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EXPERIMENTAL MEASUREMENTS AND ANALYTICAL ANALYSIS RELATED TO GAS TURBINE HEAT TRANSFER

Summary Report For Period

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HEAT TRANSFER AND PRESSURE MEASUREMENTS FOR THE SSME FUEL-SIDE TURBOPUMP*

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ABSTRACT

A measurement program is currently underway at the Calspan-UB Research Center (CUBRC) which utilizes the Rocketdyne two-stage fuel-side turbine with the engine geometric configuration reproduced. This is a full two-stage turbine for which the vane rows and the blades are the engine hardware currently used on the Space Shuttle turbopump. This paper provides a status report of the experimental program and a description of the instrumentation and the measurements to be performed. The specific items that will be illustrated and described are as follows: (a) the gas flow path, (b) the heat-flux instrumentation, (c) the surface-pressure instrumentation (d) the experimental conditions for which data will be obtained and (e) the specific measurements that will be performed.

1. INTRODUCTION

The data to be obtained for this program are intended to serve two purposes: (a) to provide help with code validation and (b) to provide comparison data for a blow down test rig that is under construction at Marshall Space Flight Center (MSFC). Concerning the code validation aspect, the philosophy of this author is to involve the data user very early in the program so as to increase the probability of generating a data set to best meet the needs of the user and one that is most adaptable to the code capabilities. For the initial instrumentation package, both vane rows but only one blade row are instrumented. The particular instrumentation locations on the first stage vane and blade row were selected on the basis of preliminary calculations presented by the participants during joint meetings with the author and representatives of NASA Lewis, NASA Marshall, Scientific Research Associates, Rocketdyne, and Pratt and Whitney. The people involved in this process are part of a NASA turbine team working group and via this mechanism

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they will be kept actively involved while the measurements are being performed. Concerning the comparison data for the rig under construction at MSFC, the surface pressure data on the first vane row will be used for this purpose because the MSFC instrumentation suite does not contain the capability to perform heat-flux measurements nor does it contain the capability to perform surface pressure measurements on the rotating blades. The heat-flux gage locations on the second vane correspond to locations of surface pressure measurements in the MSFC rig.

The flow and heat transfer that occur in a turbine stage (or stages) represent one of the most complicated environments seen in any practical machine: the flow is always unsteady, can be transonic, is generally three-dimensional, and is subjected to strong body forces. Despite these problems, satisfactory designs have been achieved over the years due to advances in materials and manufacturing processes, as well as to the development of a sound analytical understanding of the flow and heat-transfer mechanics that define performance. These analytical developments were made possible by a series of approximations, in which the level of detail retained in the modeling was sufficient to reveal important physical effects, while still allowing solutions to be found by available analytical/numerical methods.

The major milestones in the development of these methods have been the approximations that flow through each blade row is steady in coordinates fixed to the blades, that three-dimensionality can be handled by treating a series of two-dimensional flows in hub-to-shroud and blade-to-blade surfaces, and that the effects of viscosity can be estimated by non-interacting boundary-layer calculations and by loss models to account for secondary flow.

This technology base is surrounded by many analyses and numerical codes which can treat the flow on higher levels of approximation, and which are used from time to time to provide refined estimates of the flowfield and heat transfer, typically near a design point. Three-dimensional and unsteady flow effects are two areas where recently developed computational tools can provide useful information on the flow conditions, at least for the first stage of a multistage turbine. However, in the second and subsequent stages, these effects become more pronounced. The current state-of-the-art analyses can predict the second stage vane pressure distribution but their ability to predict the heat-flux levels on the second vane has been found to be inadequate (1). These analyses are also probably inadequate on the second rotor row, but at this time there are no generally available experimental data that can be used to illustrate the lack of agreement between prediction and experiment.

Unsteadiness and three-dimensionality are direct consequences of the interaction of blades moving through vane wakes and the impact of multiple blade rows, The environment associated with the SSME turbopump turbine lends itself to a multistage analysis. Until very recently, such an analysis would have been envisioned as a complete, time-accurate, fully three-dimensional description of the flowfield. Some first steps toward the calculation of such flows can be seen in the work of Rai (2, 3), but it is clear that the computational costs of this approach could very quickly become prohibitive if one does not have access to unlimited cost-free computation.

Another approach to the problem is that described by Rao and Delaney (4), which until the present time, has only been applied to a single stage. The method proposed by these authors solves the quasi-three-dimensional Euler/Navier-Stokes equations using the explicit hopscotch scheme. The full stage computation is performed by coupling vane and blade solutions on overlapping O-type grids. In Reference (5), Dunn et al., present comparisons between the predictions of Reference (4) and experimental data that were obtained for a full-stage turbine using the same experimental techniques described in this paper. Comparisons are presented for the time-averaged surface pressure, the unsteady envelope of the surface pressure, and the phase-resolved surface pressure on the vane and on the blade. The agreement between the predictions and the measurements was found to be very good. Detailed heat-flux data of the same type mentioned above were also obtained and will be presented in the literature in the very near future.

An alternate approach that has begun to receive attention in the recent past is based on Adamczyk's formulation of the passage-averaged equations (6, 7), which until now have been used only as an analysis tool. It is apparent that this technique holds promise as the basis of a design method whose physical basis is considerably advanced beyond the current state of the art, and whose numerical implementation is simple enough to achieve without the need for excessive hours of supercomputer time.

The formulation of closure models necessary to exploit Adamczyk's formulation relies on the availability of time-resolved flowfield data. Some of this information can be obtained from the work of Dring et al. (8) who have probed the field within and around a one-and-one-half stage rotating turbine. Further results to support this modeling effort can be found in numerical calculations of interacting stages; some of this work has been done at CUBRC (Taulbee and students), under support from NASA Lewis. Additional data that may be useful for the application of this technique to the SSME fuel-side turbopump analysis will be generated during the course of the measurement program described here.

2. DESCRIPTION OF THE EXPERIMENTAL TECHNIQUE, THE TURBINE FLOW-PATH, AND THE INSTRUMENTATION

2.1 The Experimental Technique

The measurements to be performed utilize a shock-tunnel to produce a short-duration source of heated and pressurized gas that will subsequently be passed through the turbine. A wide variety of gases could be used for this purpose, but air has been selected for these experiments. A schematic of the experimental apparatus illustrating the shock tube, an expansion nozzle, a large dump tank and a device that houses the turbine stage and provides the flow path geometry is shown in Figure 1(a). The shock tube has an 18.5-inch diameter by 40-feet long driver tube and an 18.5-inch diameter by 60-feet long driven tube. The driver tube was designed to be sufficiently long so that the wave system reflected from the endwall (at the left-hand end of the sketch) would not terminate the test time prematurely. At the flow conditions to be run for these measurements, the test time is very long for a short-duration facility being on the order of 40 milliseconds.

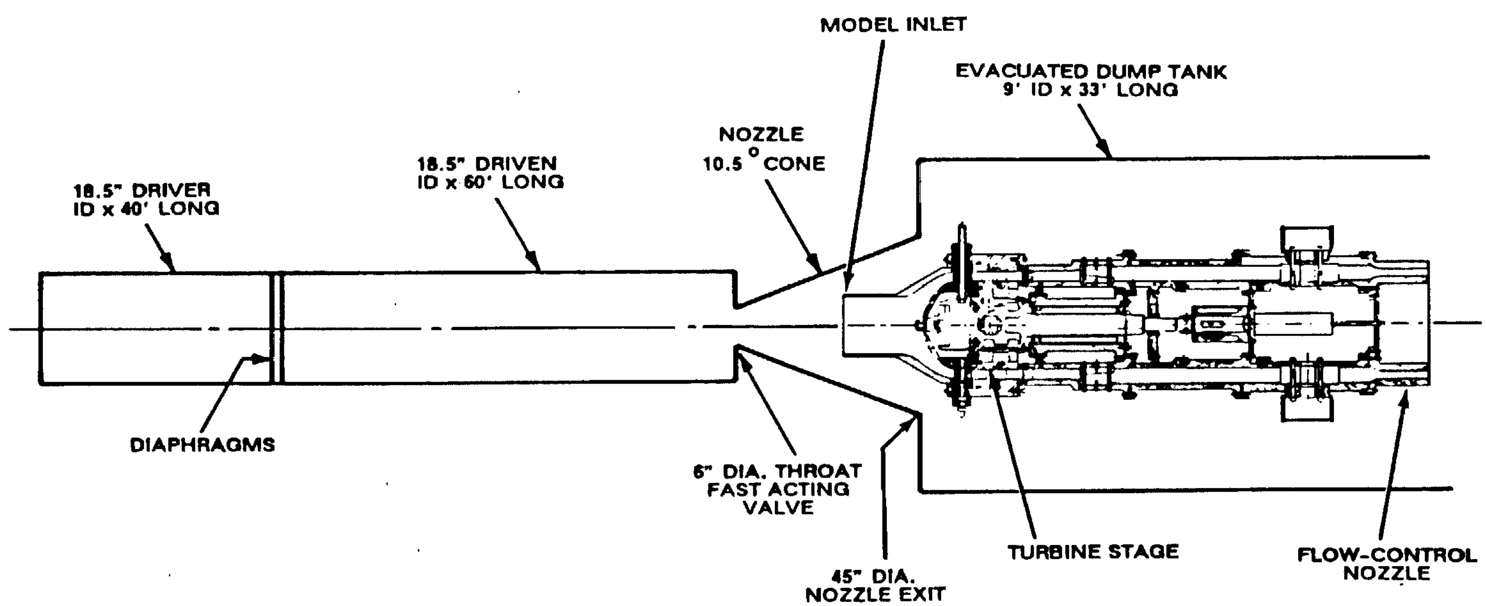

Figure 1 (a) SKETCH OF A TYPICAL TURBINE STAGE LOCATED IN A SHOCK-TUNNEL

In order to initiate an experiment, the test section is evacuated while the driver, the double diaphragm section and the driven tube are pressurized to predetermined values. These pressure values are selected so that the flow conditions at the NGV inlet will duplicate the design flow function ($W\sqrt{\theta/\delta}$), the ratio of wall temperature to total temperature (T_W/T_O), the total to total pressure ratio across the two stages (or the total to static pressure ratio across the two stages) the total to total temperature ratio across the stages, and the corrected rotor speed. The shock-tunnel facilities used for gas turbine research have the unique advantage that the value of T_O can be set at almost any desired value in the range of 800°R to 3500°R. The design pressure ratio across the turbine is established by altering the throat diameter of the flow control nozzle located at the exit end of the device housing the turbine. Simple one-dimensional calculations provide a very good first estimate of the necessary exit area. On the basis of previous experience only one or two runs are generally required to confirm that the proper exit area has been selected so that the desired stage pressure ratio can be established to within an acceptable tolerance. Another, unique advantage of this facility is that the total pressure or the Reynolds number at the entrance to the NGV can be changed over a rather wide range by moving the turbine model axially in the expanding nozzle flow so as to intercept the flow at a different free stream Mach number. If this doesn't provide sufficient range, then the reflected-shock pressure can be increased or the total temperature can be decreased in order to increase the Reynolds number.

