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LOCKHEED MISSILES & SPACE COMPANY
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COMPARISONS OF EXPERIMENTAL SUPERSONIC FLOW FIELDS WITH RESULTS OBTAINED BY USING A METHOD OF CHARACTERISTICS SOLUTION

April 1966

Contract NAS8-20082

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FOREWORD

This document presents the results of work performed by the Lockheed Missiles & Space Company, Huntsville Research & Engineering Center, while under subcontract to Northrop Space Laboratories (NSL-PO5-09287) in support of the Aero-Astrodynamics Laboratory of Marshall Space Flight Center (MSFC) Mission Support Contract NAS8-20082. This task was conducted in response to the requirement Schedule Order B-49, Paragraph A. The Technical Director for this task was Dr. Richard C. Farmer NASA/MSFC R-AERO-AT.

ACKNOWLEDGEMENTS

Appreciation is hereby expressed to R. J. Prozan and H. W. Butler for their contribution to this work. The assistance of Dr. R. C. Farmer in obtaining experimental data from various sources is appreciated. Acknowledgement is also extended to J. F. Tabor for his patience and skill in preparing the many figures for this report.

SUMMARY

This report presents the results of a comprehensive comparison study which was carried out to verify the LMSC/HREC method of characteristics program as a working tool to compute nozzle and exhaust plume flow fields. Comparisons were made with experimental data such as rocket nozzle wall contour pressures, jet exhaust plume shapes, and shock locations in rocket exhaust flow fields.

This report is comprised of three basic parts. First, theoretical results from the LMSC/HREC method of characteristics program were compared with theoretical results obtained from another independent method of characteristics program. Next, the program results were compared with experimental data. (These experimental comparisons are subdivided into groups dealing with ideal gases and groups dealing with reacting gases.) Finally, the program's versatility is demonstrated by treating several problems of a somewhat unusual nature.

In all cases where comparisons between experimental data and the method of characteristics are shown, the starting information used in the analysis was exactly that provided by the data source. The raw experimental data accuracy was not questioned or modified in any way, but, in some instances some modification may have been justifiable. However, the study was conducted under the premise that one would not normally know the resultant flow field before the computations were made. Thus, some of the comparisons are very good while others do not correlate quite so well. It has been the philosophy of those preparing this report to present the data, for the most part, in the form of experimental schlieren photographs overlayed with the theoretical data. This eliminates any bias which would be inherent in the transfer of the data to another graphical form.

In general, the comparisons are excellent considering the broad range of pressure ratios, types of gases, (both ideal and reacting), and nozzle configurations studied.

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SYMBOLS

M_j	Mach number at nozzle exit
O/F	oxygen to fuel ratio
P_∞	ambient pressure, psia
P_t	chamber pressure, psia
P_j	pressure at nozzle exit, psia
R	radial distance, inches
R_j	radius of nozzle exit, inches
X	axial distance, inches
γ	ratio of specific heats
θ	nozzle exit half angle, degrees

INTRODUCTION

The flow of exhaust gases from rocket nozzles can cause a variety of complex design problems. For example, many problems arise depending on vehicle altitude and the proximity of a rocket nozzle to adjacent surfaces, if the exhaust gases impinge on such surfaces. Such problems are extremely acute in near vacuum conditions which cause large billowing jet plumes. For vehicle flights within the earth's atmosphere, expanding plumes may cause separation of the air flow over the vehicle afterbody introducing unstable aerodynamic characteristics. During separation of vehicle stages, rocket exhaust plume impingement may induce forces and pressures on the vehicle which must be evaluated. Attitude control rockets used for orbit corrections also cause direct impingement of exhaust gases on parts of the vehicle structure. Exhaust plumes also can cause heating problems to the base regions of vehicles by their radiation characteristics, thus knowledge is required concerning the spatial distribution of flow properties, i.e., temperature, density, etc.

In view of the many problems associated with rocket exhaust plumes, various studies have been carried out in an attempt to gain insight into their behavior. Most of these studies, however, have been restricted to ideal gas exhaust plumes at relatively low rocket chamber to ambient pressure ratios.

The method of characteristics has long been recognized as an accurate method for theoretically describing compressible fluids in supersonic flow. Therefore, a procedure for obtaining numerical solutions to problems in supersonic steady flow by the method of characteristics was programmed by LMSC/HREC for digital computer computations, Reference 1. This report presents correlations of theoretical predictions, made by using methods described in Reference 1, with experimental data obtained from various

sources and covering a wide range of conditions including reacting gas effects. Other comparison studies, such as that reported in Reference 4, have correlated one set of experimental data with their theoretical predictions. This document, which reports on comparisons between theory and experimental data from such a wide variety of sources, contains what is thought to be the most comprehensive group of comparisons that can be found.

In addition to the comparison studies, demonstration of the computer program's flexibility is also shown in this report.

Section 1
TECHNICAL DISCUSSION

1.1 THEORETICAL COMPARISONS

In order to theoretically verify the flow field solutions obtained by the LMSC method of characteristics program, Reference 1, a comparison with theoretical data from Reference 2 was made. Reference 2 presents data generated by a method of characteristics program developed by Convair. These data are presented in the form of free boundaries, shock waves, and Mach number contours. Figure 1 shows the comparison between the LMSC/HREC method of characteristics program and the results shown in Reference 2, for a slightly under expanded nozzle. As can be seen in Figure 1, agreement between the two solutions is quite good. Figure 2 presents data comparisons for a highly under-expanded nozzle. Here again, agreement is excellent even when the free boundary and shock wave exceed 90 degrees. In many instances, the plume shape is of primary interest; therefore, a comparison of free boundaries at various altitudes has been made. Figure 3 shows the results of the two solutions indicating very good agreement. Since the data presented in Reference 2 is restricted to ideal gases, all of the theoretical comparisons shown in Figures 1, 2, and 3 were computed for an ideal gas at the conditions noted on each figure.

1.2 EXPERIMENTAL COMPARISONS

1.2.1 Cold Flow (Ideal Gas) Cases

Several schlieren photographs of cold gas plumes flowing from a small air nozzle were obtained from Mr. Joseph L. Sims of the NASA/MSFC Aero-Astrodynamics Laboratory. Chamber-to-ambient pressure ratios and nozzle geometry were also supplied so the theoretical comparison of the experimental plume free boundary and shock waves could be performed. Figure 4 shows the schelieren photograph of a plume wherein the shock wave emanates from the nozzle lip and goes through a regular reflection at the axis. The theoretical comparison, indicated by dots on the photograph, shows fair agreement for both the free boundary and the shock waves. It was possible, for this case, to continue the theoretical solution downstream because of the regular reflection of the shock waves at the axis.

It can be seen that the theoretical points deviate progressively as the solution proceeds downstream from the first reflected shock. It was felt that the downstream deviation obtained for this case is probably due to a lack of accurate pressure ratio data. Any inaccuracy introduced in the initiation of the flow problem is then accumulative as the solution is continued through each succeeding shock reflection.

Figure 5 presents another schlieren photograph of an exhaust plume with superimposed theoretical shocks and boundaries. Excellent agreement is obtained here for the free boundary and especially the shock wave. Solutions downstream of the Mach disc were not attempted for this case. Theoretical predictions for both Figures 4 and 5 were accomplished using ideal gas solutions and the nozzle flow conditions presented on each figure.

Other experimental data were obtained from Reference 3 which

describes cold flow studies of a 1/56 scale F-1 engine nozzle. Static pressures on the interior nozzle contour and schlieren photographs of the plume flow field were obtained from that investigation. Two cases from Reference 3 were arbitrarily chosen for correlation with theoretical calculations using the LMSC/HREC computer program. The first case, shown in Figure 6, and identified as run number 2302-7-0 in Reference 3, is a slightly over-expanded jet and indicates good agreement between the calculated shock and free boundary as evidenced by the superimposed dots on the schlieren photograph. Comparison of the calculated and experimental static pressures on the interior nozzle contour for this same case are shown in Figure 7. Some deviation between theory and experiment is apparent in this comparison but is felt to be partially due to lack of accurate information about the nozzle contour, and inaccuracy of the recorded data and data reduction procedures.

Figure 8 presents the second case chosen, with theoretical data superimposed as dots on the schlieren photograph. Good agreement is again apparent for both the shock waves and the free boundary. Note that in Figure 8 there are two shock waves that appear to emanate from the nozzle lip. The inner one is a strong nozzle shock while the outer one develops by the coalescence of Mach waves reflecting from the free boundary. The theoretical solution for this case was continued downstream by using the experimental location of the Mach disc and reflecting the nozzle shock at this point. An approximate solution was made assuming that the flow behind the normal shock was swallowed by a pipe having the diameter of the Mach disc and extending down the axis. As such, this portion of the flow had no further influence on the method of characteristics solution. Both cases presented in Figures 6 and 8 were computed from the nozzle throat all the way through the nozzle and then out into the plume.

Another set of schlieren photographs was obtained from NASA, Langley Research Center, Hampton, Virginia, Reference 4. Reference 4 presents both experimental and theoretical studies of free jets exhausting

into still air from both sonic and supersonic nozzles at relatively high pressure ratios. The working gas in these tests was cold air, and as such, was treated in the theoretical analysis as an ideal gas. Figures 9, 10 and 11 show theoretical comparisons of the free jet boundary and internal shock structure for the Mach 1.0 nozzle over a range of pressure ratios. These photographs were obtained from Langley Research Center and their theoretical predictions are shown as white dots, whereas the predictions made by Lockheed using the LMSC/HREC method of characteristics program are shown as red dots. Excellent correlation was obtained as shown in the figures. Note that the Lockheed theoretical predictions include the shock wave, even for the high pressure ratio case. Some comment is in order concerning the apparent deviation of the predictions in the far wake region. This is probably caused by viscous spreading effects, as the prediction techniques are restricted to an inviscid solution. There is also reason to believe that flow instability may have occurred during the photographing of the jet flow field. Figure 12 shows the correlation obtained with the Mach 2.2 nozzle. Excellent agreement is again shown for the free boundary and for the shock wave down to the Mach disc location.

1.2.2 Hot Flow (Reacting Gas) Cases

A portion of the overall experimental verification of the LMSC method of characteristics program is concerned with the confirmation of the computational procedure used to facilitate the handling of reacting gases. Reference 1 presents a detailed description of the derivations and assumptions which are basic to the reacting gas method of characteristics computer program. It is sufficient here to say that reacting gas analyses are accomplished within the program by the use of tabular functions of the reacting gas properties. The reacting gas property data are obtained from the NASA/Lewis Thermochemical Computer Program, Reference 5. It is the purpose of this section of the report to present results showing that the methods employed and the theoretical models used, result in realistic reacting gas flow field calculations.

