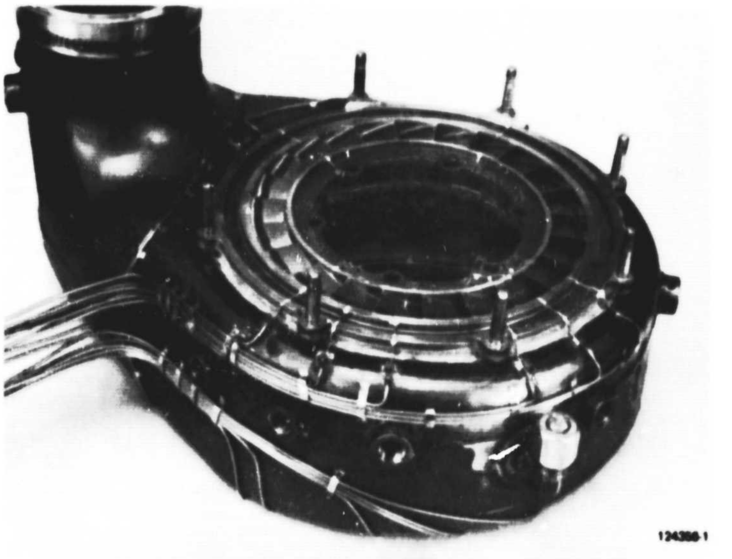
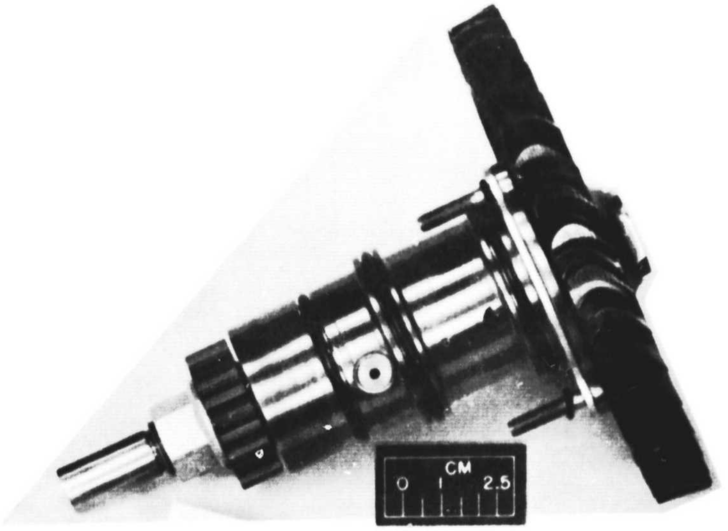


Figure 3. - Manifold and stator assembly.



C-80-1291

Figure 4. - Rotor assembly.

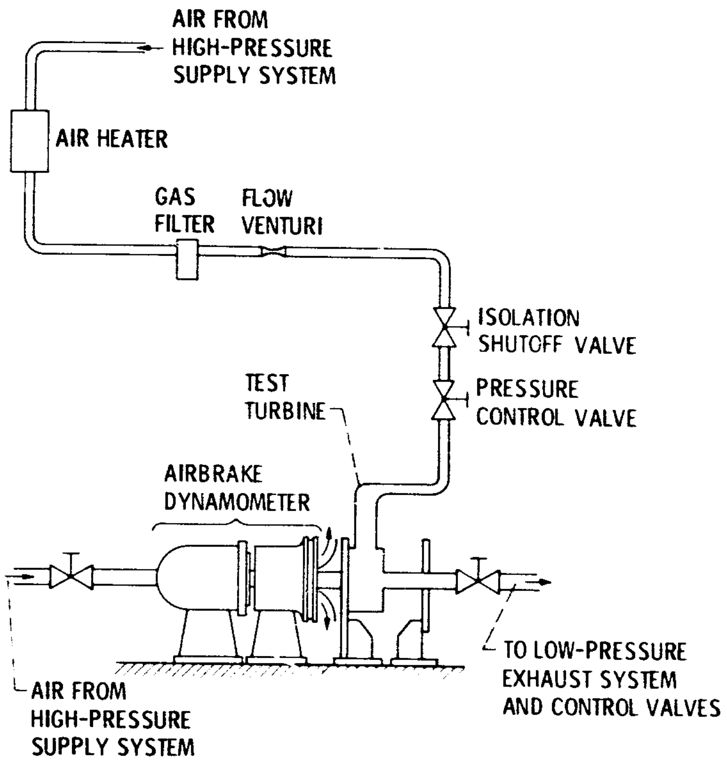


Figure 5. - Experimental equipment schematic.