Figure 1(b) is a photograph of the facility illustrating many of the components described in the preceding paragraph. Figure 1(c) is a wave diagram for the shock tube. The gas that subsequently passes through the turbine has been processed by both the incident and the reflected shock shown in Figure 1(c). The reflected-shock reservoir gas is expanded in the primary nozzle which has the effect of increasing the flow velocity, decreasing the total pressure and maintaining the total temperature at the reservoir value. The device housing the turbine will not pass all of the weight flow available in the primary nozzle, so the inlet must be carefully located in order to avoid a hammer shock. That is, there must be sufficient flow area for a normal shock to establish outside the inlet entrance and for the remainder of the flow not passed through the turbine to pass between the lip of the inlet and the nozzle wall. If the inlet is placed too far into the nozzle, the nozzle flow will be blocked and very large short-duration forces can be exerted on the model with potentially disastrous effects. The flow downstream of the inlet normal shock is subsonic at a pressure determined by the shock strength at the particular pick-off location in the expansion.

2.2 The Turbine Flowpath

The device to house the turbopump turbine that is shown located in the expansion nozzle in the sketch of Figure 1(a) has been designed and constructed. Figure 2 is a drawing of this device illustrating the extent to which the flowpath ahead of the turnaround ducts has been reproduced for the upcoming measurements. For these measurements, reproduction of the turnaround duct geometry was not felt to be important by mutual agreement among members of the NASA turbine working group. The device housing the turbine is designed so that the flow area is constant between the inlet and the entrance to the NGV row. The Mach number in this region is low, gener-

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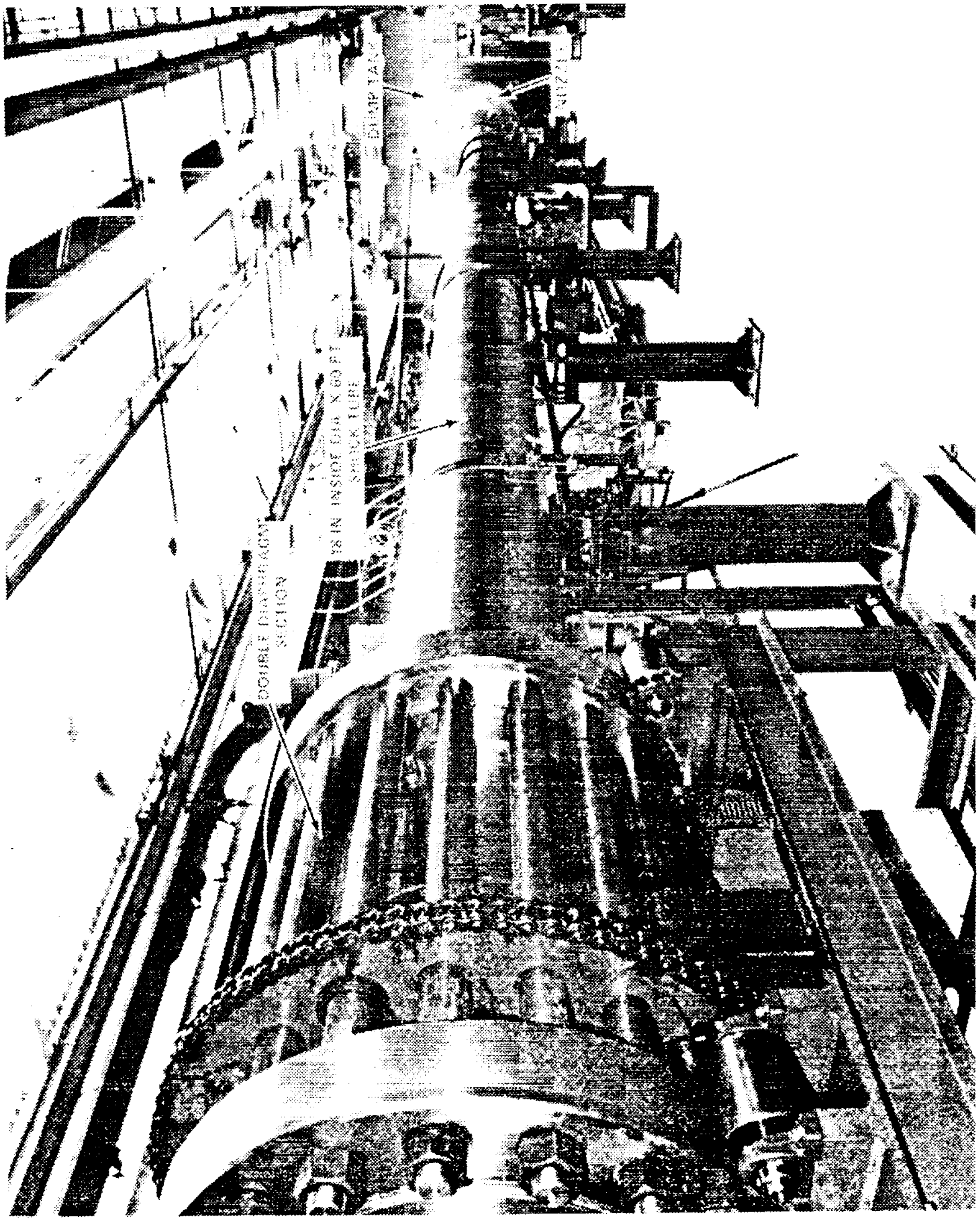

Figure 1 (b) PHOTOGRAPH OF CALSPAN's SHOCK-TUNNEL FACILITY FOR TURBINE RESEARCH

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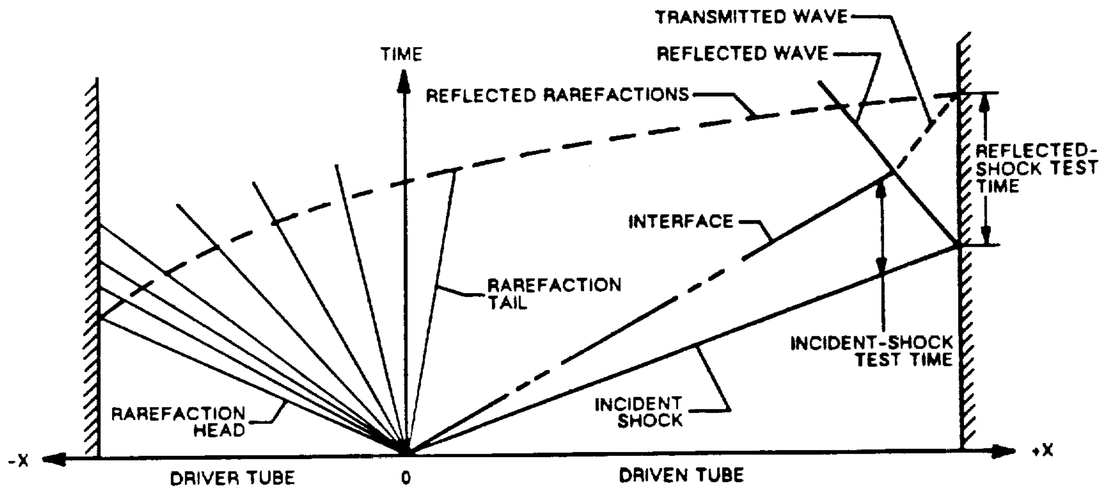

Figure 1 (c) SKETCH OF A TYPICAL SHOCK-TUBE WAVE DIAGRAM

ally on the order of 0.05 to 0.10. The first-stage vane and rotor as well as the second-stage vane and rotor are shown. The dome and the bolt that actually exist at its center are also reproduced. In addition, the thirteen struts upstream of the first-stage vane as well as the 12 flow straighteners and 6 struts downstream of the second rotor have been included corresponding to the engine hardware. Figure 3 is a photograph of the assembled model illustrating some of the components. Figure 4 is a photograph of the dome showing the thirteen struts located downstream of the dome and just upstream of the inlet to the first-stage turbine nozzle guide vane. The bolt holes in the dome and the slots ahead of the struts will be filled with Epoxy and contoured flush with the surface prior to final assembly of the unit.

2.3 Heat-flux Instrumentation

The heat-flux measurements to be obtained in this research will use thin-film resistance thermometers. These devices represent an old and very well established technology that was developed as part of the early hypersonic flow research work in the late 50's for measurement of heat-flux distributions in short-duration facilities. The thin-film gages are made of platinum (-100 A thick) and are hand painted on an insulating Pyrex substrate in the form of a strip that is approximately 0.004-inches wide by about 0.020-inches long. The response time of the elements is on the order of 10^{-8} s. The substrates containing the heat-flux gages are placed within the metallic skin at known locations throughout the turbine stage and secured with Epoxy. The substrate onto which the gage is painted can be made in many sizes and shapes, depending upon the