Full scale measured pressures on the Saturn V, F-1 engine skirt extension were obtained from Reference 6. For the tests reported in Reference 6 the F-1 engine was burning LOX/RP-1 and generated thrust of 1.5 million pounds. Correlations with the F-1 engine experimental data were attempted using a reacting gas analysis based on the recorded chamber pressure and oxygen/fuel (O/F) ratio from the test conditions. Figure 13 presents the correlation of measured and experimental pressures for the engine nozzle extension. The nozzle contour used in the analysis was slightly modified to account for the boundary layer displacement thickness due to turbine exhaust gas injection along the nozzle extension, Reference 7. In order to develop a better understanding of the effect of using reacting gas analyses, two other correlations with the F-1 pressure data were performed using ideal gas solutions. These cases are also shown in Figure 13. One case was computed using the chamber value of the ratio of specific heats, (γ) and the other case was analyzed using a value of γ at the nozzle exit plane. Of the three

cases shown in Figure 13, the reacting gas solution compares best with the experimental data.

Other full scale measured pressure data were obtained from Reference 8. These data are in the form of measured static pressures along the nozzle contour of the Saturn I H-1 engine. The H-1 engine burns LOX/RP-1 and generates about 188,000 lb. thrust. Comparison of the theoretical and experimental pressures is shown in Figure 14. Figure 15 presents a photograph of an H-1 firing with a superimposed theoretical boundary. It was thought that perhaps the low energy boundary layer gas may not be following the same flow pattern as the high energy core flow; therefore, an attempt was made to correlate the free boundary shown in the photograph (Figure 15) by including a boundary layer at the nozzle lip and expanding the supersonic portion inviscidly to the boundary pressure.

In Reference 9, the results of several rocket nozzle hot firing tests are reported. Variations in chamber pressure, O/F ratio, and type of propellants utilized characterize each run. One of these tests was arbitrarily chosen to compare with a theoretical model. The experimental results were presented in Reference 9 in the form of radial temperature and CO_2 partial pressure profiles across the nozzle exit plane. Initial attempts to theoretically compute these profiles proved unsatisfactory when the real gas properties employed were based upon a constant O/F ratio. Examination of these results and the discussion presented in Reference 9 indicated that due to injector design, the O/F ratio downstream of the injector plate was not constant but actually varied radially across the nozzle. Using the experimental temperature profile, and the calculated pressure distribution, an estimate of the O/F ratio variation at the nozzle exit was determined. This O/F ratio radial profile was assumed similar throughout the nozzle; an assumption which is justifiable if viscous and diffusive effects can be neglected. This assumption determines the throat O/F ratio variation. The problem was then re-run obtaining an improved estimate of the exit plane properties. This procedure was repeated until the exit plane properties did not change between successive

computer runs. The results of this analysis are presented in Figures 16 and 17. In Figure 16 it can be seen that the exit plane temperature profile predicted by assuming a constant O/F ratio does not compare at all with the experimental profile. The theoretical prediction, made by assuming a variable O/F ratio, however, compares quite favorably. Figure 17 shows a comparison of the experimental and theoretical CO_2 partial pressure profiles across the nozzle exit plane. Again it may be observed that the profile obtained using the constant O/F ratio assumption does not agree very well with the experimental profile. However, by using a variable O/F ratio the predicted profile compares much more closely with the experimental values.

No conclusions are made here concerning the validity of the theoretical predictions. It is apparent, however, that the O/F ratio variations caused by injector design is an important consideration and further study is warranted.

Further confirmation of the theoretical reacting gas method of characteristics computational procedure was obtained by comparing the plume shape, i.e., free boundary, and shock structure of rocket exhaust plumes photographed during firings. The photographs, engine nozzle configurations, and gas flow parameters were obtained from the Rocketdyne Division of North American Aviation, Reference 10.

All of the analyses performed to compare with these photographs were made using gas property data generated by the NASA/Lewis Thermochemical Computer Program. The propellants utilized, the O/F ratio and chamber pressures were all provided by Reference 10.

Figures 18 and 19 show comparisons of theoretical predictions as superimposed dots on the photographs of actual experimental firings at the chamber pressures and O/F ratios shown on the figures. Both of these firings utilized N_2O_4/N_2H_4-UDMH propellants.

For Figure 18, conditions were such that the incident shock wave was very weak and the method of characteristics mesh size used did not permit detection of this shock until its strength increased near the axis. Therefore only the shock reflection point on the axis is shown for this weak shock. In Figure 19 the presence of the Mach disc limits the application of the theory to regions upstream of the normal shock. The solution was continued downstream by reflecting the shock wave at the experimental location of the Mach disc and using the approximate technique already described in Section 1.2.1 and previously presented in Figure 8.

Figures 20 and 21 present correlations of theoretical free boundaries and shock waves for hot firing plume photographs of a LOX/ALCOHOL engine. The propellant combination was LOX/92.5% C_2H_5OH-7.5% H_2O. Propellants utilized, chamber pressures and O/F ratios are shown on each figure. The theoretical predictions agree very well for these two cases as far downstream as the solutions were carried. For Figure 21 the Mach disc again limits the prediction technique to portions of the flow field upstream of the normal shock. However, the solution was continued by reflecting the shock at the experimental location of the Mach disc and using the approximate technique described in Section 1.2.1.

1.3 PROGRAM CAPABILITY DEMONSTRATION

The most common and probably the most important applications of the LMSC method of characteristics program are in the areas of standard rocket nozzles and axisymmetric plumes. Many other complex flow fields can be treated if full use is made of the program's flexibility and options. In this section several unusual types of cases are described to demonstrate the program's versatility. Figure 22 shows the boundaries and shock waves computed for a problem with unusual boundary conditions. This figure also illustrates the program's ability to compute multiple right-running shock waves provided the preceding shock terminates by dissipation into the lower free boundary before the succeeding shock is initiated.

Another unusual application of the program is depicted by Figure 23. This solution is analogous to thrust vector control by secondary injection for a two-dimensional configuration, Reference 11. The penetration of the dividing stream line and internal shock are shown for various injection angles The external free stream flow imposes severe boundary conditions on the method of characteristics solution, but as shown by the results in this figure the program was able to handle these conditions satisfactorily.

Section 2
CONCLUSIONS

All the comparisons presented in this report show excellent agreement considering the broad range of pressure ratios, types of gases, i.e., ideal and reacting, and nozzle configurations studies. It is therefore concluded that the LMSC/HREC method of characteristics program, Reference 1, will successfully and accurately compute theoretical, inviscid supersonic flow fields for a wide variety of conditions.

Several problem areas were encountered during the course of this investigation,however, which need further investigation. The program used in this study is restricted to inviscid flow fields. A method which would couple the viscous-inviscid interaction phenomena would further enhance flow field prediction capabilities. The effects of inadequate mixing of the fuel and oxidizer in the combustion chamber, either accidental or deliberate (for nozzle wall cooling effect), bears further investigation in order to improve practical nozzle and plume flow field predictions. It was found during this study that in many cases the nozzle boundary layer growth had to be included as a perturbation of the inviscid flow field solution. More investigation of this problem is also warrented.

Several of the correlations attempted involved a Mach disc in the exhaust plume. Different approximate solutions are available for locating the triple point and some were used in order to continue the flow field solution downstream of the Mach disc. Lockheed has studied this problem in considerable detail; however, none of the solution techniques investigated were applicable in the general case. More study should be carried out in this vital area.