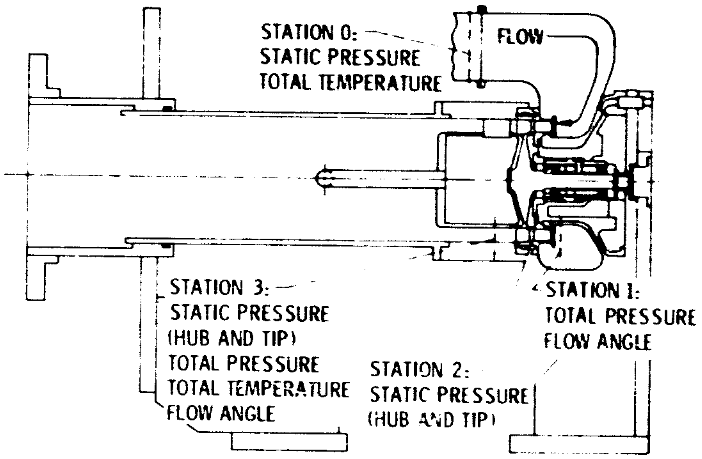
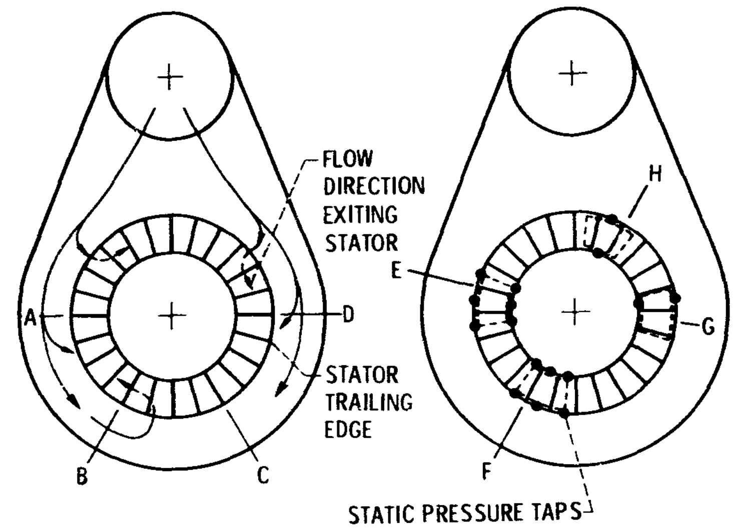
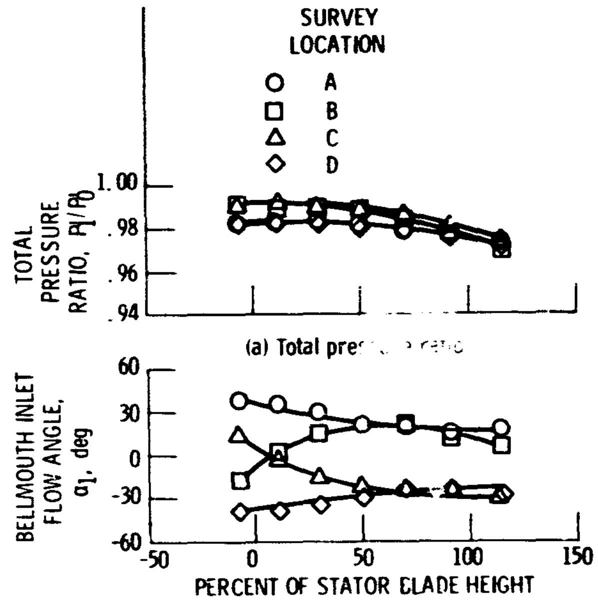


Figure 6. - Turbine instrumentation.