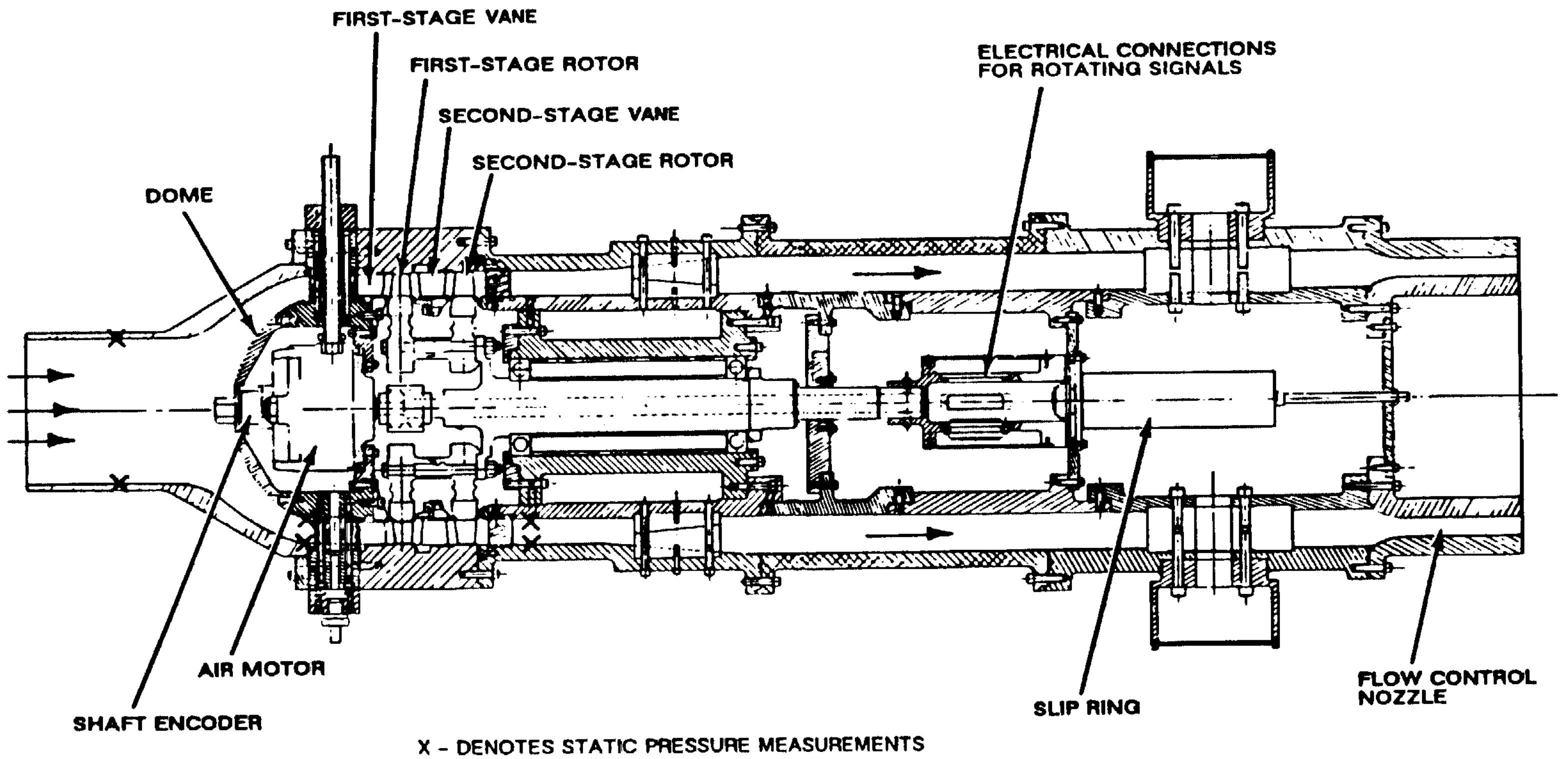

Figure 2 SKETCH OF DEVICE HOUSING SSME TURBINE STAGE

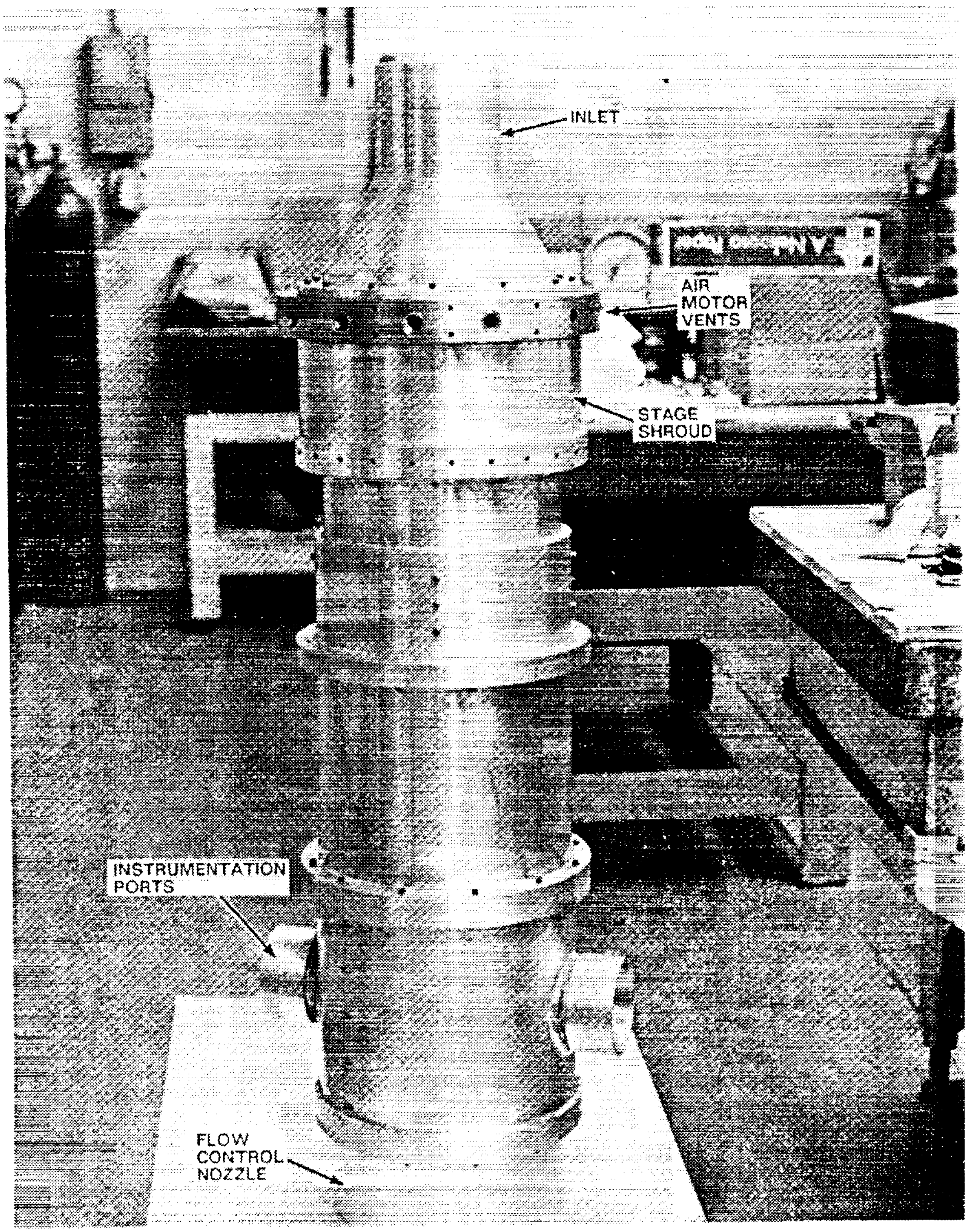

Figure 3 PHOTOGRAPH OF ASSEMBLED DEVICE USED TO HOUSE STAGE

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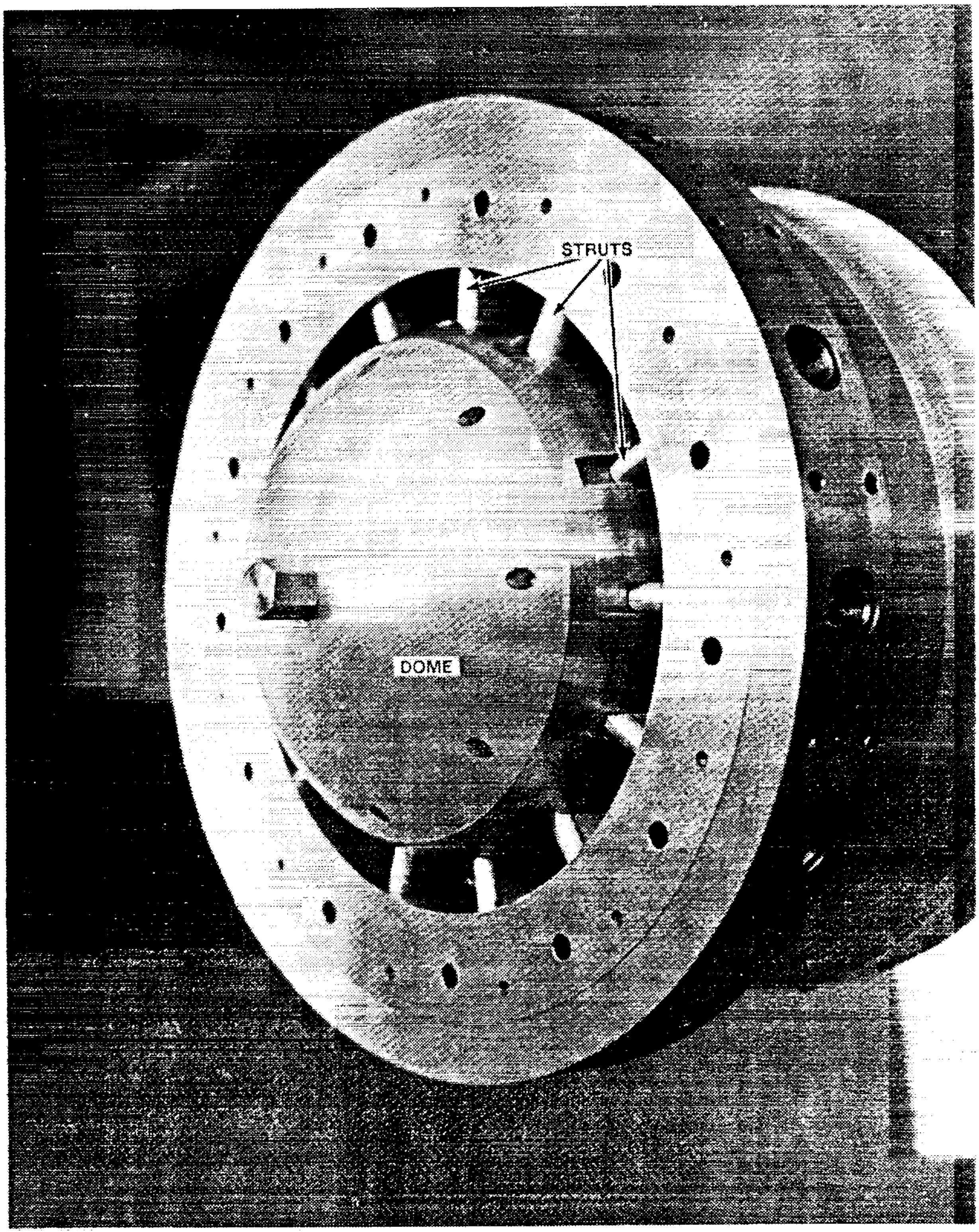

Figure 4 PHOTOGRAPH OF DOME AND STRUTS UPSTREAM OF NOZZLE GUIDE VANES

particular portion of the turbine stage for which one is interested in obtaining data; i.e. the button gage, the contoured leading-edge gage, contoured strip gages or stagnation-point gages.

Both the button-type gages and the contoured leading-edge inserts have been used on many of the previous research programs conducted at Calspan and CUBRC. Figure 5(a) and 5(b) are photographs of two of the SSME turbopump rotor blades that have been instrumented with button-type gages. For the button-type gages, the substrate is 0.040-inches in diameter by 0.032-inches thick. The pressure surface button-type gage instrumentation shown in Figure 5(a) has 3 gages at 90% span, 5 gages at 50% span and 3 gages at 10% span. Figure 5(b) is a photograph of the blade suction surface button-type instrumentation illustrating 3 gages at 90% span, 5 gages at 50% span and 4 gages at 10% span. The suction side of the platform contained 2 gages. Table 1 gives the locations of all of the heat-flux instrumentation on the first-stage rotor blades. The photographs of Figure 5 suggest that the blade surface is quite rough which will result in relatively high heat transfer levels. The surface roughness for this blade has been measured by UTRC and is available for input to the analysis of the heat-transfer data. Figure 6 is a close-up photograph of one of the heat-flux gages installed in the blade which illustrates the surface roughness relative to the gage installation. The gages have been installed flush with the top of the roughness. Figure 7 is a photograph of the leading-edge insert which contains a total of 7 gages. The pressure surface side of the insert has three gages, the suction surface side has three gages, and one gage is located at the geometric stagnation point. The surface of this insert is smooth from the leading edge to the last gage. It will be of interest to compare the distribution of the heat-flux values measured on the insert with that measured by the button-type gages which are located on the rough blade surface.

Figure 8 is a photograph of the front and rear view of the first-stage NGV prior to installation of the surface-pressure and the heat-flux gages. At the time of this writing, installation of the pressure transducer instrumentation was not complete and thus finished photographs were not available. However, Table 2 provides a tabulation of the heat-flux instrumentation locations on this vane row. These locations will undergo minor revision when final measurements are obtained after installation. Figure 9(a) is a photograph of the second-stage NGV prior to installation of the heat-flux gage instrumentation. Figure 9(b) is a photograph of the same vane row but with the gages installed. All of the heat-flux gages installed on this second stage vane were placed at midspan and Table 3 provides a tabulation of their locations. The gage distribution selected was 10 gages on the suction surface and 7 gages on the pressure surface. No pressure transducers were put on this stage.

2.4 Surface-pressure Instrumentation

The measurement program also uses miniature flush diaphragm pressure transducers located on the first-stage vane and the first-stage blade. The particular gages being used are Kulite Model LQ-062-600A with an active pressure area of 0.025 by 0.025 inches, a frequency response of about 100 kHz in the installed configuration, and a pressure range of 0 to 600 psia. This large value of the upper pressure limit is necessary because of the experimental conditions necessary to

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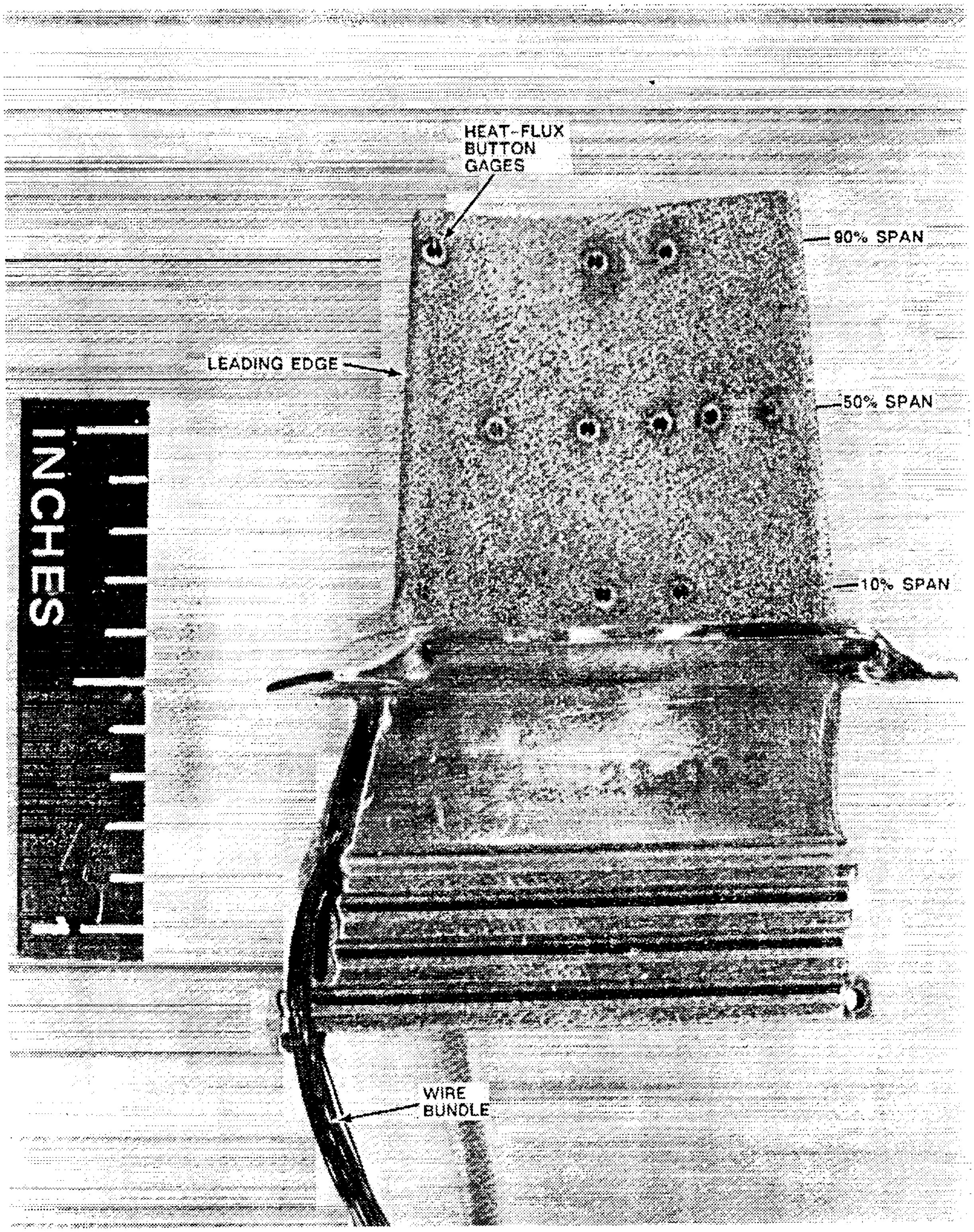

Figure 5 (a) BUTTON-TYPE HEAT-FLUX GAGES ON FIRST-STAGE BLADE PRESSURE SURFACE

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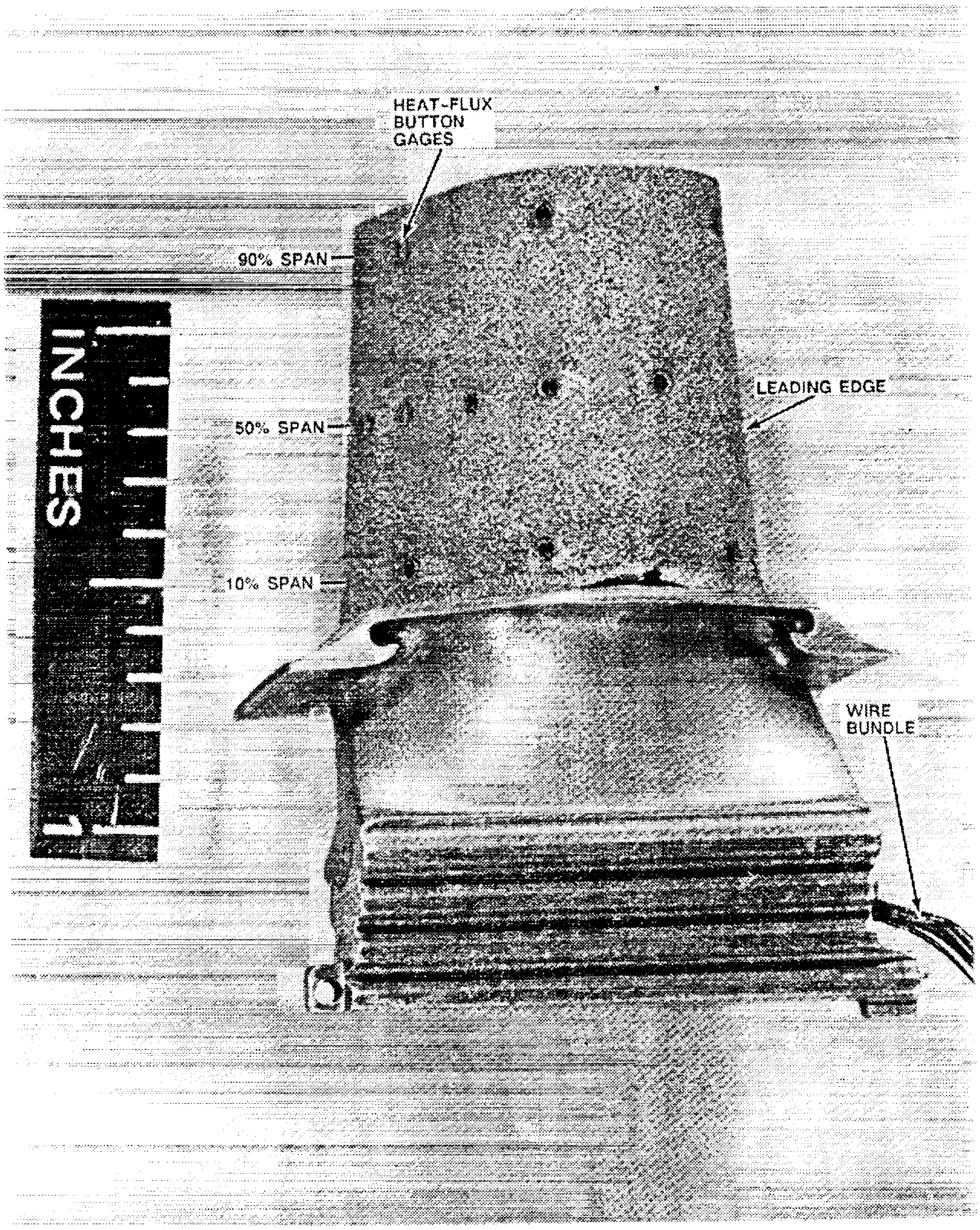

Figure 5 (b) BUTTON-TYPE HEAT-FLUX GAGES ON FIRST-STAGE BLADE SUCTION SURFACE

Table 1
LOCATION OF HEAT-FLUX GAGES ON FIRST-STAGE ROTOR BLADES

Type	Location	Wetted Distance Along Surface, %
Button	10% span, suction surface	5.4, 48.6, 65.1
Button	50% span, suction surface	28.5, 48.4, 64.1, 82.0, 92.8
Button	90% span, suction surface	6.8, 46.4, 60, 70, 82.0, 90
Button	10% span, pressure surface	7.3, 16.1, 51.6, 80.2
Button	50% span, pressure surface	23.6, 44.4, 60.4, 72.6, 87.5
Button	90% span, pressure surface	4.8, 45.2, 62.5
Button	Platform, Near Suction Surface	22.1, 59.1
L.E. insert	50% span, stagnation point	0
L.E. insert	50% span, suction surface	5.8, 11.8, 17.7
L.E. insert	50% span, pressure surface	8.7, 16.1, 21.9
Button	Blade tip	20, 40, 60, 82
Button	Shroud	10, 30, 50, 70, 90

achieve the extreme Reynolds number which represents the upper limit to which some of the experiments will be run. A total of 28 pressure transducers were installed on the vanes and 24 pressure transducers were installed on the blades. As noted earlier, the instrumentation installation on the first-stage vane row was not completed at this writing so photographs are not available. However, it should be noted that the transducers were placed on several different vanes so as not to disturb the integrity of the surface. Table 4 provides the locations of the surface pressure transducers on the first-stage vane row.