Section 3
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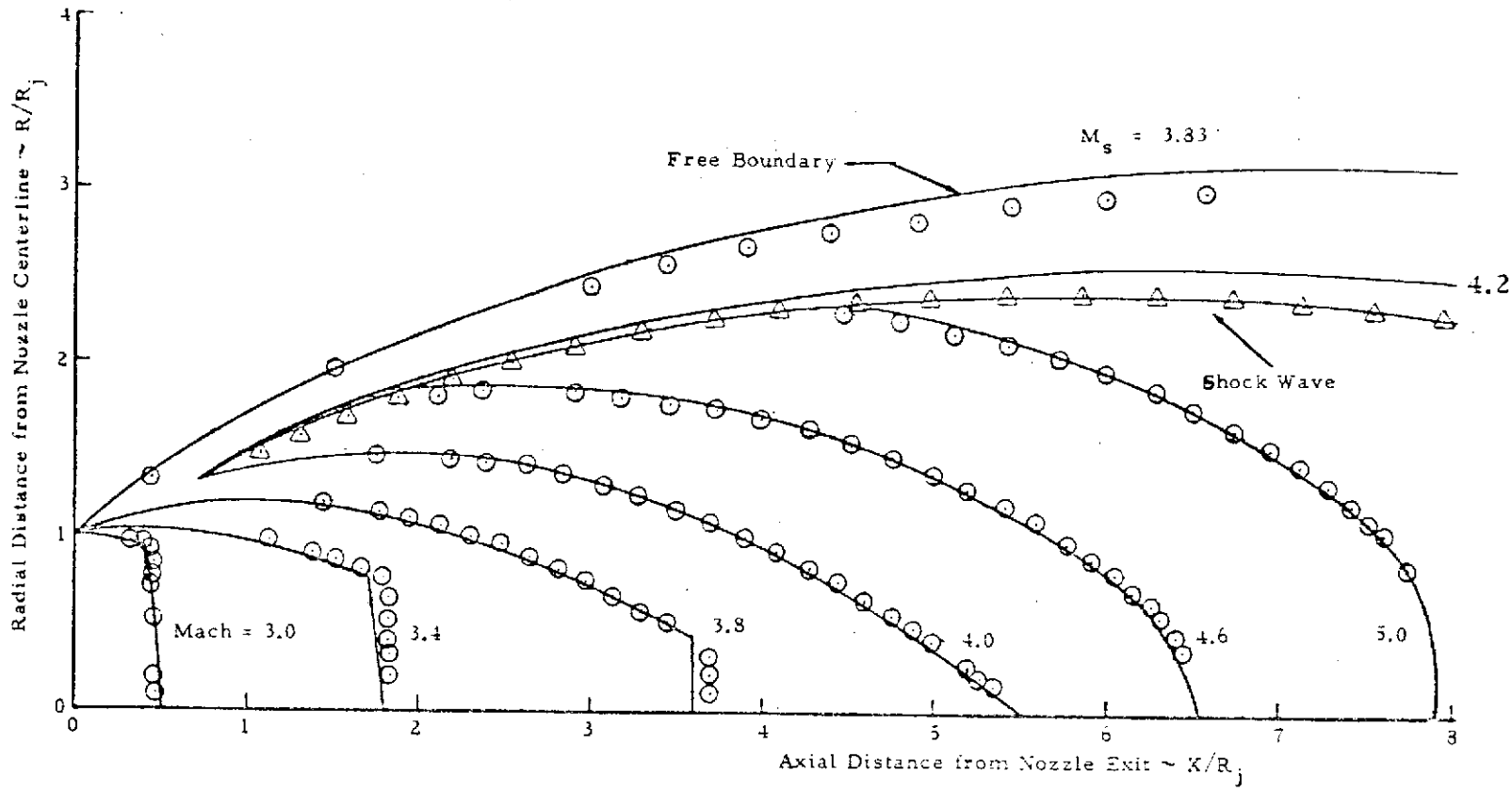
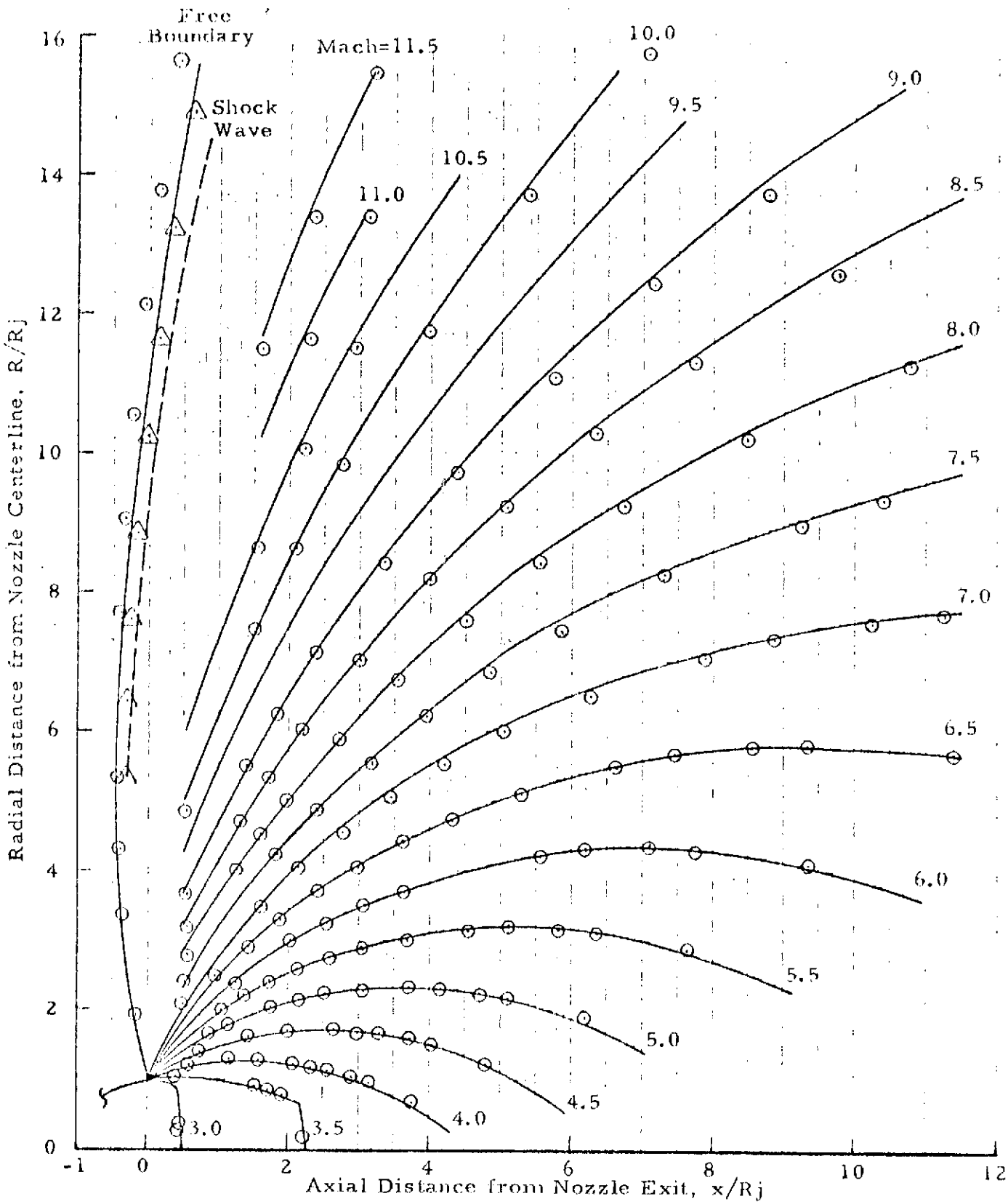
P1. Prozan, R. J., Interaction of Channel Exhaust with Primary Approach Flow and Analyses of Tail Interference for the DC-MAW Vehicle, Technical Report LMSC/HREC A712700, Lockheed Missiles & Space Company, Huntsville Research & Engineering Center, December 1965.



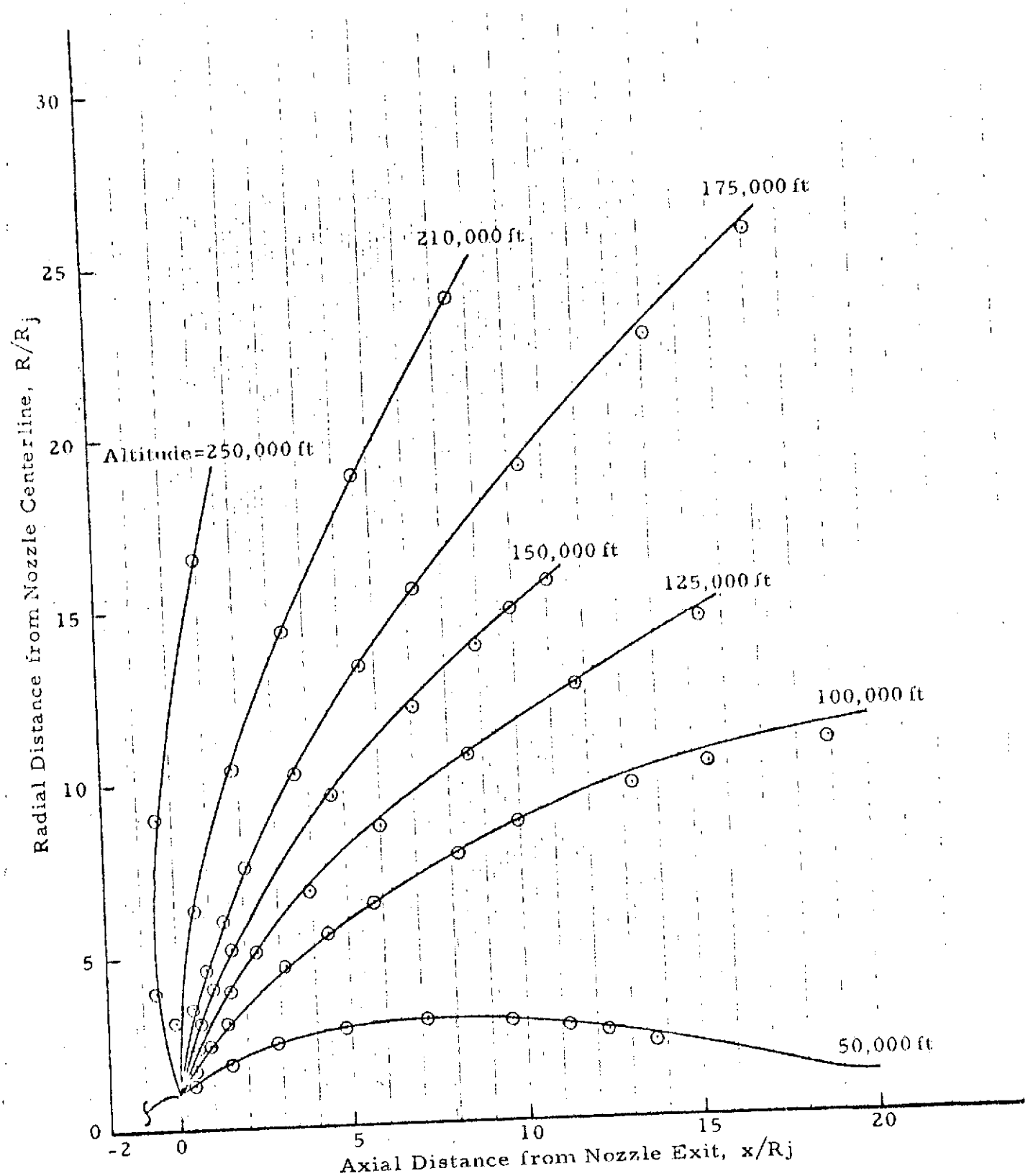
Note: Solid Lines Represent Data from Reference 2
Symbols Represent Data Generated by LMSC/HREC
Method of Characteristics Program

Figure 1 - Comparison of Constant Mach Number Contours In a Rocket Exhaust Flow Field at 50,000 Feet Altitude



Note: 1 Solid lines represent data from Reference 2
2 Symbols represent data generated by LMSC/HREC method of characteristics program

Figure 2 - Comparison of Constant Mach Number Contours in a Rocket Exhaust Flow Field at 250,000 ft Altitude



Note: 1 Solid lines represent data from Reference 2
2 Symbols represent data generated by LMSC/HREC Method of Characteristics Program

Figure 3 - Comparison of Rocket Exhaust Plume Boundaries at Various Altitudes

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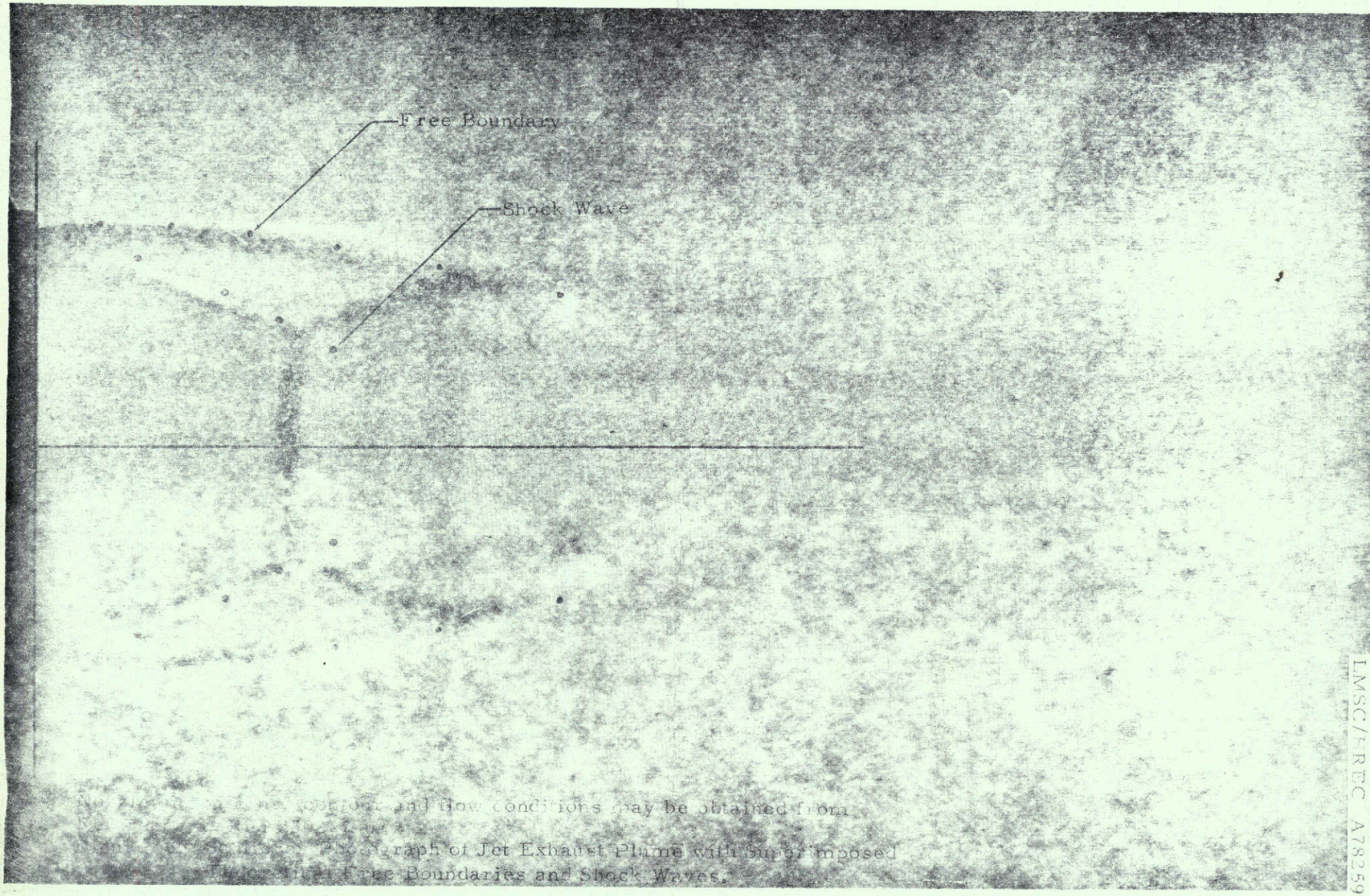
Figure 4. Schlieren Photograph of Jet Exhaust Plume with Superimposed Theoretical Free Boundaries and Shock Waves



position and flow conditions may be obtained from

Photograph of Jet Exhaust Plume with Superimposed Free Boundaries and Shock Waves.

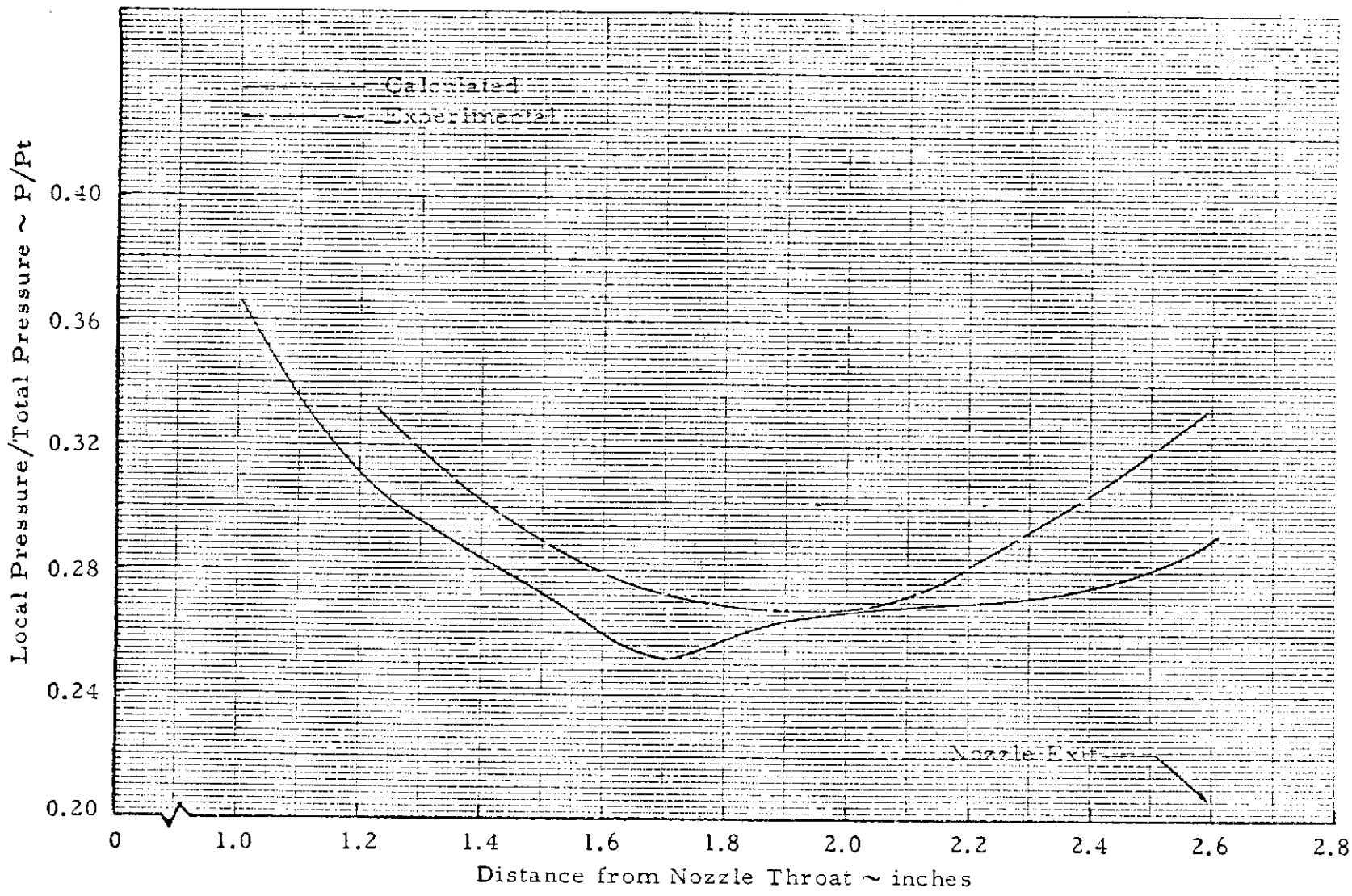


Figure 7 - Comparison of Measured and Computed Static Pressures on Interior Nozzle Contour of Scale Model F-1 Engine

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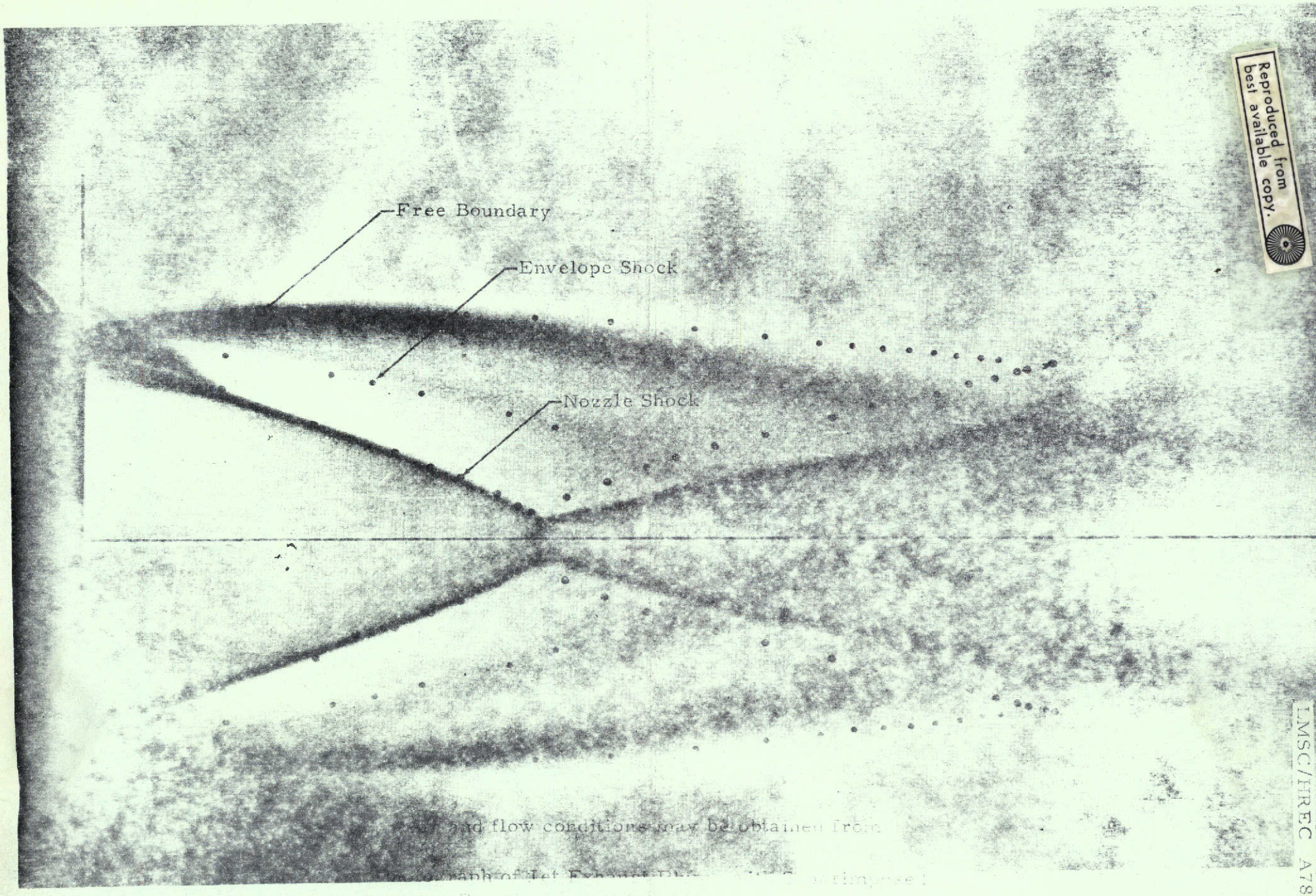