(a) Bellmouth inlet survey locations. (b) Stator exit survey sectors.

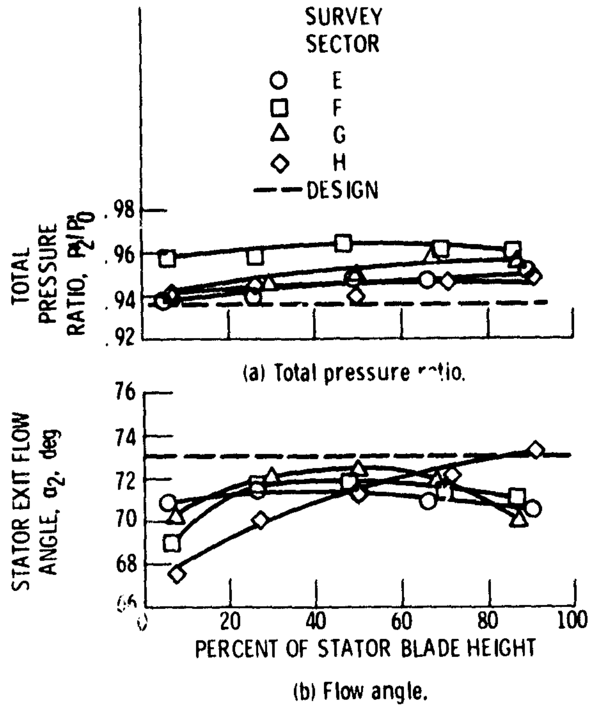
Figure 7. - Bellmouth inlet and stator exit survey locations (view looking into manifold).



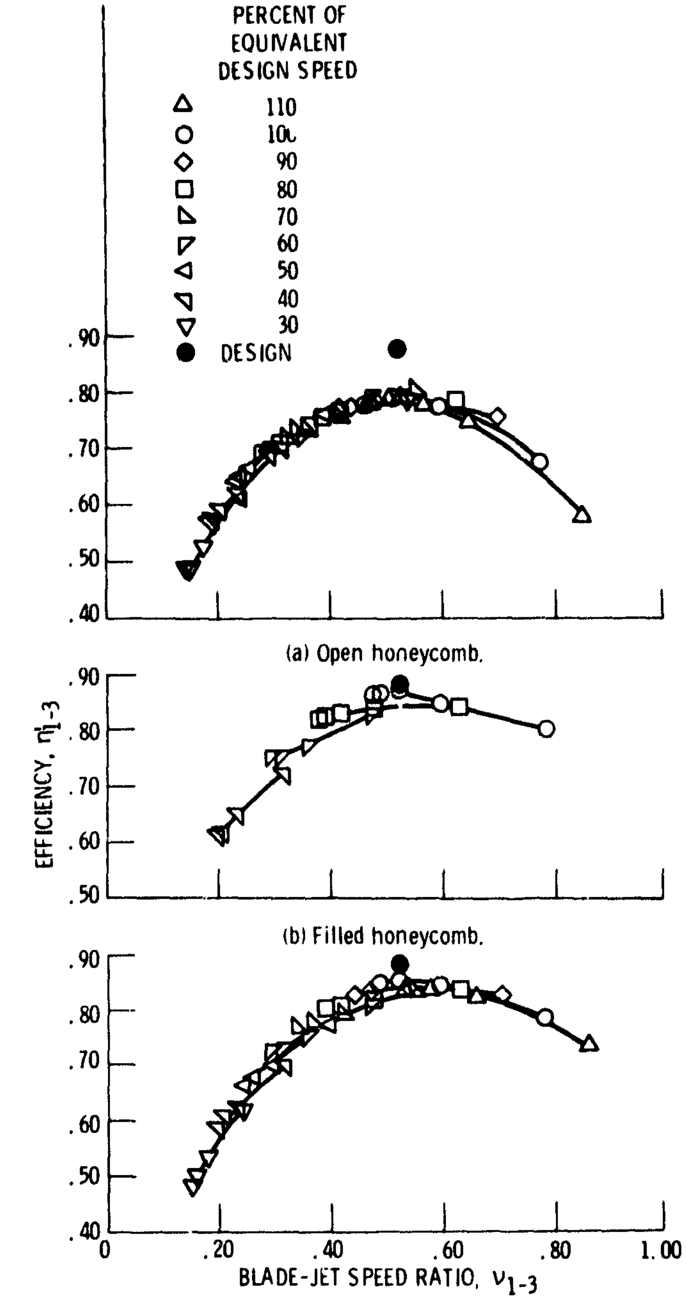
(a) Total pressure ratio.

(b) Flow angle.

Figure 8. - Bellmouth inlet survey results.



e 9. - Stator exit survey results.



(a) Open honeycomb.

(b) Filled honeycomb.

(c) Half-depth honeycomb.

Figure 10. - Variation of efficiency with blade-jet speed ratio for the three honeycomb shroud configurations.

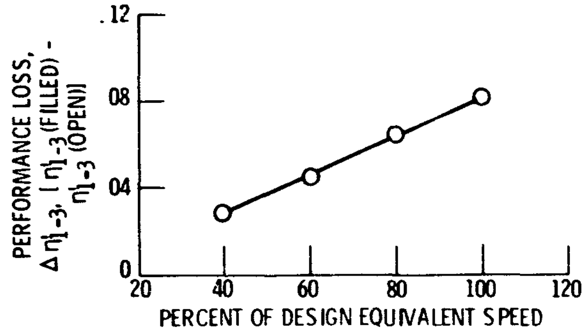


Figure 11. - Performance loss between the filled and open honeycomb shrouds at design blade-jet speed ratio.

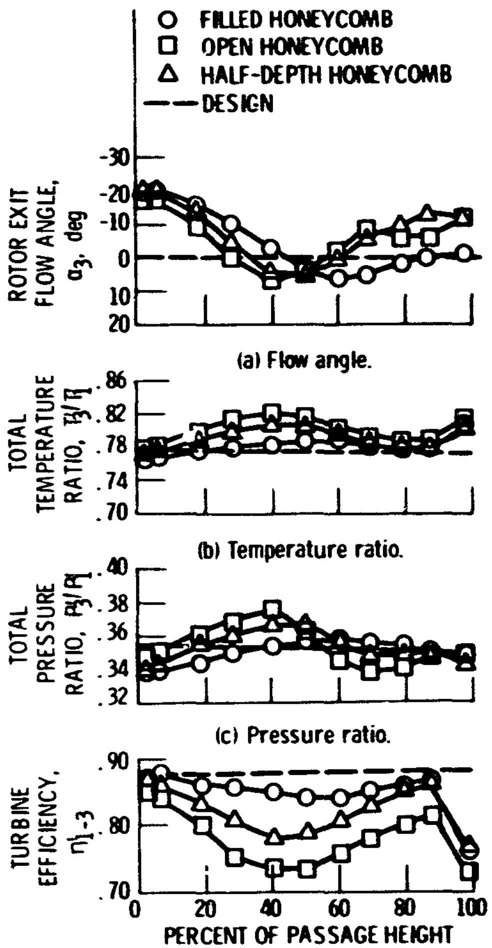


Figure 12. - Rotor exit survey results.

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15. Supplementary Notes Jeffrey E. Haas, Propulsion Laboratory, AVRADCOM Research and Technology Laboratories, Lewis Research Center, Cleveland, Ohio; Richard J. Roelke, Lewis Research Center; and Paul Hermann, Sundstrand Corporation, Rockford, Illinois.
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16. Abstract

An experimental investigation was made at NASA Lewis Research Center to determine the reasons for the low aerodynamic performance of a 13.5-centimeter-tip-diameter aircraft engine starter turbine. The investigation consisted of an evaluation of both the stator and the stage. An approximate ten percent improvement in turbine efficiency was obtained when the honeycomb shroud over the rotor blade tips was filled to obtain a solid shroud surface.

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Honeycomb shroud; Starter turbine; Axial-flow turbine; Efficiency; Aerodynamic performance	Unclassified - unlimited STAR Category 07

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