The pressure transducers were also distributed over several different blades. Table 5 provides the locations of the pressure transducers on the blades. Figure 10(a) is a photograph of several transducers installed at midspan on the blade suction surface, Figure 10(b) is a photograph of three pressure transducers at 90% span, and Figure 10(c) is a photograph of several transducers at 10% span on the suction surface.The white material covering the active area of the transducer is a thin coating of silastic material that is used to provide thermal isolation from the external flow for these short-duration experiments. The black material seen on the photographs is Epoxy which is used to fill the lead wire channels and is then contoured to the blade geometry. Eleven transducers are on the pressure surface and seventeen transducers are on the suction surface. The data obtained on the rotor is transferred to the laboratory frame via a Freon-oil cooled slip-ring unit. The particular unit being used here contains 200 gold rings.

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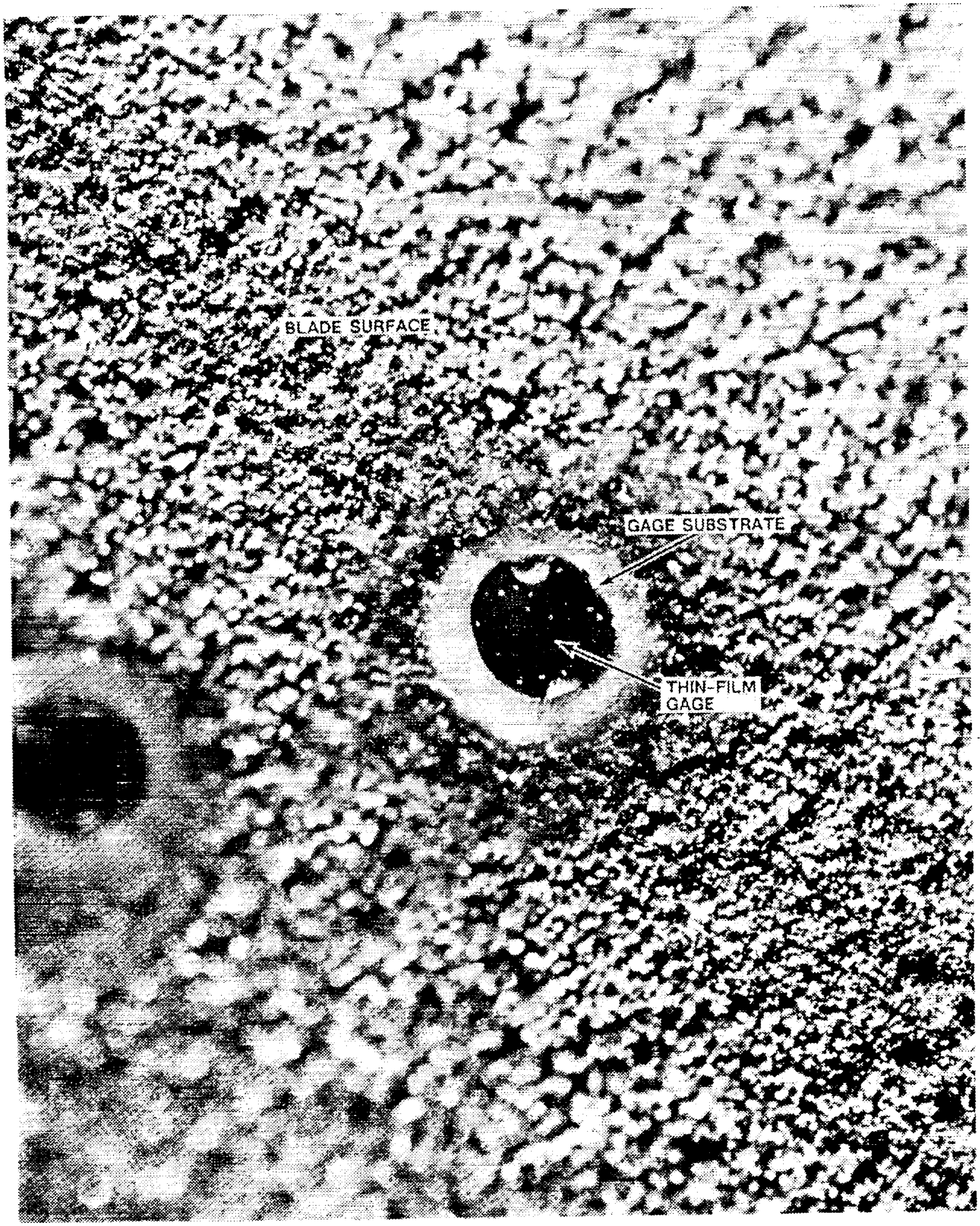

Figure 6 CLOSE-UP PHOTOGRAPH OF BUTTON-TYPE GAGE INSTALLED IN FIRST-STAGE ROTOR BLADE

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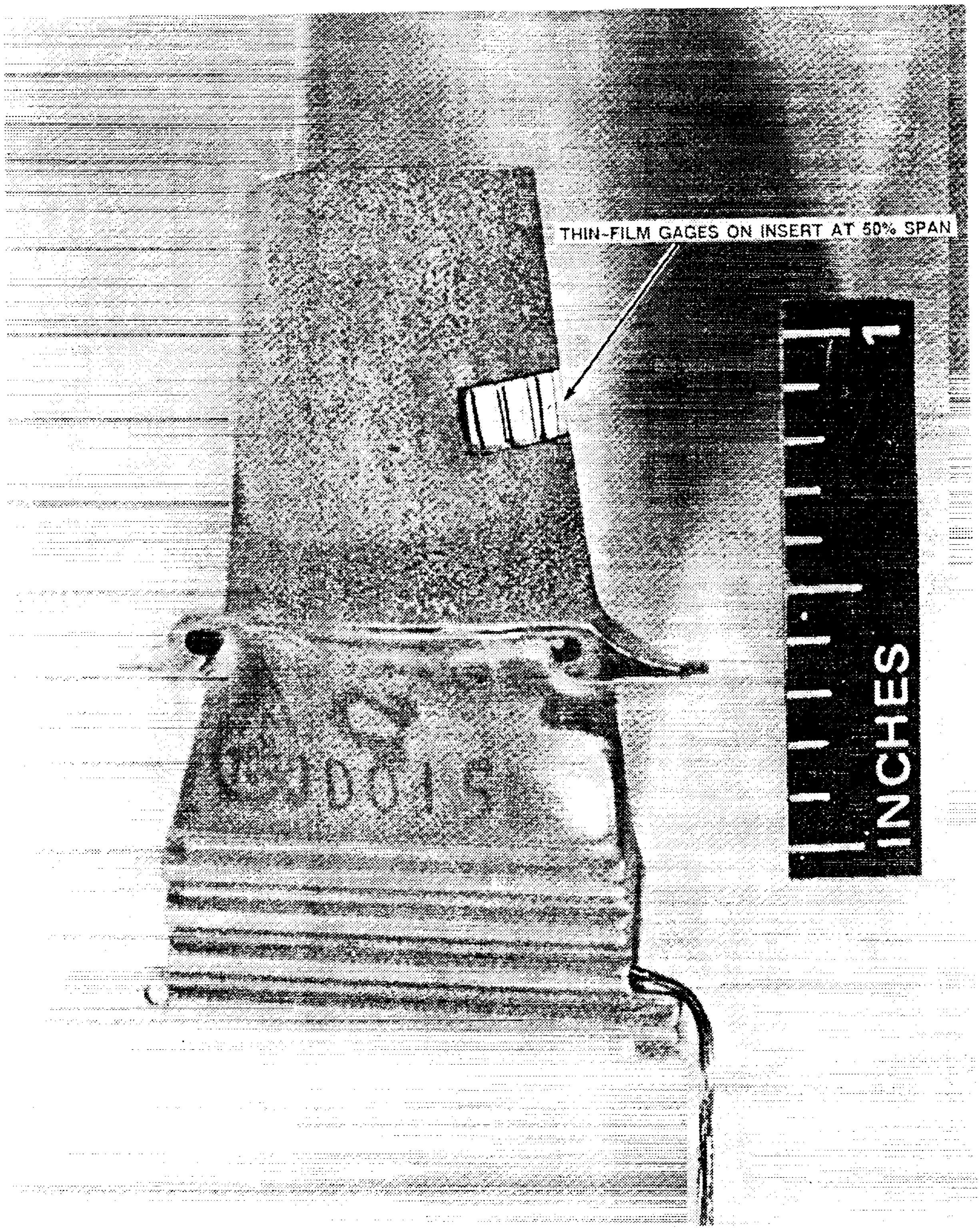

Figure 7 PHOTOGRAPH OF LEADING-EDGE INSERT HEAT-FLUX GAGES ON FIRST-STAGE BLADE

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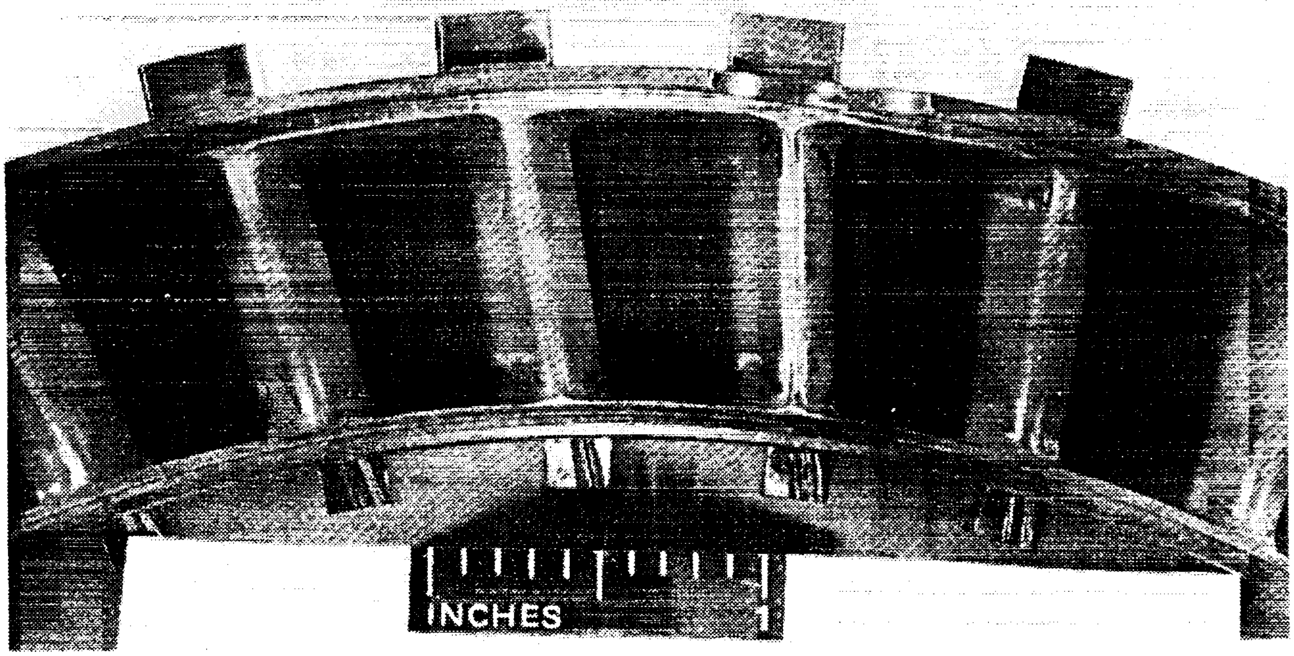
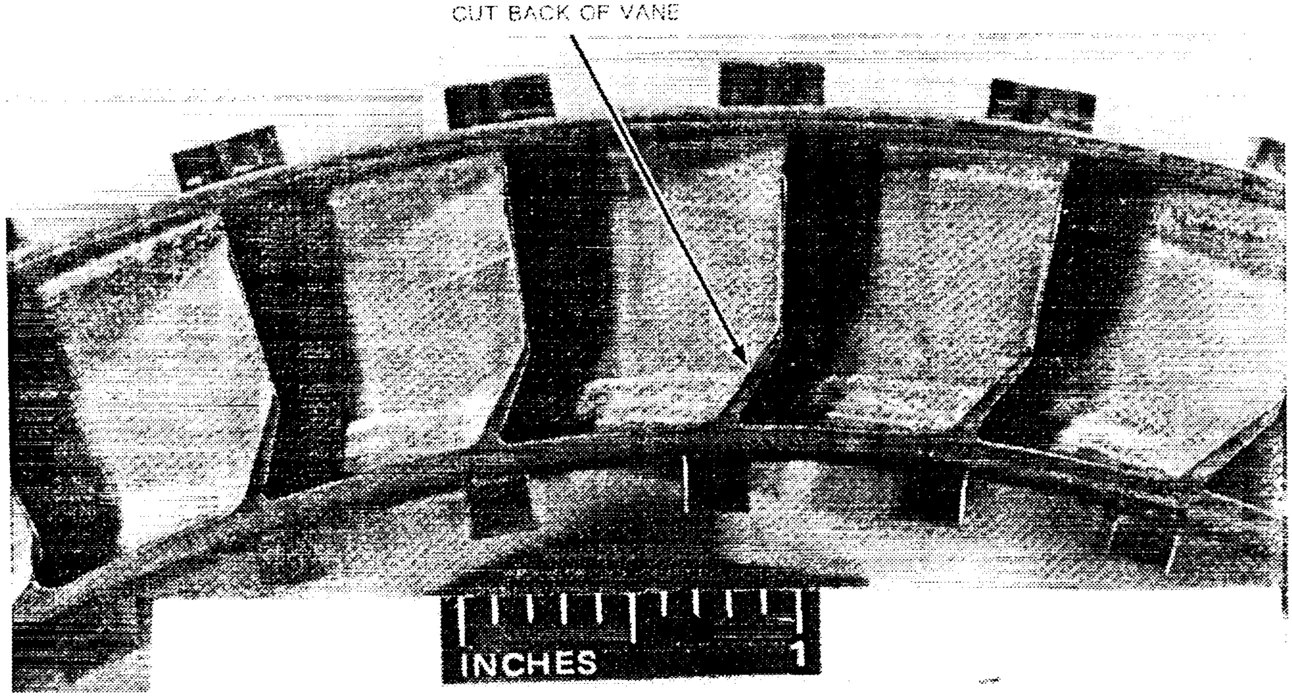


(a) FRONT VIEW



(b) REAR VIEW

Figure 8 PHOTOGRAPH OF SSME FUEL SIDE TURBINE FIRST-STAGE VANE

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Table 2
LOCATION OF HEAT-FLUX GAGES ON FIRST-STAGE VANE

Location	Wetted Distance Along Surface, %
10% span, suction surface	5, 22, 35, 70, 81
50% span, suction surface	5, 21, 34, 49, 65, 81
90% span, suction surface	5, 21, 48, 66, 83
10% span, pressure surface	6, 10, 36, 57, 77
50% span, pressure surface	6, 10, 36, 59, 76, 90
90% span, pressure surface	6, 11, 38, 61, 75

3. EXPERIMENTAL CONDITIONS

The experimental conditions for which the CUBRC measurements are to be performed will include a test point that can be compared with the MSFC blow-down rig results and another test point that can be compared with the UTRC large scale rotating rig results. Table 6 presents a tabulation of test conditions that are currently under consideration. The current plan is to obtain detailed data at the Reynolds number corresponding to the MSFC experiment and at the Reynolds number corresponding to the UTRC experiments.

Data from the UTRC measurement program are already available and are being reported in this particular session of this conference. The NASA MSFC data should become available at about the same time as the CUBRC data. The current plan is to begin running these experiments in the August-September 1990 time frame.

4. COMPONENT MEASUREMENTS TO BE PERFORMED

The measurements to be performed on the stage components (vane and blade) consist of the following: (a) for the first-stage vane, time-averaged surface pressure and heat-flux (Stanton number) distributions at 10%, 50% and 90% span; (b) for the first-stage blade, time-averaged surface pressure and heat-flux (Stanton number) distributions at 10%, 50% and 90% span. In addition, the unsteady envelope of surface pressure and heat flux on both pressure and suction surfaces of the blade. Phase-resolved surface pressure and heat flux for selected locations on the blade. Finally, time-averaged heat flux along the blade tip and on the stationary shroud; (c) for the second-stage vane, time-averaged heat flux at midspan for the pressure and suction surfaces.

The type of data noted above have previously been obtained at Calspan and/or CUBRC and have been reported in the literature, e.g. (4), (9) -(16). It is helpful for the purposes of code validation to have the predictions completed prior to obtaining the experimental results. Civinskas, et al. (17) have presented predictions with which the CUBRC measurements can be compared. Unfortunately, the geometry used in (17) is not the same as the engine hardware being

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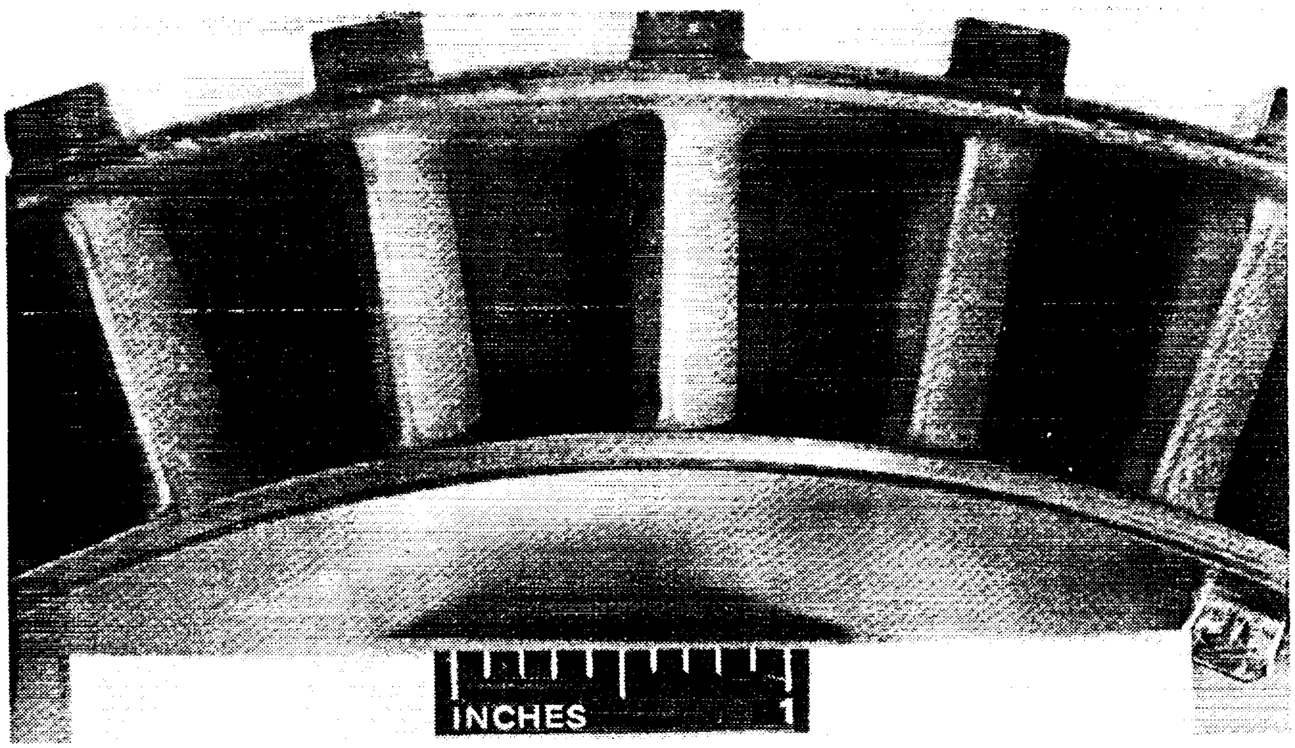
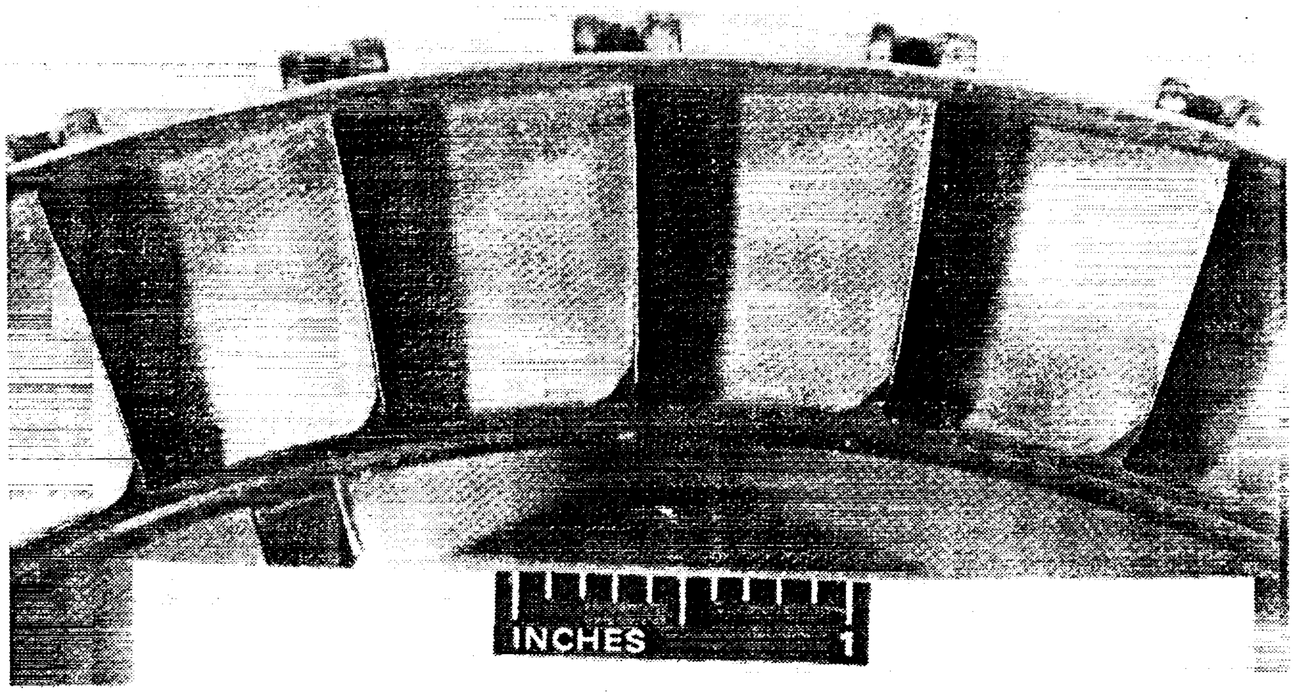


FRONT VIEW



REAR VIEW

Figure 9 (a) PHOTOGRAPH OF SSME FUEL SIDE TURBINE SECOND-STAGE VANE

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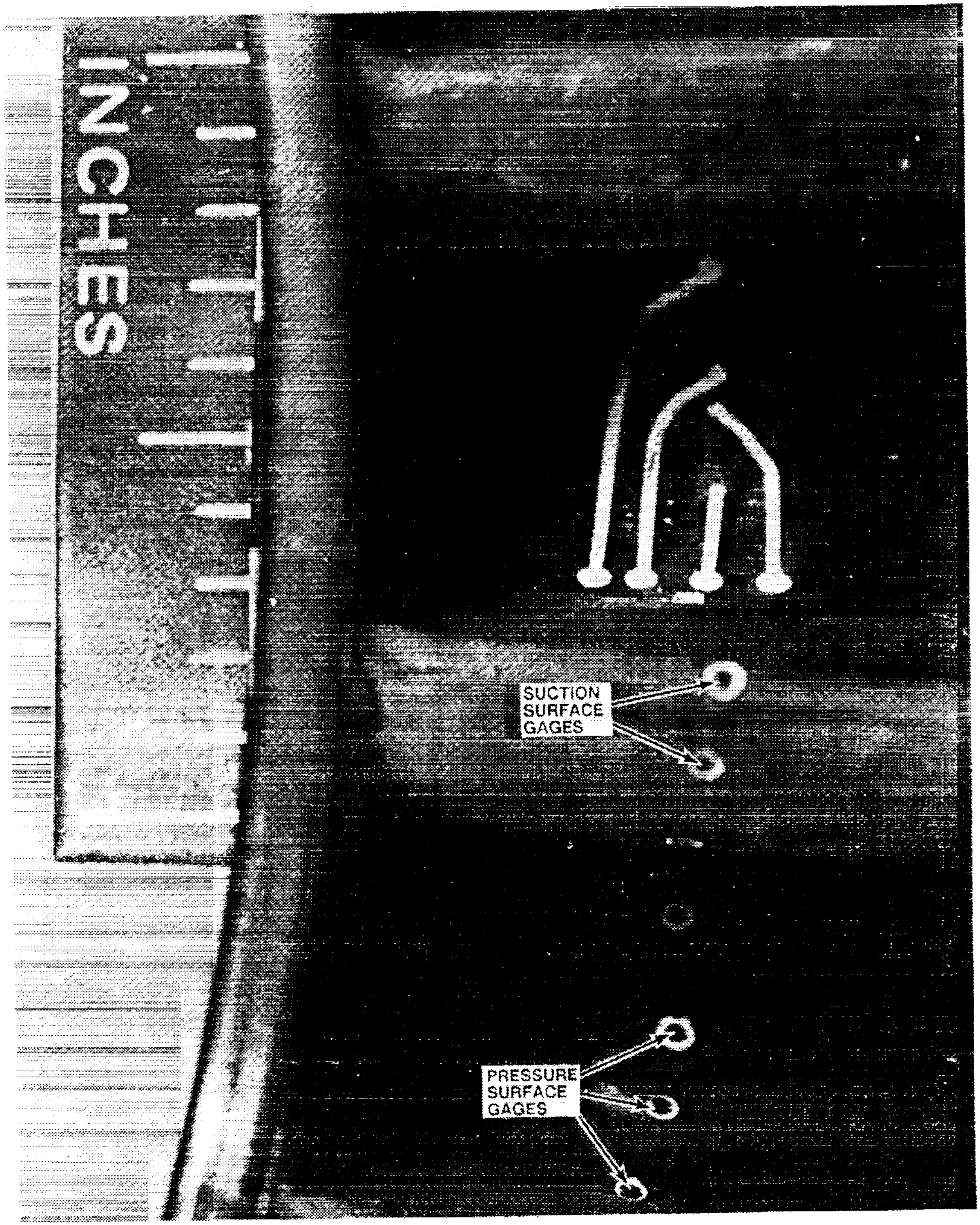

Figure 9 (b) PHOTOGRAPH OF BUTTON-TYPE GAGES ON INSTRUMENTED SECOND-STAGE VANE

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Table 3

LOCATION OF HEAT-FLUX GAGES ON SECOND-STAGE VANE

Location	Wetted Distance Along Surface, %
50% span, suction surface	0, 7.9, 21.7, 31.5, 51.6, 56.4, 66.8, 75.3, 79.2, 89.4
50% span, pressure surface	1.1, 7.3, 12.1, 36.9, 50.8, 61.4, 76.9

Table 4

LOCATION OF SURFACE-PRESSURE TRANSDUCERS ON FIRST-STAGE VANE

Location	Wetted Distance Along Surface, %
10% span, suction surface	5, 22, 35, 70, 81
50% span, suction surface	5, 12, 21, 34, 49, 65, 81
90% span, suction surface	5, 21, 48, 66, 83
10% span, pressure surface	6, 36, 77
50% span, pressure surface	6, 14, 36, 59, 76
90% span, pressure surface	6, 38, 75

Table 5

LOCATION OF SURFACE-PRESSURE TRANSDUCERS ON FIRST-STAGE ROTOR BLADES

Location	Wetted Distance Along Surface, %
10% span, suction surface	6.8, 19.0, 48.9, 73.7
50% span, suction surface	5.5, 12.1, 18.4, 25.4, 45.8, 60.3, 79.4
90% span, suction surface	6.0, 16.6, 77.8
10% span, pressure surface	5.0, 46.9, 62.6
50% span, pressure surface	4.3, 13.6, 43.6, 72.7
90% span, pressure surface	4.94, 40.3, 63.2

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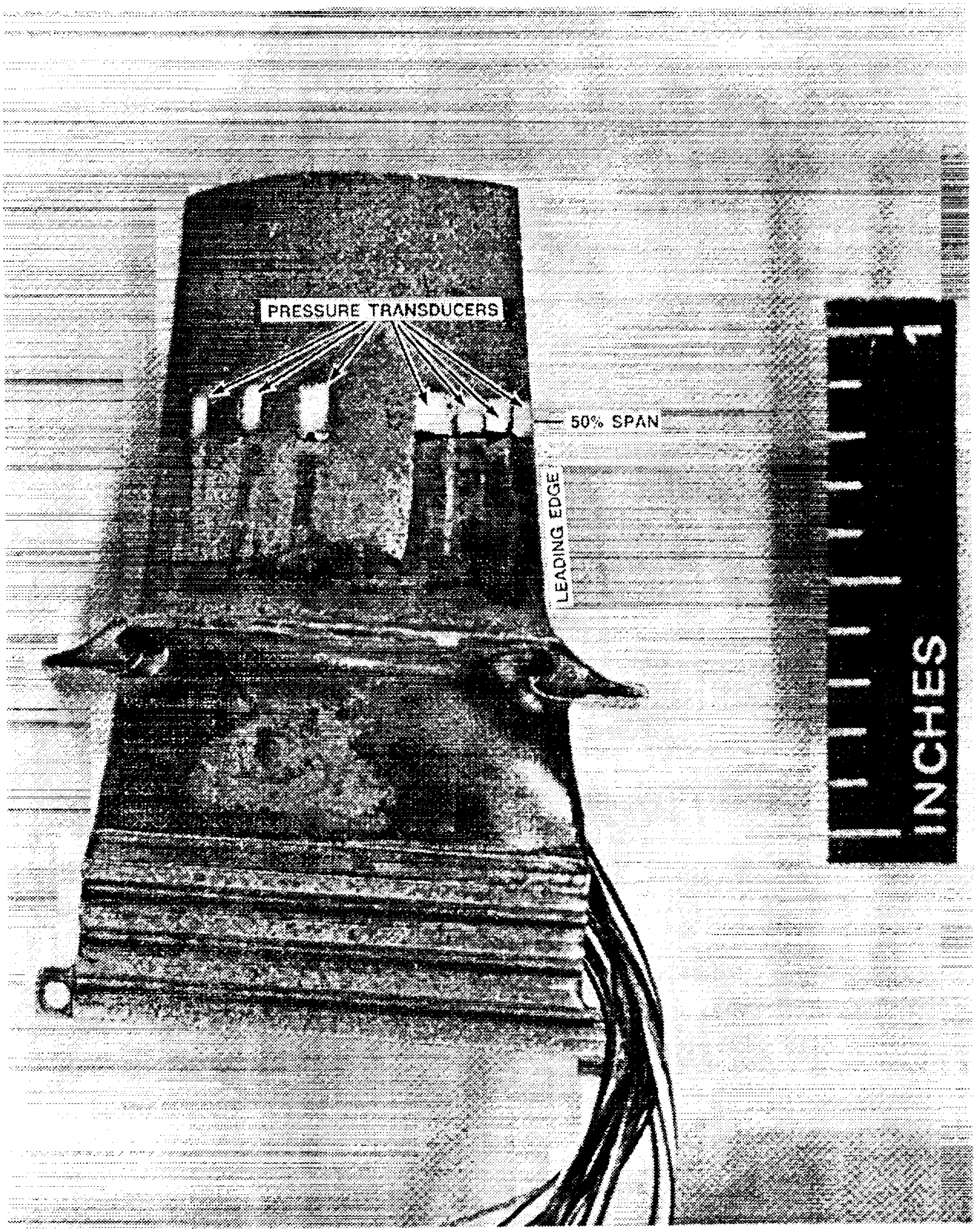

Figure 10 (a) PHOTOGRAPH OF SEVERAL PRESSURE TRANSDUCERS AT BLADE MIDSPAN ON FIRST-STAGE BLADE

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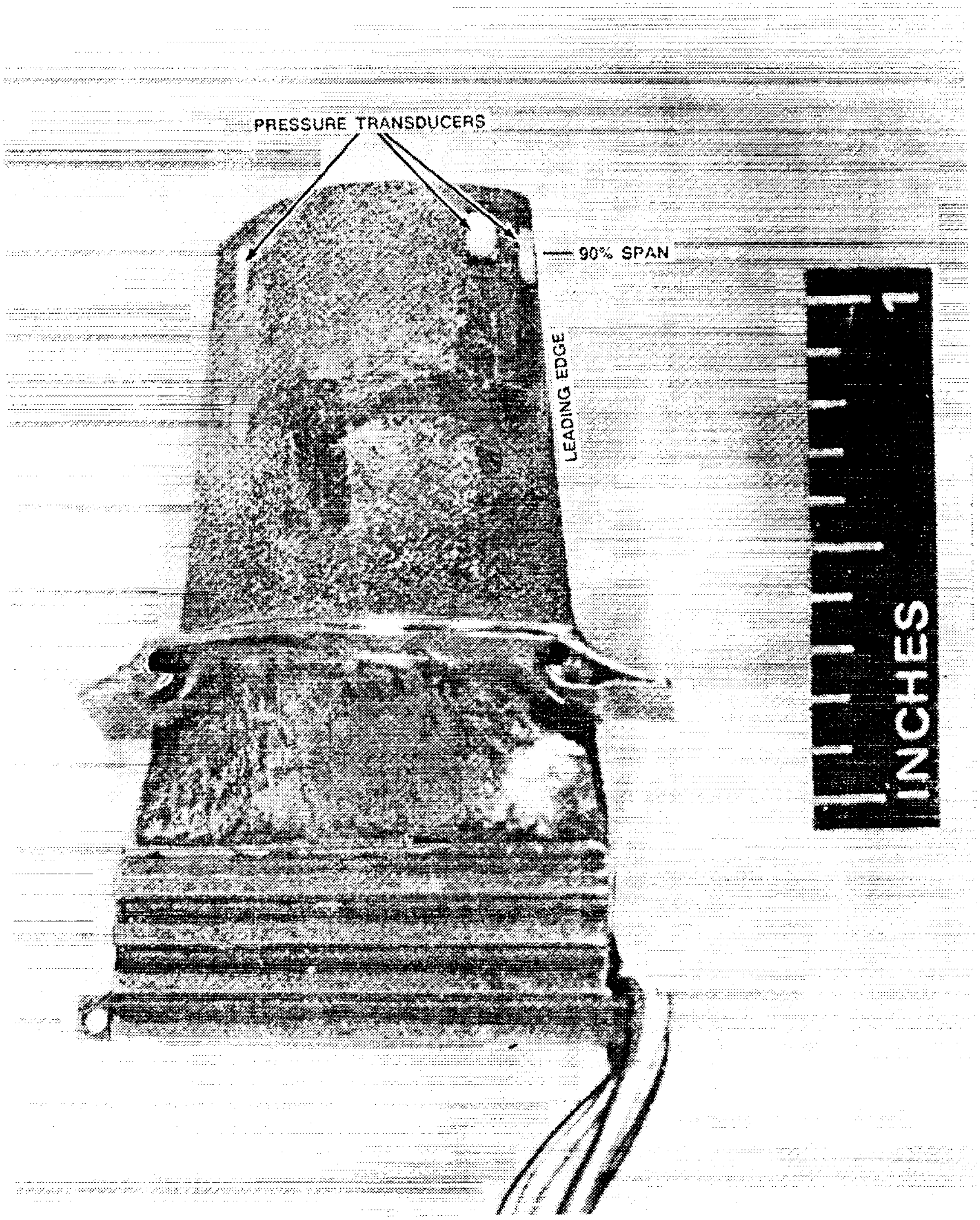

Figure 10 (b) PHOTOGRAPH OF PRESSURE TRANSDUCERS AT 90% SPAN ON FIRST-STAGE BLADE SUCTION SURFACE

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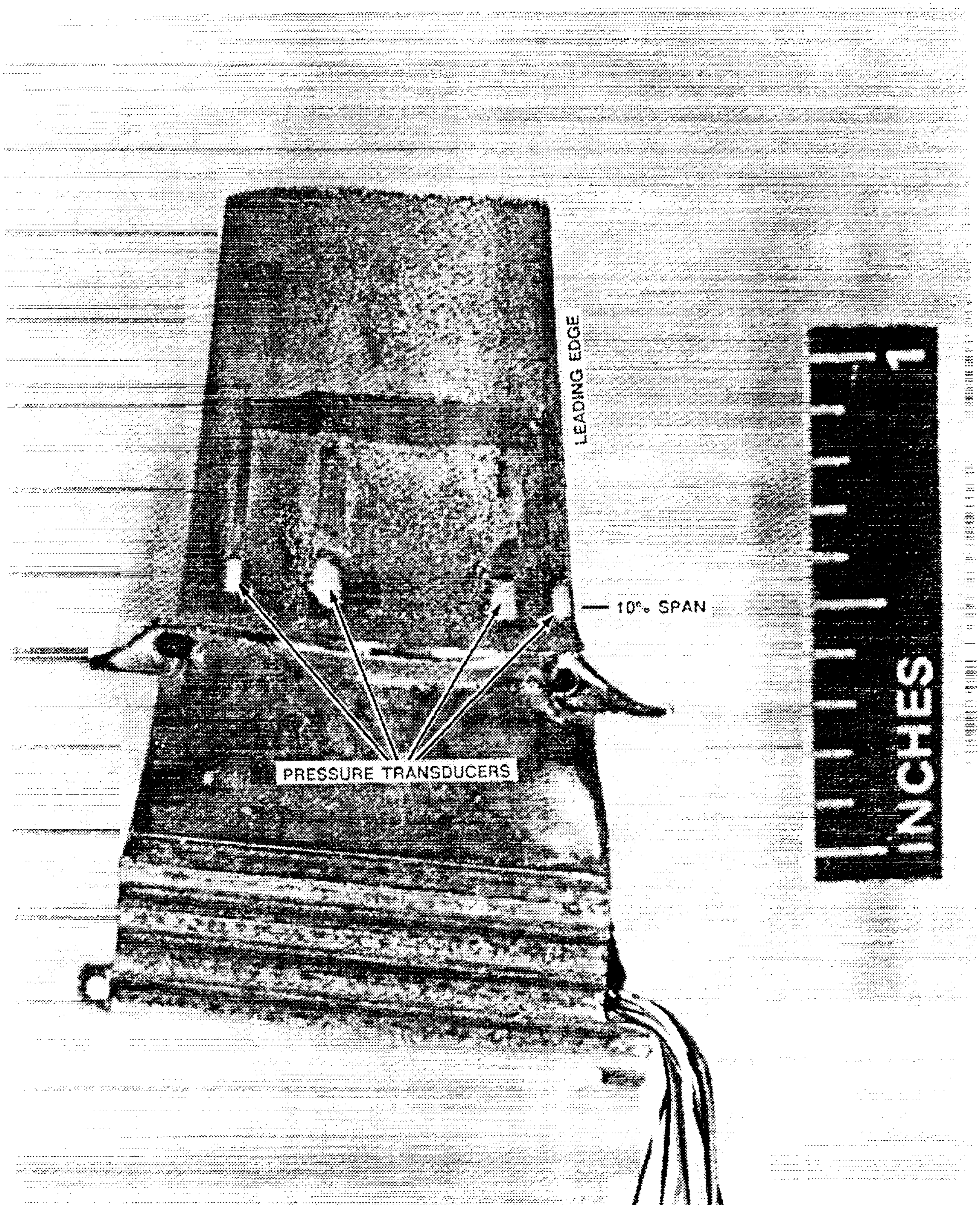

Figure 10 (c) PHOTOGRAPH OF PRESSURE TRANSDUCERS AT 10% SPAN ON FIRST-STAGE BLADE SUCTION SURFACE

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Table 6
TEST CONDITIONS FOR TURBOPUMP MEASUREMENTS

Group	Hardware	Flow Conditions at Inlet to 1st NGV				Stage Parameters	
		T_o, °R	P_o, psia	F.F.	Re_{chord}	Total to Total Press. Ratio	Rotor Speed, rpm
NASA MSFC	SSME	530	150	2.27	1.02×10^6	1.463	6891
UTRC	Large Scale	~ 530	~ 1 ATM	—	2.6×10^5	Rotor exit velocity triangles are correct	
CUBRC	SSME	1000	262	2.27	1.02×10^6	1.463	9292
CUBRC	SSME	1000	100	2.27	3.83×10^5	1.463	9292
CUBRC	SSME	1000	50	2.27	1.86×10^5	1.463	9292
CUBRC	SSME	1000	611	2.27	2.4×10^6	1.463	9292

$$N_{corr} = \frac{N_{phy}}{\sqrt{\theta}} = 6675 \text{ rpm}$$

$$\text{F.F.} = \dot{W}\sqrt{\theta}/\delta; \quad \theta = T_o/516; \quad \delta = P_o/14.7$$

used here, but to the authors knowledge there are no other predictions currently available that can be used for comparison. It is currently envisioned that the appropriate surface-pressure and heat-flux predictions with which the data described above can be compared will become available in the July 1990 time frame.

5. CONCLUSIONS

The SSME fuel-side turbopump turbine measurements that have been in the preparation stage at CUBRC for some time will be performed in the near future. Most of the instrumentation has been or is now being installed, the device to house the stage has been designed and constructed, and the shock-tunnel facility is operational.

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