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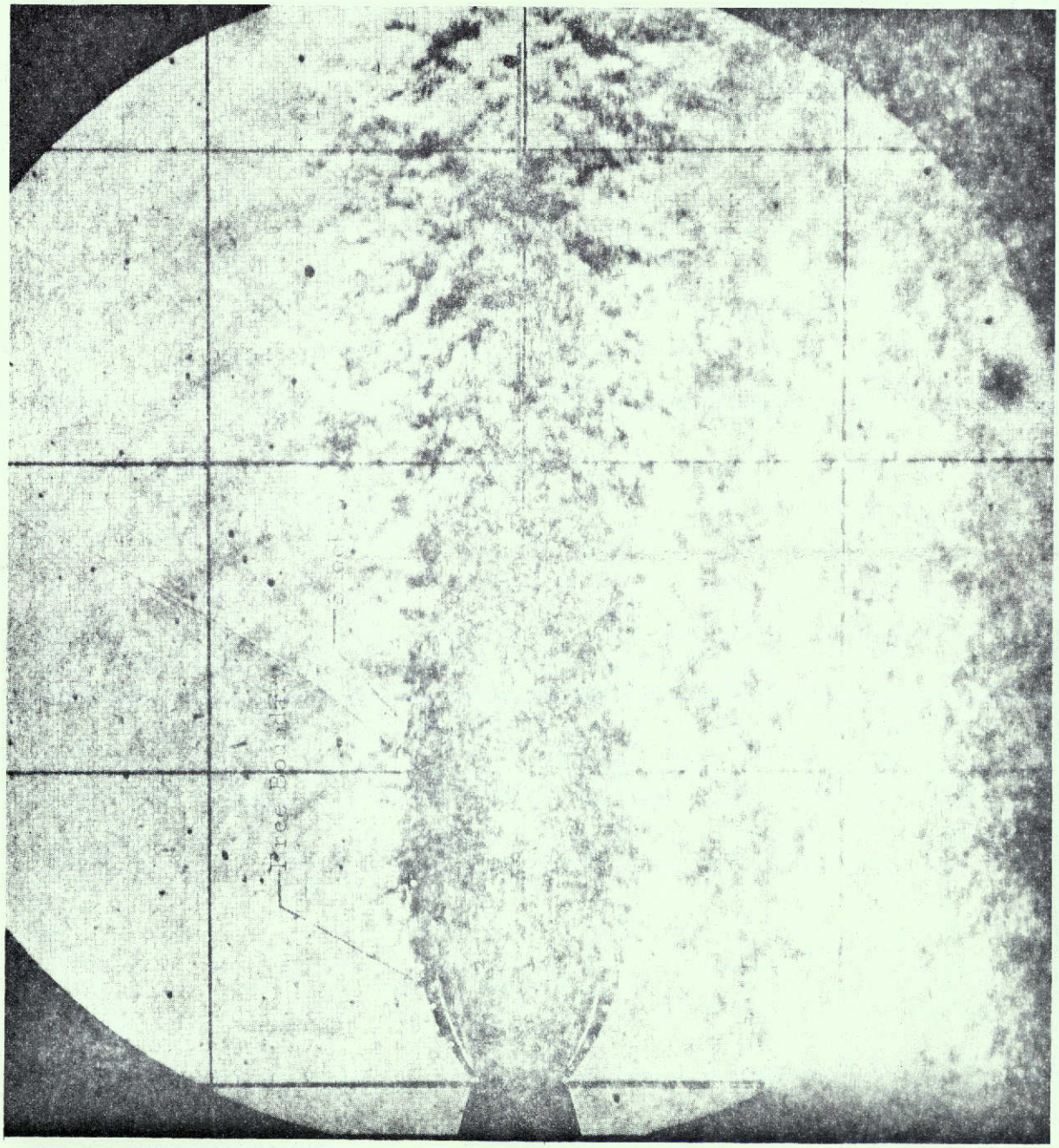
[illegible] graph of Jet Exhaust Pl[illegible] Simpson [illegible]


Tree Boundary

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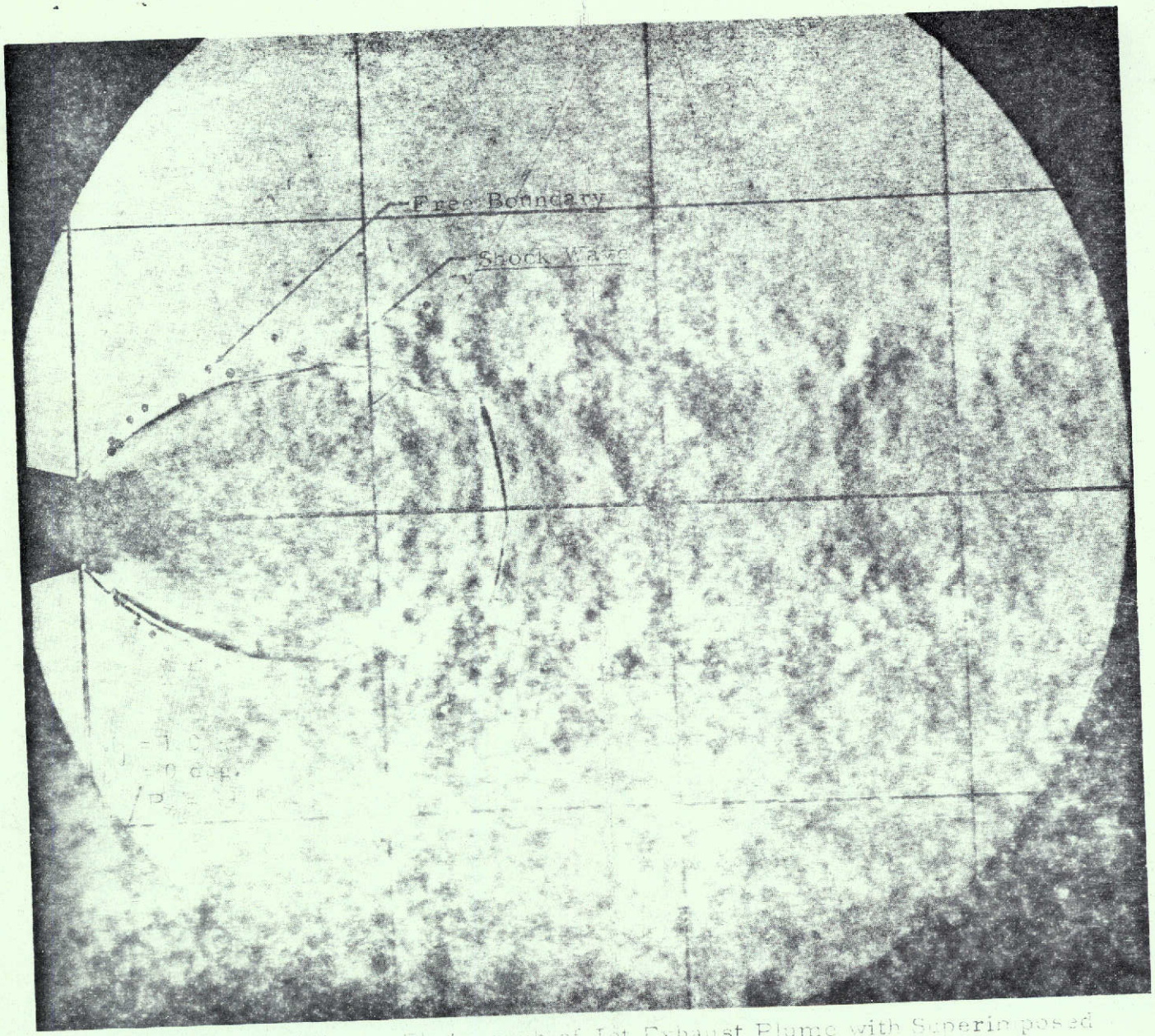
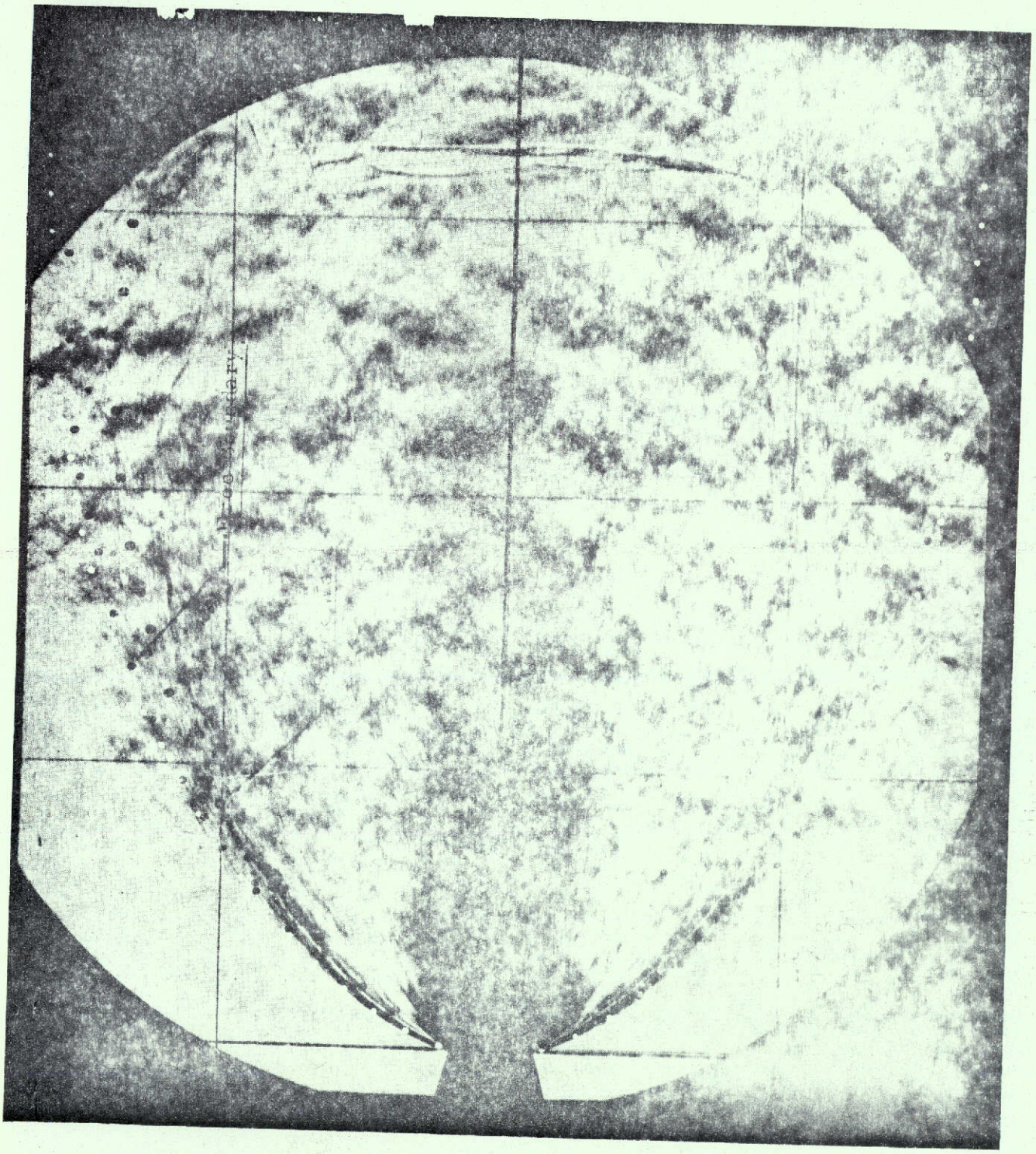


Figure 10. Schlieren Photograph of Jet Exhaust Plume with Superimposed Theoretical Free Boundaries and Shock Waves.



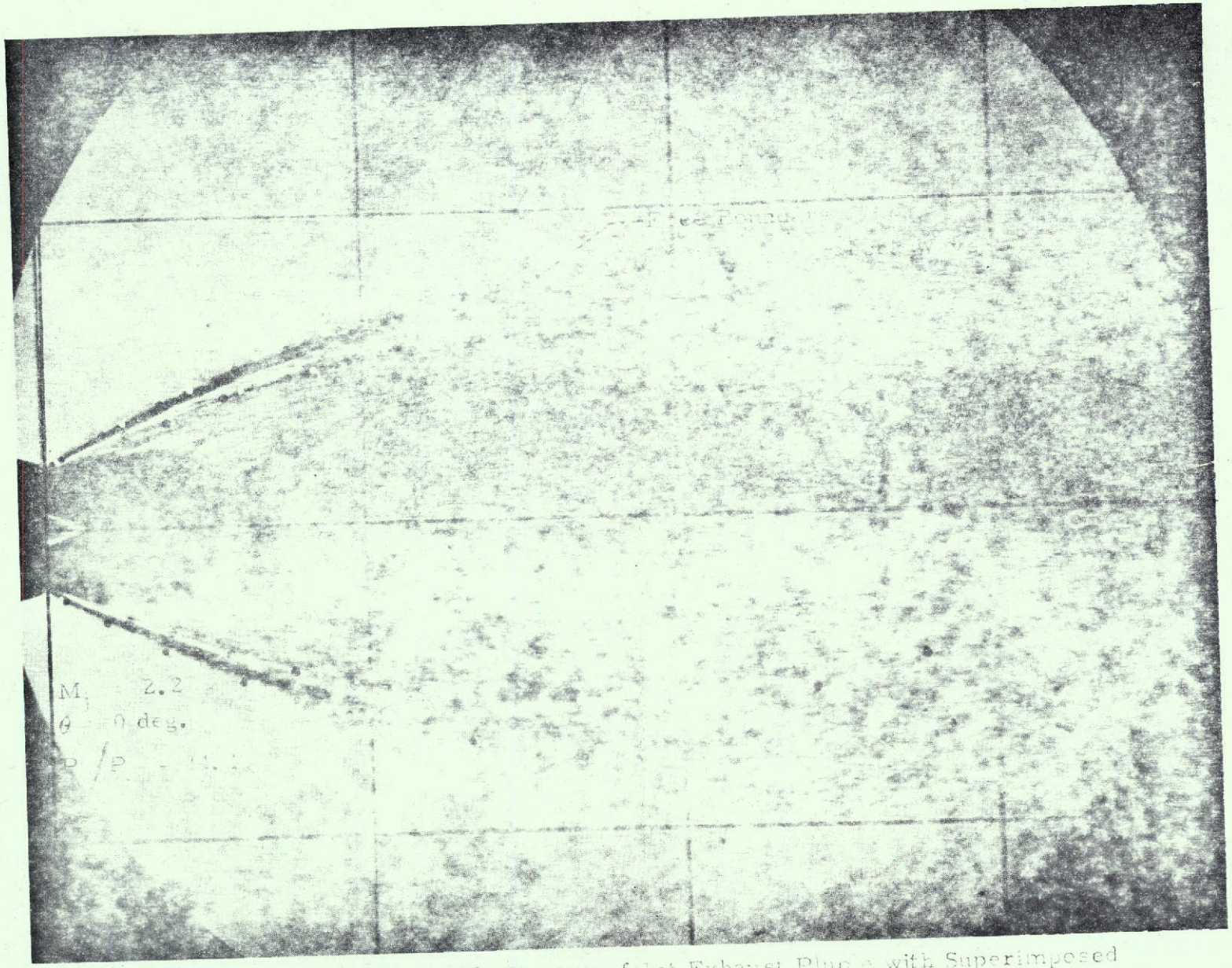


Figure 12. Schlieren Photograph of Jet Exhaust Plume with Superimposed Theoretical Free Boundaries and Shock Waves.

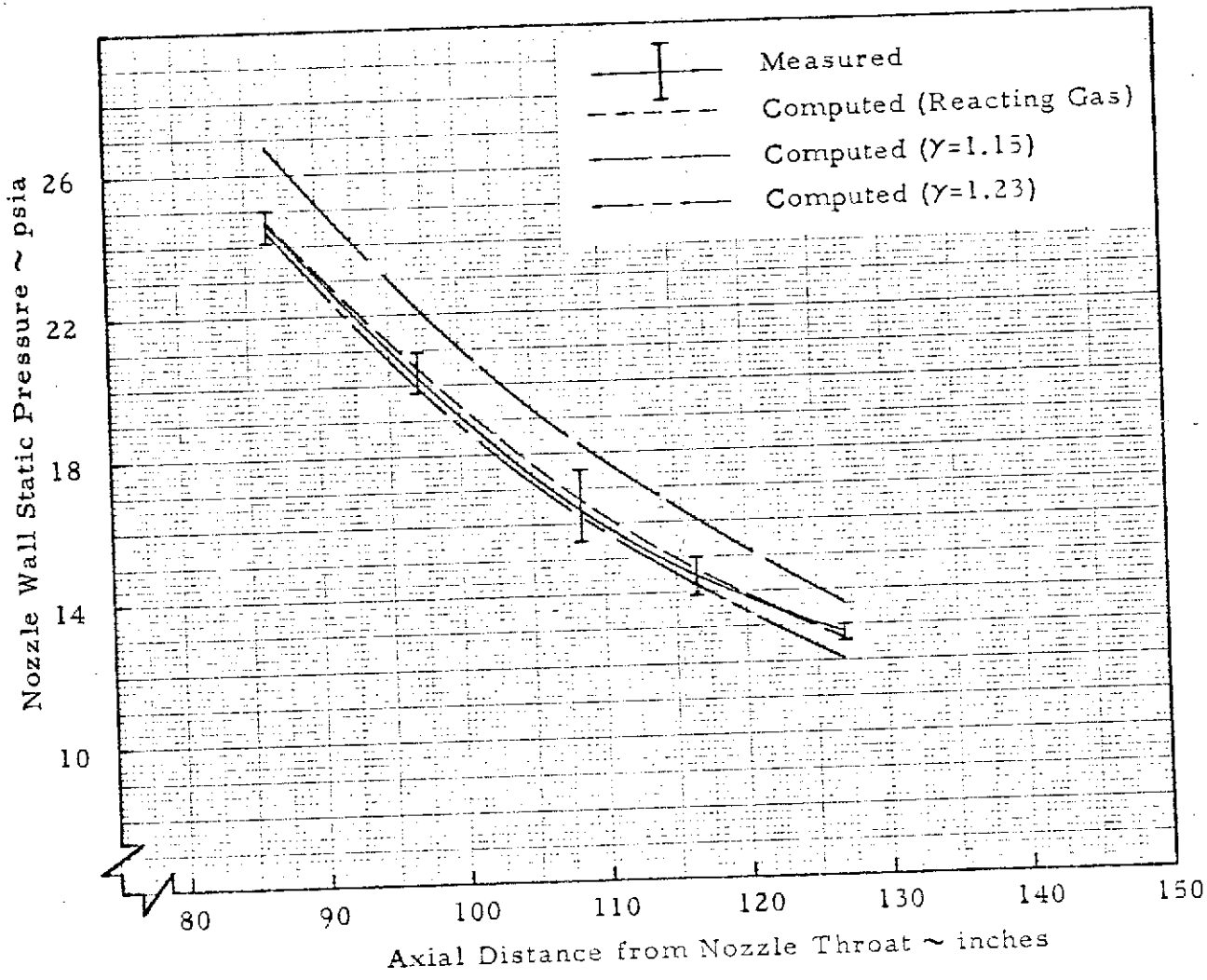


Figure 13 - Comparison of Measured and Computed Static Pressures on Interior Nozzle Skirt Extension of Full Scale F-1 Engine

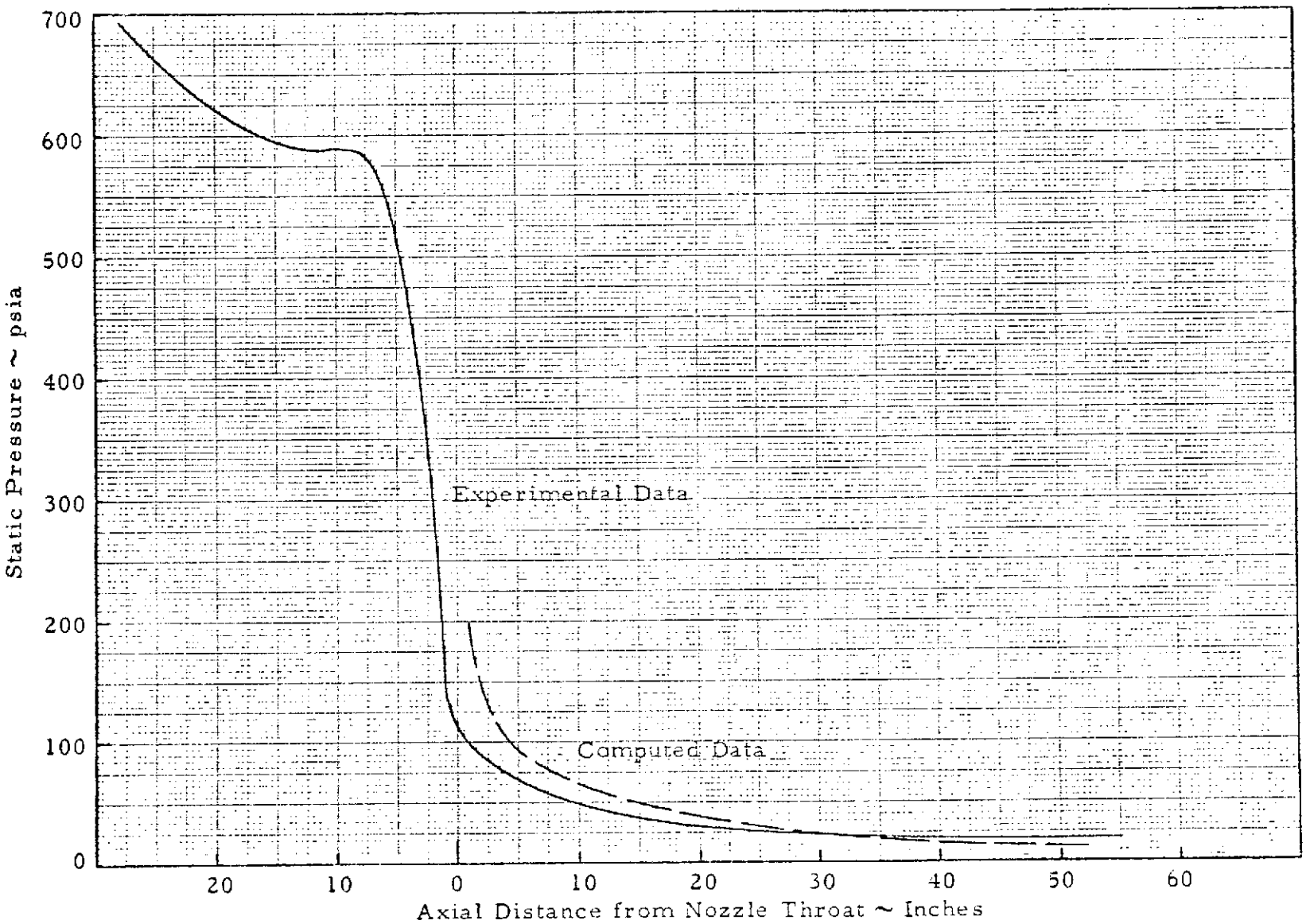


Figure 14 - Comparison of Experimental and Theoretical Nozzle Wall Static Pressures for the H-1 Engine

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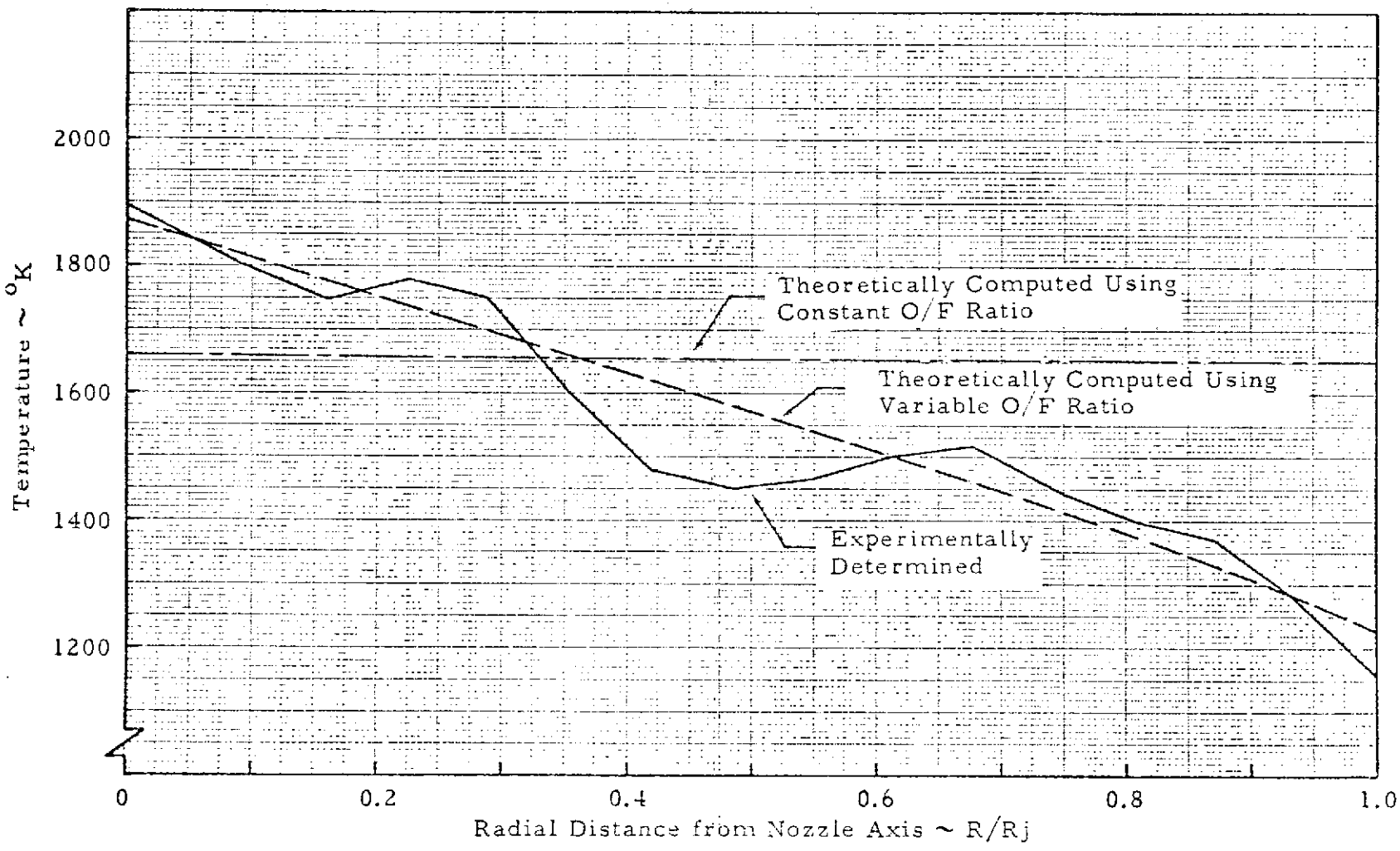

Figure 16 - Comparison of Experimental and Theoretical Temperature Profiles Across the Nozzle Exit Plane Flow Field

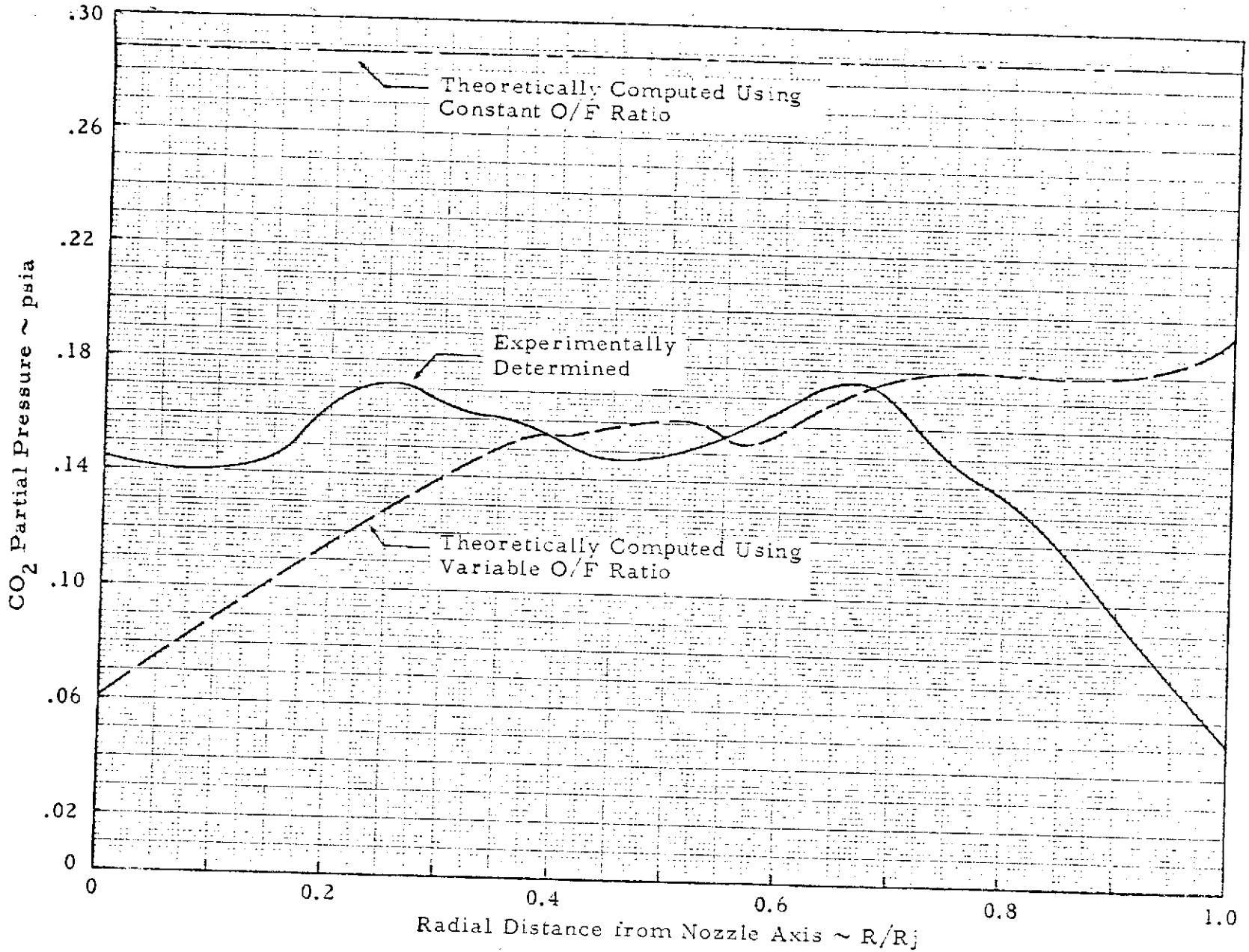

Figure 17 - Comparison of Experimental and Theoretical CO_2 Partial Pressure Profiles Across the Nozzle Exit Plane Flow Field

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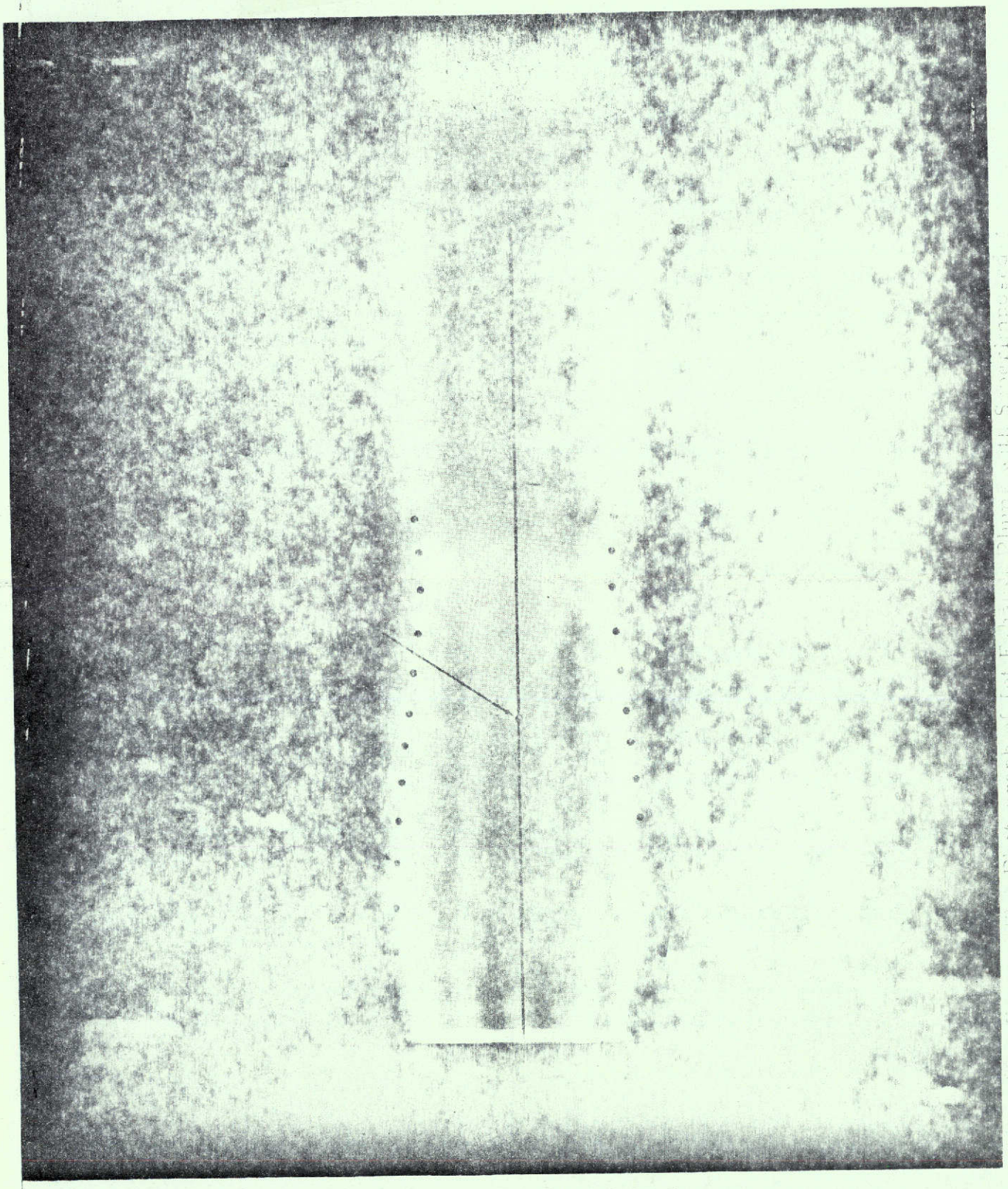

Fig. F-15. Photograph of Jet Exhaust Plume with Superimposed Theoretical Free Boundaries and Shock Waves.

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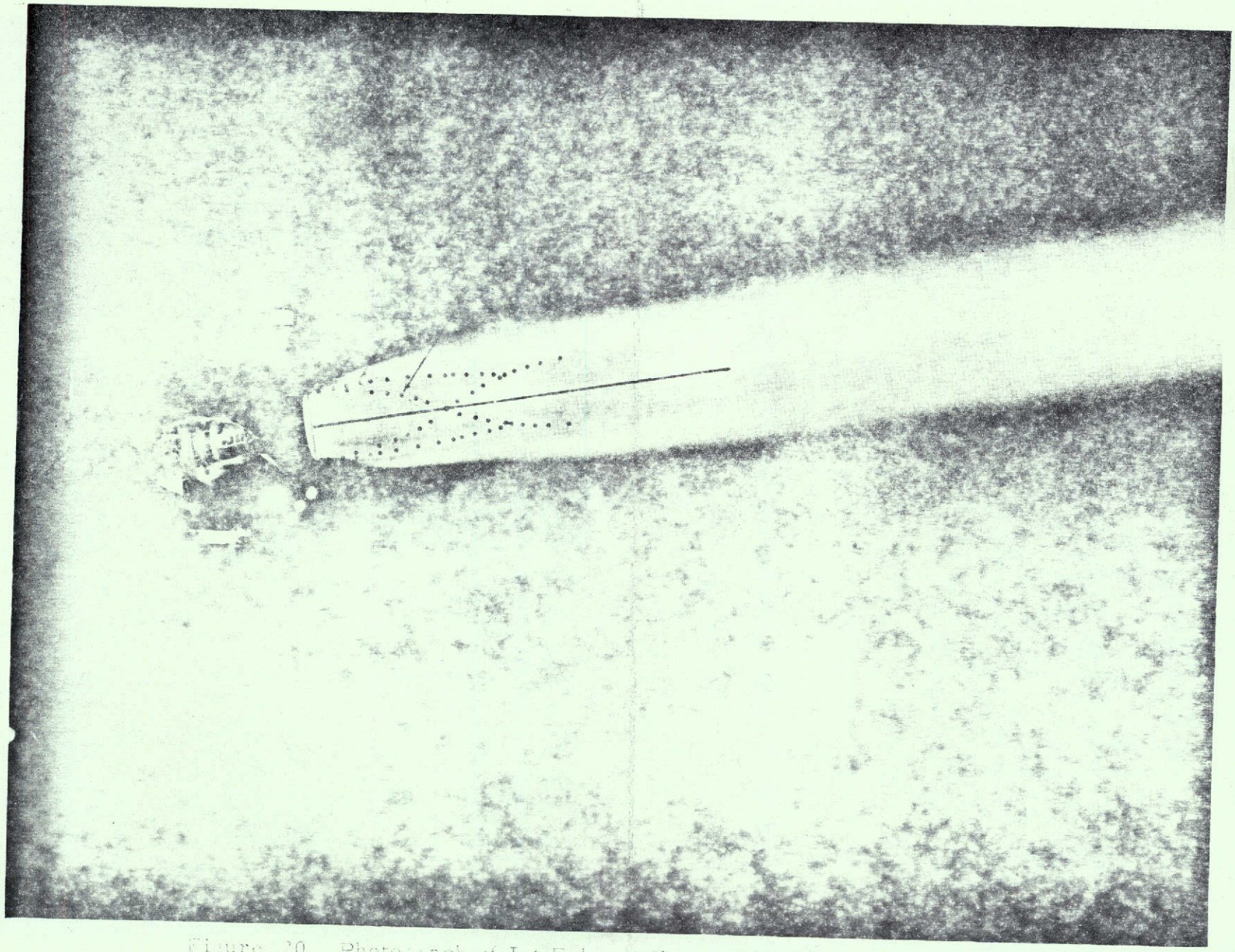

Figure 20. Photograph of Jet Exhaust Plume with Superimposed Theoretical Free Boundaries and Shock Waves.

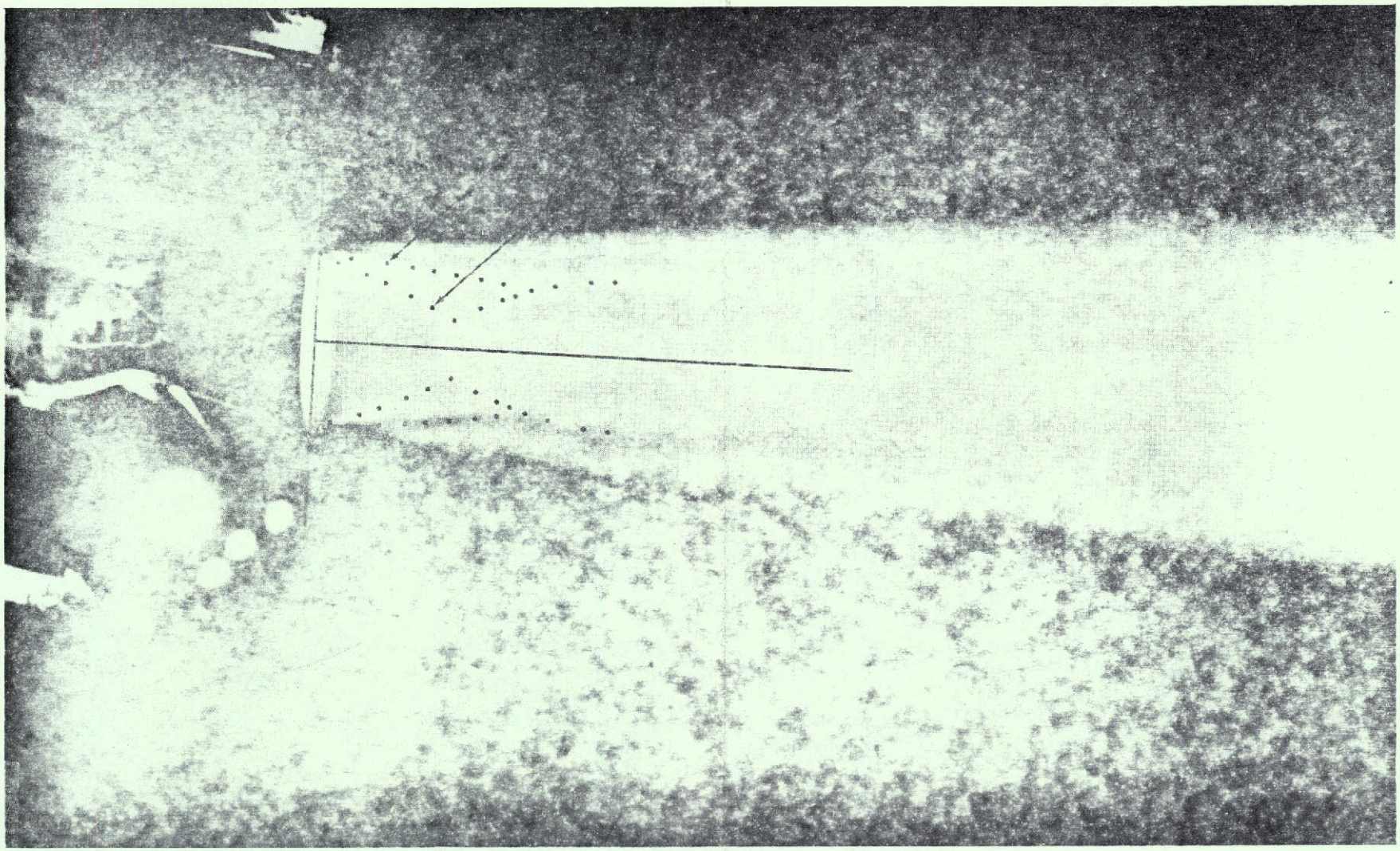

Figure 21. