NASA TECHNICAL MEMORANDUM

NASA TM X-2397

NASA TM X-2397

DETERMINATION OF NORMAL-SHOCK POSITION IN A MIXED-COMPRESSION SUPERSONIC INLET

by Miles O. Dustin, Gary L. Cole, and Robert E. Wallhagen

Lewis Research Center
Cleveland, Ohio 44135

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • NOVEMBER 1971

DETERMINATION OF NORMAL-SHOCK POSITION IN A MIXED-COMPRESSION SUPERSONIC INLET

by Miles O. Dustin, Gary L. Cole, and Robert E. Wallhagen

Lewis Research Center

SUMMARY

Some criteria for determining inlet normal shock position from wall static pressure profiles were investigated. By using methods investigated in this report, it should be possible to control an inlet with the shock closer to the throat. This, generally, increases total pressure recovery and decreases distortion of the total pressure profile at the diffuser exit. Also, it may be possible to reduce the complexity of normal shock control feedback signal schedules that must be functions of such things as flight Mach number and angle of attack.

Static pressure profiles were measured in an axisymmetric, mixed-compression inlet at three circumferential locations by means of closely spaced cowl wall static taps. Tests were confined to the inlet design Mach number.

Sensors at four circumferential locations may be necessary to handle variations in angle of attack and side slip because of shock assymetry in the inlet.

One criterion would determine shock position by comparing each individual wall static to a reference pressure. Because static pressures can be affected by angle of attack variations, a total pressure measurement would better serve as the base for the reference pressure. This criterion was applied to the profiles presented in the report and fairly consistently indicated shock position two tap spacings downstream of the actual shock position.

Special probes were investigated which allowed the static taps to be elevated above the cowl surface boundary layer. One probe gave profiles which had a steeper pressure gradient in the vicinity of the shock than did the profiles measured by the wall statics. The error in shock position, as determined by the criterion of comparing the wall static pressures with a reference pressure, was reduced from an average of two to an average of one tap spacing.

INTRODUCTION

For high propulsion system efficiency, a mixed-compression supersonic inlet must operate with the normal shock near the inlet throat. This maximizes total pressure recovery and minimizes distortion of the total pressure profile at the diffuser exit. A margin is required, however, to keep the shock from moving upstream of the throat and causing an inlet unstart when the inlet is subject to airflow disturbances. The size of the margin depends largely on a knowledge of the exact position of the normal shock and available normal shock control capability. If the uncertainty of shock position is great, the required stability margin must be large. This results in lower pressure recovery and higher distortion and, hence, poorer propulsion system performance.

A static pressure measured downstream of the normal shock is often used in supersonic inlet control systems to measure shock position. This measurement is subject to inaccuracies due to such things as operating the inlet at off-design Mach numbers, altitudes, and angles of attack. Therefore, the pressure signal representing shock position must have biases as functions of these variables to maintain any degree of accuracy. If the shock position could be measured directly, uncertainties due to off-design conditions could be minimized and a smaller stability margin would be possible.

A program is being conducted at Lewis to study various methods for determining normal shock position from wall static pressure profiles in the vicinity of the shock.

For inlets with internal compression, the ideal static pressure profile occurs as shown in figure 1. The minimum supersonic flow Mach number occurs at the inlet throat. Downstream of the throat the Mach number increases, and static pressure decreases as

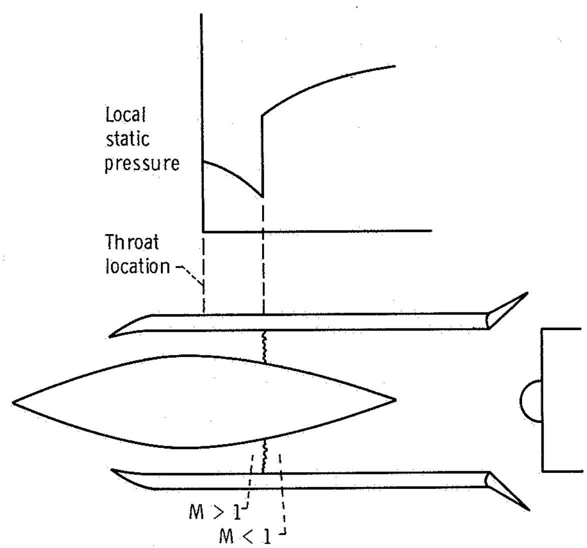


Figure 1. - Ideal static pressure profile in vicinity of normal shock.

area increases. At the normal shock there is a discontinuous rise in pressure. Downstream of the shock the pressure continues to rise, since the flow is subsonic, and area continues to increase. In a real inlet, the pressure profile can be measured by a series of closely spaced static pressure taps.

Figure 2 shows typical cowl-wall static-pressure profiles for a mixed-compression inlet for three different shock positions. Each profile was obtained with the leading edge of the shock just downstream of a tap, as noted in the figure. The non-ideal nature of the profiles is due to such things as shock - boundary-layer interaction, oblique shock reflections, and a finite shock train thickness. Although the pressure rise across the shock is not discontinuous as in the ideal case, there is a rather large pressure gradient in the vicinity of the shock. This gradient may be used to advantage for determining shock position.

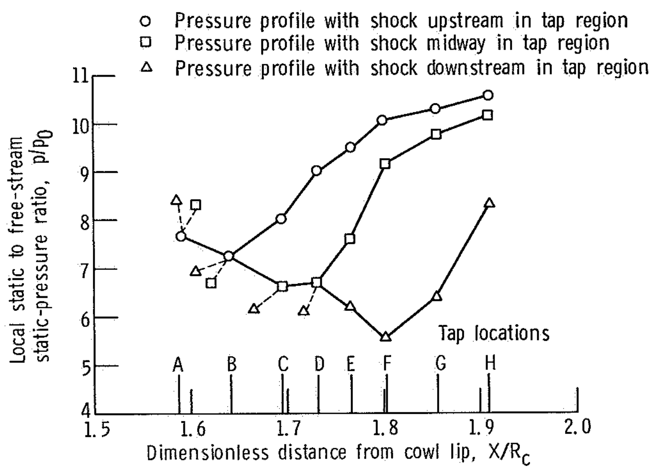


Figure 2. - Typical inlet cowl static-pressure profiles in vicinity of inlet throat with normal shock in three different locations. Free stream Mach number, 2.5; angle of attack, 0^o.

Two schemes based on wall static pressure distribution were developed before this program. One scheme using flueric logic means for deducing shock position is reported in reference 1. The other used electronic logic to determine shock position and is reported in reference 2. Both of these sensors were evaluated in an axisymmetric mixed-compression inlet with a translating center body. Tests were run only at 0 angle of attack and at design Mach number. Other shock sensors which used wall static pressure profiles are reported in references 3 and 4. Both of these schemes compare the static pressure levels along the inlet with a reference pressure to obtain shock position.

This report presents cowl wall static pressure profiles for various shock locations. The data were taken over a range of angles of attack at design Mach number. The inlet used was an axisymmetric mixed-compression inlet. The application of various shock position criteria to these profiles is then discussed and evaluated. Also included is an evaluation of elevated wall static pressure probes designed to minimize the effect of boundary layer on static pressure measurements.

SYMBOLS

A, B, C, D, E, F, G, H	throat static-pressure taps (see figs. 4 and 5)
M	Mach number, dimensionless
P	total pressure, N/cm^2
p	static pressure, N/cm^2
R_c	cowl lip radius, 23.7 cm
s	assumed shock position
X	distance from cowl lip, cm

Subscripts:

0	free-stream condition
th	throat station ($X/R_c = 1.426$)
56	throat exit static location 56.1 cm downstream of the cowl lip

EXPERIMENTAL WIND TUNNEL TESTS

Apparatus

The static pressure profiles presented in this report were measured during tests conducted on an axisymmetric mixed-compression inlet designed for Mach 2.5. Approximately 60 percent of the supersonic area contraction occurred internally at the design condition.

An isometric view of the inlet is shown in figure 3. The inlet capture area was 1760 square centimeters based on a cowl lip radius of 23.66 centimeters. A plate with a choked orifice was located at the diffuser exit station. The inlet was equipped with a translating centerbody and six overboard bypass doors, two of which are shown in figure 3. The bypass doors were located symmetrically around the inlet just upstream of the diffuser exit. The normal shock position was varied by varying the bypass flow. The bypass door exits were choked.

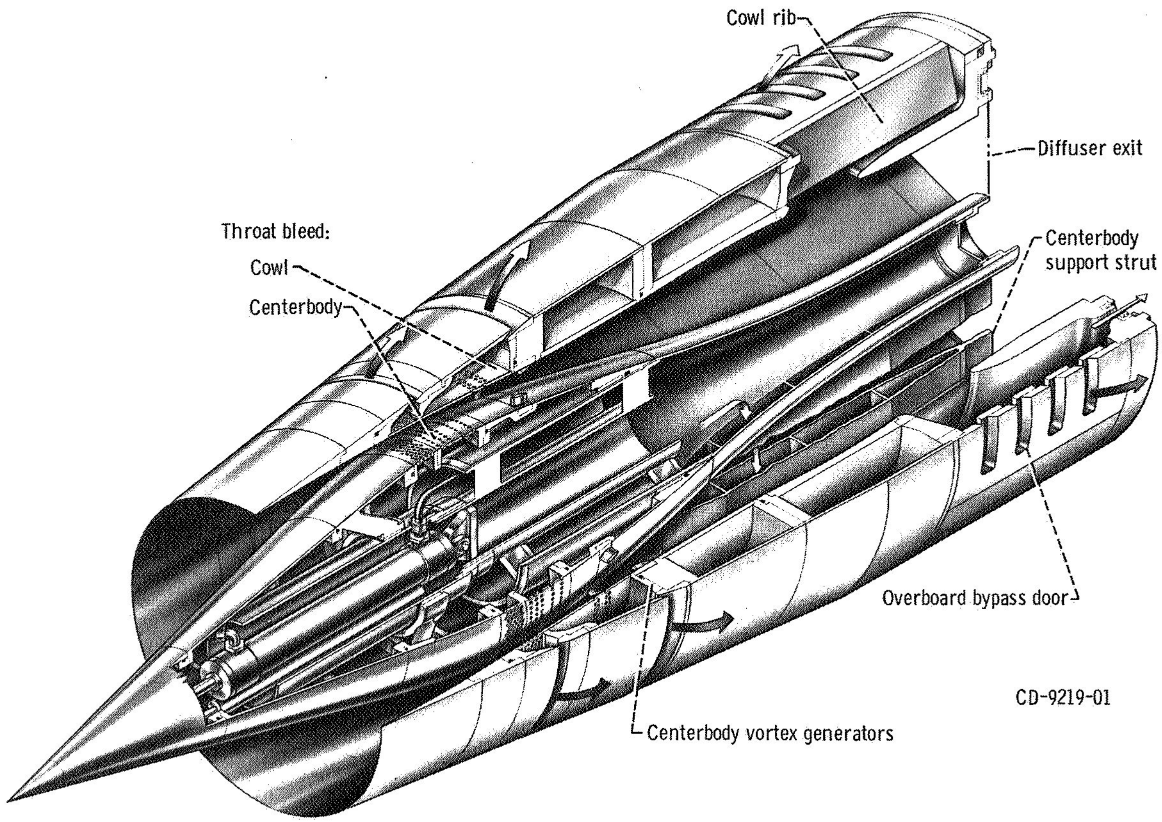


Figure 3. - Isometric view of inlet.

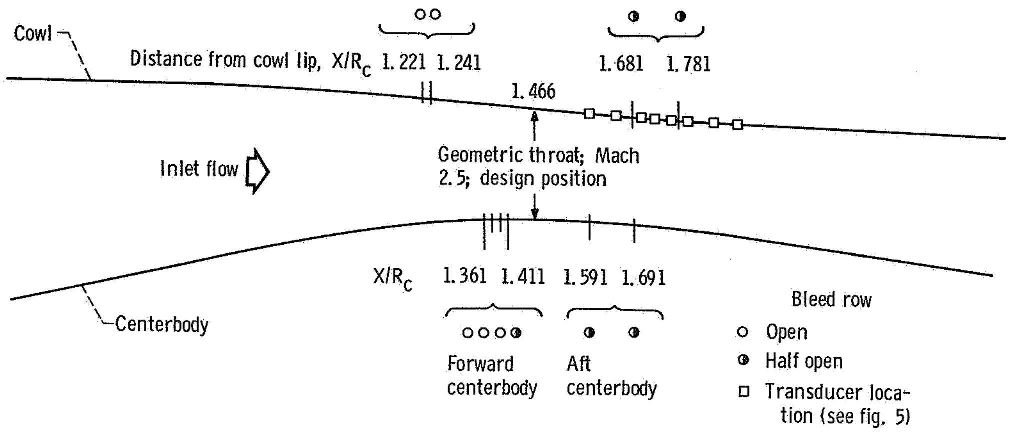


Figure 4. - Inlet bleed configuration. Bleed hole diameter, 0.3145 centimeter.

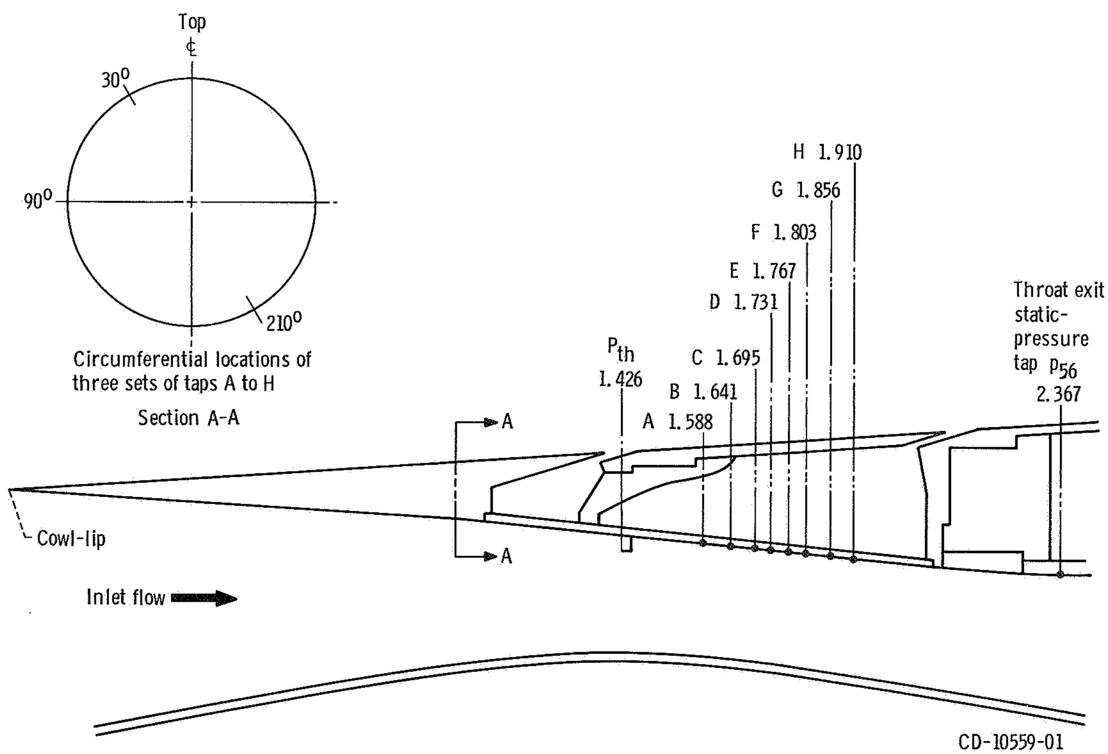


Figure 5. - Inlet pressure measurement locations. Dimensions from cowl lip, X/R_C.

The inlet had porous bleed regions on both the centerbody and cowl surfaces, as indicated in figure 3. The exact bleed configuration is shown in figure 4. The bleed regions improved inlet performance and stability. Information regarding optimization of the bleed patterns can be found in reference 5, and steady-state performance of the inlet is reported in reference 6.

The locations of pressure measurements used in the investigation are shown in figure 5. Three rows of eight static pressure taps each were used to measure cowl wall static pressure profiles at three different circumferential locations. The circumferential locations were at 30°, 90°, and 210° measured counterclockwise from the top centerline looking downstream. The taps of each set were located within $\pm 5^{\circ}$ of the nominal circumferential location. The axial locations of the three sets of taps were identical and are shown in figure 5. Since the inlet throat remained fixed relative to the translating centerbody, the pressure taps were properly located with respect to the throat only when the centerbody was at its Mach 2.5 design position. For this reason, tests were conducted only at Mach 2.5 (Reynolds number 4.5×10^{6} based on cowl-lip diameter) in the Lewis 10- by 10-Foot Supersonic Wind Tunnel.

Two other pressure measurements that were used during the investigation are P_{th}, a throat total pressure, and p_{56}, a throat exit static pressure located 56 centimeters from the cowl lip. Both are shown in figure 5. All pressure measurements were made by means of strain-gage electronic pressure transducers.

Static Tests

Procedure. - The wind-tunnel tests were conducted by positioning the normal shock within the range of the static-pressure taps and reading the static pressures. The shock was then moved to a new location and new pressure measurements were made. This procedure was repeated for several shock positions at angles of attack of -1°, 0°, $+1^{\circ}$, and $+2.5^{\circ}$. The static pressures were recorded at all three circumferential locations, 30°, 90°, and 210°.

Test results. - Three sets of static pressure profile data, each for a constant value of throat exit static pressure p_{56} (see fig. 5), are presented in figures 6 to 8. The profiles in figure 6 were taken at the highest value of p_{56} or with the shock in the tap region nearest the inlet throat. The profiles in figure 7 were taken at a lower value of p_{56} corresponding to a midtap region shock position. And figure 8 profiles are for the lowest values of p_{56} with the shock near the downstream end of the tap region. The dashed line of each profile indicates the supersonic profile or the profile that occurs when the normal shock is downstream of tap H. The solid line indicates the profile that is obtained when the shock is moved up into the tap region. The leading edge of the shock is assumed to be at the point where the measured profile begins to deviate from the supersonic profile

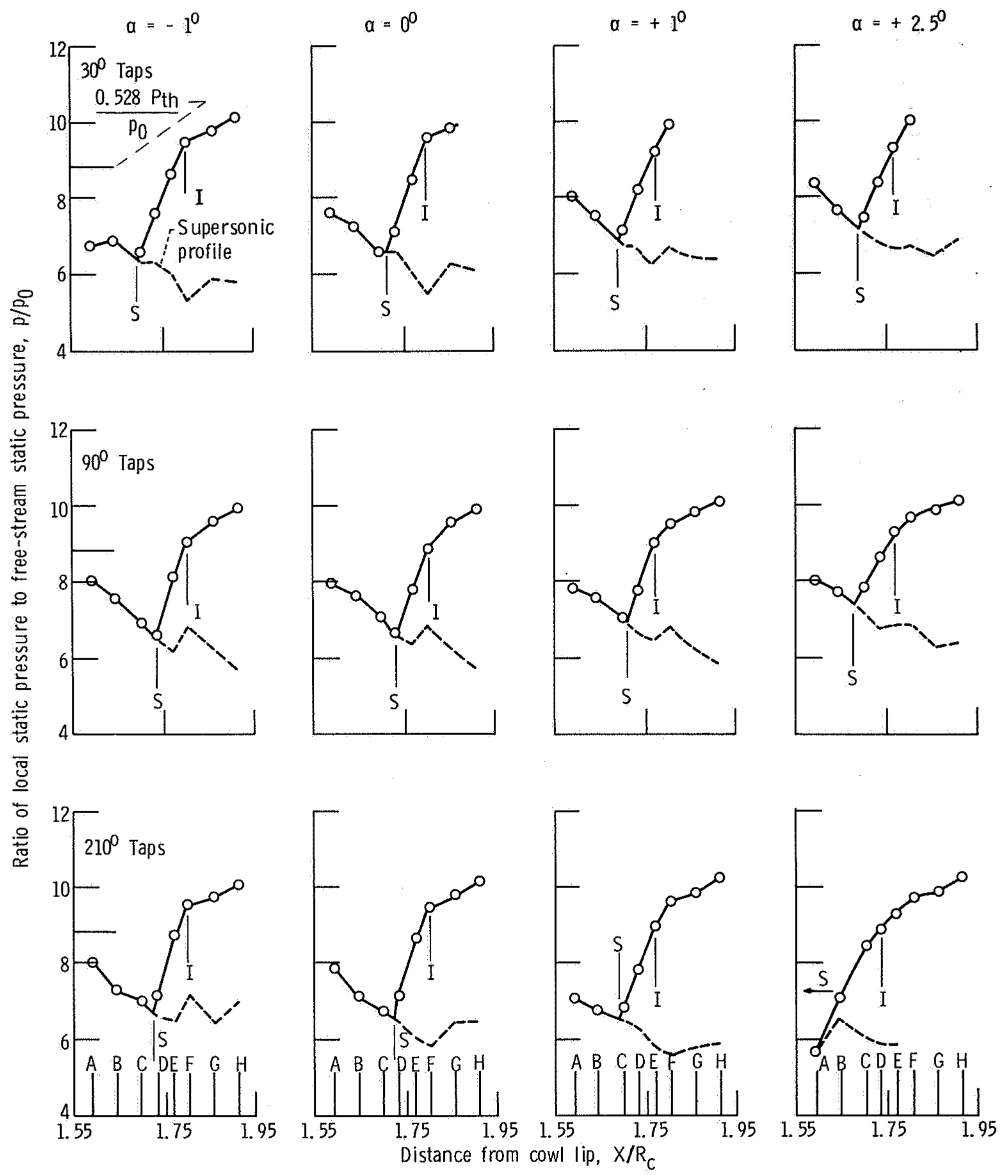


Figure 6. - Static-pressure profiles with normal shock positioned in upstream region of static tap location. "S" marks the actual shock position while "I" marks the indicated shock position using 0.528 of the throat total pressure for reference.

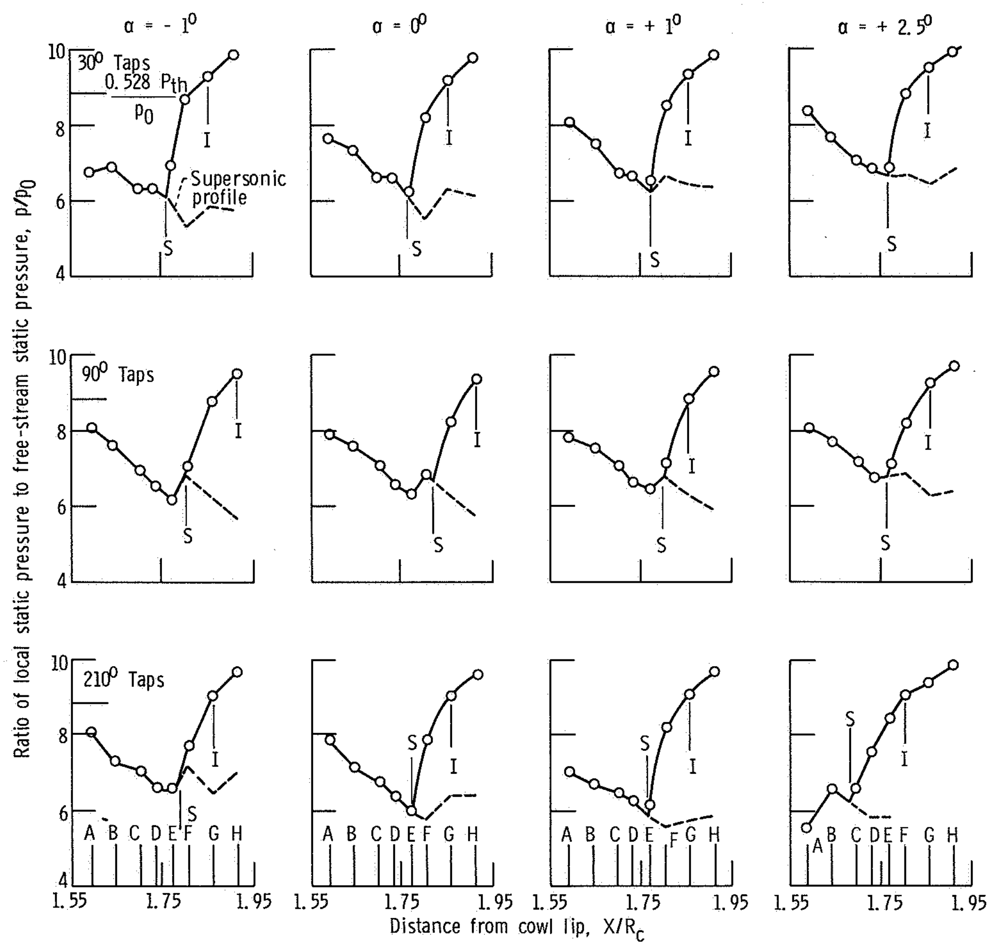


Figure 7. - Static pressure profiles, with normal shock positioned midway in region of static tap location. "S" marks the actual shock position; I marks the indicated shock position using 0.528 of the throat total pressure for a reference.

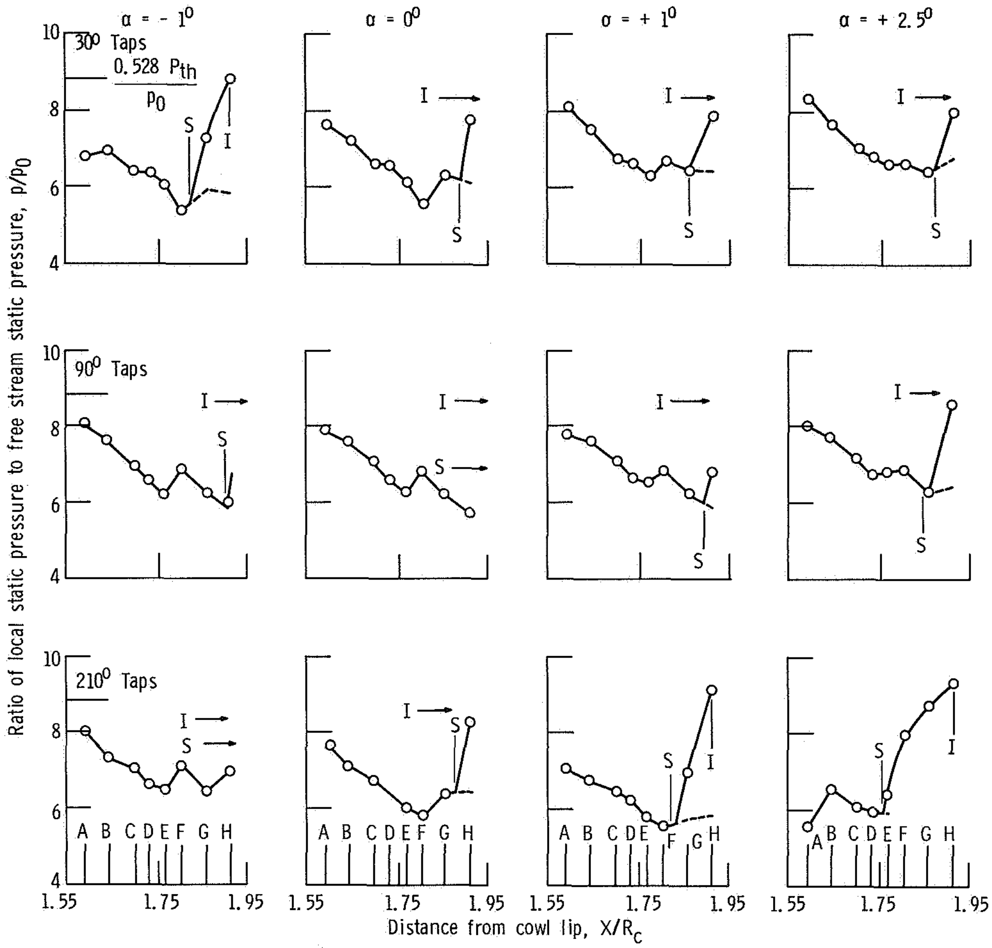


Figure 8. - Static pressure profiles with normal shock positioned in downstream region of static tap location. "S" marks the actual shock position; "I" marks the indicated shock position using 0.528 of the throat total pressure for a reference.

(dashed line). The measured profile coincides with the supersonic profile upstream of the shock leading edge. The assumed location of the shock leading edge is denoted by an s in the figure.

DISCUSSION

In determining the criterion to use in measuring normal shock position, one must consider the effect of angle-of-attack changes and of off-design Mach number operation on the cowl static pressure profiles. Unfortunately, the effect of off-design Mach number operation on the cowl static pressure profiles could not be considered because, in this particular inlet, throat position shifts with respect to the cowl static pressure taps as the centerbody is moved to accommodate off-design Mach numbers. Angle-of-attack changes introduce two effects on the static-pressure profiles: (1) a change in the shape of the profile and (2) an angularity and nonplanar condition of the normal shock leading edge with respect to a plane perpendicular to the inlet longitudinal axis.

Effect of Angle of Attack on Static Pressure Profile

The shape of the pressure profile can be altered by uncancelled oblique shocks that pass through the inlet throat. As angle of attack varies, the impingement points of the oblique shocks will vary. This, in turn, results in changes of the Mach number distribution and, hence, static-pressure distribution along the cowl surface. It is possible that these shocks can create conditions that might result in a false indication of shock position, depending on the criterion being used. A specific example of this will be given in the section Selection and Evaluation of Shock Position Criteria. It can be seen from the profiles of figures 6 to 8 that the profiles are altered considerably by angle-of-attack changes - particularly, the supersonic portion of the profile. For example, in figure 6, the supersonic profile for the 30° taps changes shape completely as the inlet goes through the angle of attack range. A similar effect can be noted at the 90° and 210° stations.

Effect of Angle of Attack on Shock Location

As the inlet goes through angle of attack, the normal shock does not remain planar or perpendicular to the inlet longitudinal axis. This effect was also noted by Hurrell, et al. when investigating ram-jet-engine inlet controls (ref. 7). Figure 9 shows the extent of this effect for the particular inlet tested. This figure was constructed by plotting the shock positions determined in figures 6 to 8 as a function of angle of attack. Even at zero angle of attack it is seen that the normal shock leading edge does not appear to be planar. This is probably due to slight variations in annular area between the cowl and centerbody. Measuring the terminal shock position at only one circumferential location does not appear to furnish a measure of the inlet's true stability condition. For example, if only one shock sensor were used, say at the 30° station, it is clear from the 2.5° angle-of-attack data (fig. 9) that the shock at the bottom of the inlet could go forward of the throat, even when the shock sensor indicated a stable condition. To prevent this condition, it may be necessary to have sensors at multiple circumferential stations and to control the inlet from the sensor that indicates the most forward shock position. Four sensors, located 90° apart, should be sufficient to cover changes in angle of attack and sideslip conditions.

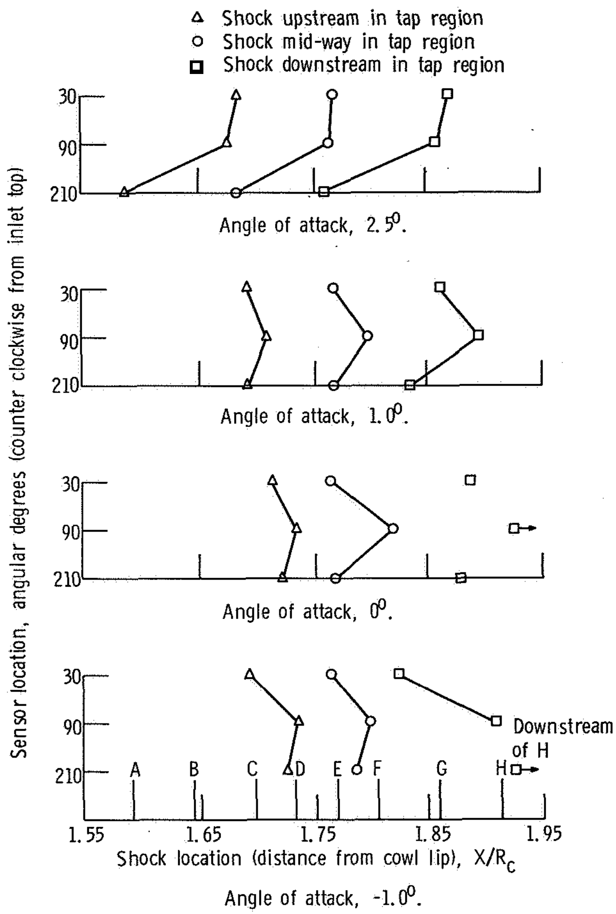


Figure 9. - Effect of angle of attack on angle of normal shock plane as measured at the 30°, 90°, and 270° stations.

SELECTION AND EVALUATION OF SHOCK POSITION CRITERIA

A normal shock position sensor criterion can only be selected after carefully studying the wall static-pressure profiles and determining the effect of the entire range of inlet operating conditions upon the profiles.

Two different criteria that have been used in references 1 to 4 will now be evaluated by means of the pressure profiles of figures 6 to 8.

Minimum in Pressure Profile

One criteria for establishing the normal shock position would be to find the minimum point in the wall static-pressure profile. Ideally, this can be done by comparing three consecutive pressure taps and determining whether the center tap is lower than the other two. If it is, then this represents a minimum in the profile, and the shock occurs between the center tap and the next tap downstream. The electronic and fluidic sensors described in references 1 and 2 used this criterion. A careful study of the profiles of figures 6 and 8 shows that a scheme of measuring shock position by means of locating the pressure minimum is not without problems. The irregularities in the pressure profiles create false minimums that can cause false shock location indications. For example, the supersonic profile at the 210° location exhibits two positive slopes when the inlet is at -1° angle of attack. The electronic sensor used a biasing scheme to eliminate the false minimums. Since the pressures were measured by electronic transducers, electrical voltages were added to or subtracted from the pressure readings before they entered the logic circuit. Thus, the signals as seen by the logic circuit were smooth and without false minimums. This scheme was used satisfactorily during both static and dynamic wind-tunnel performance evaluation of the inlet model. It should be noted, however, that tests were run only at zero angle of attack and inlet design Mach number.

Because of the alterations that occur in the pressure profile as the inlet goes through angle of attack, biases might have to be scheduled as functions of angle of attack. In addition, operation at off-design Mach numbers would undoubtedly compound the problem. Thus, this criterion would be impractical for a flight application.

Pressure Level Compared With Reference Level

Another shock sensing criterion relies on the pressure just downstream of the normal shock being higher than a reference pressure. The pressures measured by several wall static-pressure taps would be compared with another tap just upstream of the sensor wall taps. Ideally, those pressures downstream of the shock would be higher than the reference pressure, and those upstream of the shock would be lower. The sensor described in reference 3 required that three consecutive static pressures be higher than the reference pressure before a shock position would be registered. The sensor used the inlet throat static pressure as a reference pressure. This sensor was also tested only for zero angle-of-attack conditions.

Since static pressures can be significantly altered by changes in angle of attack, as indicated by the profiles of figures 6 to 8, it is felt that a static pressure is not a good choice for use as a reference pressure. If this criterion is applied to the profile of figures 6 to 8 and tap A is used as the reference pressure, for example, it is seen that incorrect shock position determination can result. In particular, the sensor at the 210° will always indicate the shock to be upstream of tap B, regardless of shock position, when the inlet is at a 2.5° angle of attack.

Although neither of the specific criterions just discussed appears to have application over a full range of flight conditions, a criterion that does appear suitable is the comparison of each static tap pressure with a reference total pressure. The reference pressure must be carefully selected such that those wall static pressures downstream of the shock are higher than the reference pressure and those static tap pressures upstream of the shock are lower than the reference pressure. This scheme would be fairly simple to implement because it does not require complicated logic. One pressure that might be considered for use as a reference pressure would be a throat total located just ahead of the static tap region. An example of this would be the pressure P_{th} shown in figure 5. Then static taps compared to about 0.528 P_{th}, should theoretically have higher values downstream of the shock (subsonic) and lower values upstream of the shock (supersonic).

Using P_{th} as the reference pressure has the advantage over a static pressure in that P_{th} is practically invariant with angle of attack. One disadvantage is the requirement to obtain a value of about 0.528 P_{th}. Electronically, this can be accomplished quite easily. Fluidically, this may not be quite so simple, but would be desirable so that it might be possible to implement the scheme using differential pressure switches. (This might require specially developed switches to meet flight requirements.)

The criterion of comparing the static taps to 0.528 P_{th} can be applied to the profiles of figures 6 to 8. The normalized reference value (0.528 P_{th}/P_o) for these tests is about 8.85 and is indicated in the figures. When this is done, it is found that the criterion gives shock indications which are usually downstream of the shock leading edge by two or three tap spacings. This error is primarily due to the fact that the normal shock does not occur as a pure discontinuity, but rather is a shock train spread out over a finite distance. This error may not be serious. And, since the error is fairly consistent, the shock sensor could be biased by this amount so that it would always be within ±1 tap spacing.

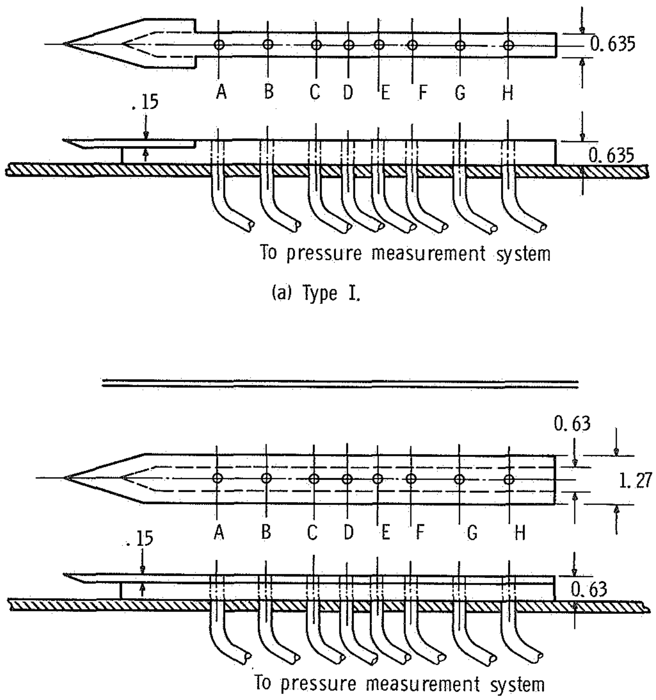
ELEVATED STATIC PRESSURE PROBES

When the normal shock intersects the wall of the inlet an interaction occurs with the boundary layer. This interaction causes separation of the boundary layer at the intersection of the shock and the wall (ref. 8). The shock then develops bifurcated ends instead of intersecting the wall normally. Because of this interaction with the wall, the pressure distribution measured at the wall does not exhibit the very abrupt pressure rise one would obtain my measuring the static pressure profile in the center of the duct.

Special elevated wall static pressure probes were built which extended away from the wall. By measuring the static pressure out of the boundary layer, it was felt that a more abrupt static-pressure profile could be obtained. Thus, a more accurate shock position determination could be made using the criterion of comparing the static tap pressures to a reference.

Probe Design

The two elevated probe designs shown in figure 10 were built and tested. The height of the probes was established from earlier boundary-layer investigations on the inlet. The blade on top of the probe is to separate the boundary layer and keep it below the top surface of the probe. The two styles built and tested were designated type I and type II. The only difference between the two types is the length of the blade as shown in the figure.



(a) Type I.

(b) Type II.

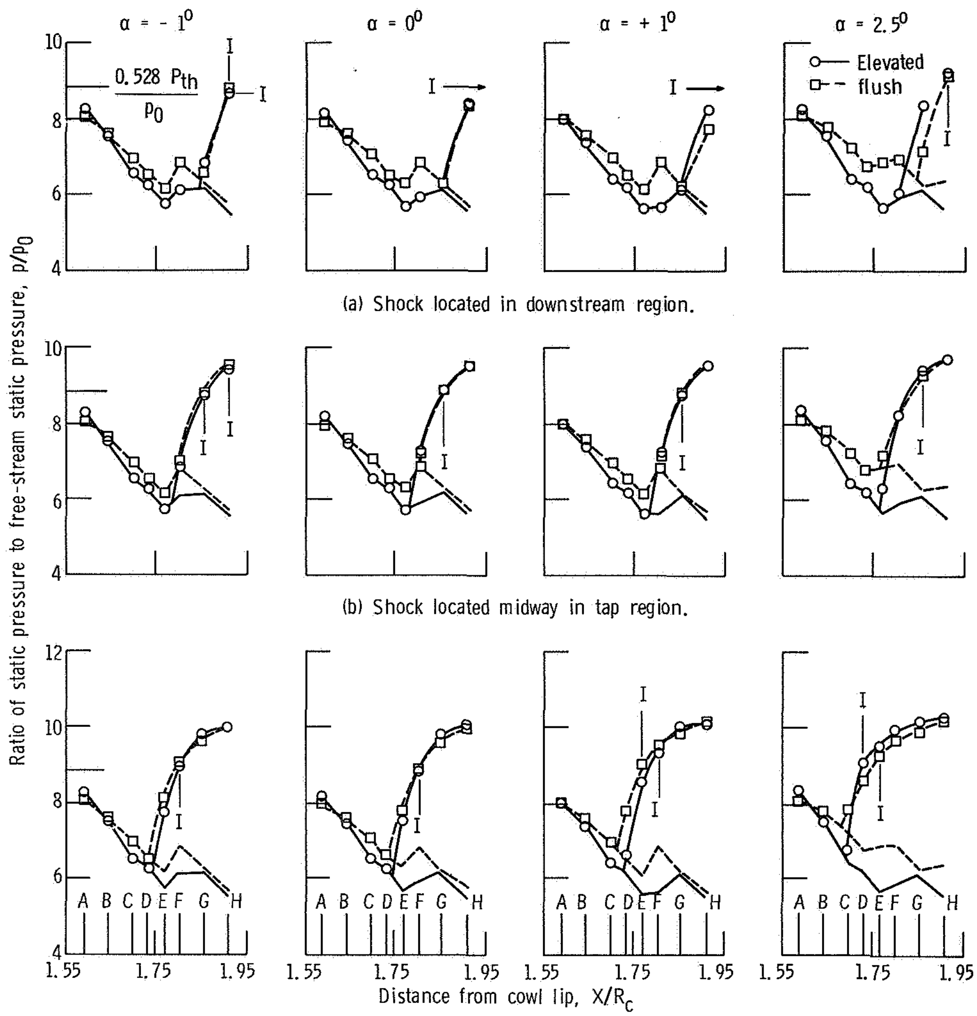
Figure 10. - Elevated wall static-pressure probes (dimensions are in cm).

Test Description

Tests on both elevated probe designs were conducted by installing the probes in the inlet in the place of previously tested flush static probes. The results of the tests were then compared to data taken by the flush probes under similar angles of attack and throat exit static pressures p_{56}. Only steady-state tests were conducted.

Test Results

The wall static pressure profiles determined by the elevated probes are compared with similar profiles from flush static pressure taps in figures 11 and 12. The profiles



(a) Shock located in downstream region.

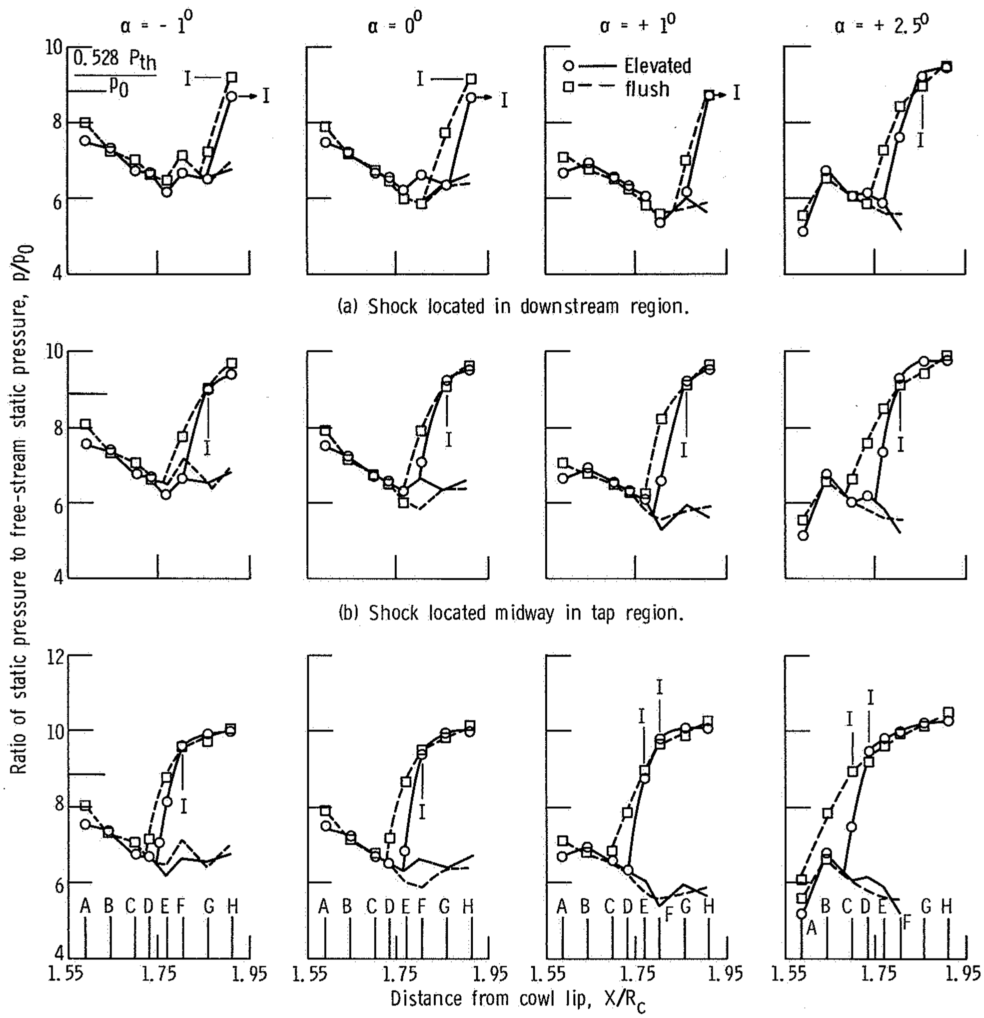
(b) Shock located midway in tap region.

(c) Shock located in upstream region.

Figure 11. - Comparison of static-pressure profiles measured with elevated taps with those measured with flush taps. Probe type I at 90° station.

were established with the shock located in (1) the downstream portion of the tap region, (2) midway in the tap region, and (3) the upstream portion of the tap region. These conditions are shown in the top center and lower rows of curves, respectively. The profiles are compared at four angles of attack, -1.0°, 0°, +1.0°, and 2.5°. Figure 11 compares profiles as determined by the type I probe located at the 90° station with profiles taken by flush taps at the same location. This probe has the short blade. Very little difference in pressure gradient is shown in the curves for angles of attack of -1.0°, 0°, and +1.0°. Some improvement is shown, however, when the angle of attack is increased to 2.5°.

Figure 12 shows considerable improvement in steepness of the pressure profiles taken by the type II probe. When the criterion of comparing the static taps (elevated



(c) Shock located in upstream region.

Figure 12. - Comparison of static-pressure profiles measured with elevated taps with those measured with flush taps. Probe type II at 210° station.

probes) to 0.528 P_{th} are applied to the profiles of figure 12, the error in shock position is usually 1 tap. Thus, some improvement is achieved by making the static pressure measurements out of the boundary layer.

SUMMARY OF RESULTS

A study was made of wall static pressure profiles to determine criteria to be used in determining the normal shock position in a supersonic inlet. From this investigation, the following results were obtained.

1. Sensors which depend upon predetermined wall static pressure profiles will not perform satisfactorily at angles of attack because changes in the inlet angle-of-attack cause the shape of the static pressure profiles to change.

2. Sensors at four circumferential locations may be necessary to accommodate variations in angle of attack and side slip because of shock assymetry in the inlet.

3. By using a shock position criterion whereby the individual wall static pressures are compared with a suitably chosen reference pressure, position indication within one tap is possible.

4. Special probes that elevated the wall static taps above the boundary layer, thus avoiding the shock boundary layer interaction, measured a steeper pressure gradient across the normal shock than did the flush taps.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, June 22, 1971,
720-03.

REFERENCES

1. Griffin, William S.: Design and Performance of a Flueric Shock Position Sensor For a Mixed-Compression Supersonic Inlet. NASA TM X-1733, 1969.

2. Cole, Gary L.; Neiner, George H.; and Crosby, Michael J.: Design and Performance of a Digital Electronic Normal Shock Position Sensor for Mixed-Compression Inlets. NASA TN D-5606, 1969.

3. Holmes, Allen B.; Gehman, Stacy E.; Egolf, David; and Lewellen, Michael: Flueric Normal Shock Sensor. Rep. TM-69-9, Harry Diamond Lab., (NASA CR-107688), Apr. 1969.

4. Johnson, Elmer G.: Control Apparatus. Patent No. 3,460,554, United States, Aug. 1969.

5. Cubbison, Robert W.; Meleason, Edward T.; and Johnson, David F.: Effect of Porous Bleed in a High-Performance, Axisymmetric, Mixed-Compression Inlet at Mach 2.50. NASA TM X-1692, 1968.

6. Cubbison, Robert W.; Meleason, Edward T.; and Johnson, David F.: Performance Characteristics From Mach 2.58 to 1.98 of an Axisymmetric, Mixed-Compression Inlet System With 60-Percent Internal Contraction. NASA TM X-1739, 1969.

7. Hurrell, Herbert G.; Vasu, George; and Dunbar, William R.: Experimental Study of Shock-Positioning Method of Ram-Jet-Engine Control. NACA RM E55F21, 1955.

8. Nussdorfer, T. J.: Some Observations of Shock Induced Turbulent Separation on Supersonic Diffusers. NACA RM E51L26, 1954.

1. Report No. NASA TM X-2397	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle DETERMINATION OF NORMAL-SHOCK POSITION IN A MIXED-COMPRESSION SUPERSONIC INLET		5. Report Date November 1971
		6. Performing Organization Code
7. Author(s) Miles O. Dustin, Gary L. Cole, and Robert E. Wallhagen		8. Performing Organization Report No. E-6373
9. Performing Organization Name and Address Lewis Research Center, National Aeronautics and Space Administration, Cleveland, Ohio 44135		10. Work Unit No. 720-03
		11. Contract or Grant No.
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration, Washington, D.C. 20546		13. Type of Report and Period Covered Technical Memorandum
		14. Sponsoring Agency Code
15. Supplementary Notes		

16. Abstract

Methods for determining inlet normal shock position from wall static pressure profiles were investigated. By using methods investigated in this report, it should be possible to control an inlet with less stability margin. Variation in inlet angle-of-attack caused drastic changes in pressure profile shape and wide variations in the angle of the shock plane with respect to the inlet axis. Thus, four sensors located around the circumference may be required to handle angle-of-attack and sideslip variations. One criterion would determine the shock position by comparing individual statics to a properly selected reference pressure.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Supersonic inlet Jet engine Shock Sensor	Unclassified - unlimited

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	20	$3.00

* For sale by the National Technical Information Service, Springfield, Virginia 22151

"The aeronautical and space activities of the United States shall be conducted so as to contribute . . . to the expansion of human knowledge of phenomena in the atmosphere and space. The Administration shall provide for the widest practicable and appropriate dissemination of information concerning its activities and the results thereof."

—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

NASA SCIENTIFIC AND TECHNICAL PUBLICATIONS

TECHNICAL REPORTS: Scientific and technical information considered important, complete, and a lasting contribution to existing knowledge.

TECHNICAL NOTES: Information less broad in scope but nevertheless of importance as a contribution to existing knowledge.

TECHNICAL MEMORANDUMS: Information receiving limited distribution because of preliminary data, security classification, or other reasons.

CONTRACTOR REPORTS: Scientific and technical information generated under a NASA contract or grant and considered an important contribution to existing knowledge.

TECHNICAL TRANSLATIONS: Information published in a foreign language considered to merit NASA distribution in English.

SPECIAL PUBLICATIONS: Information derived from or of value to NASA activities. Publications include conference proceedings, monographs, data compilations, handbooks, sourcebooks, and special bibliographies.

TECHNOLOGY UTILIZATION PUBLICATIONS: Information on technology used by NASA that may be of particular interest in commercial and other non-aerospace applications. Publications include Tech Briefs, Technology Utilization Reports and Technology Surveys.

Details on the availability of these publications may be obtained from:

SCIENTIFIC AND TECHNICAL INFORMATION OFFICE

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Washington, D.C. 20546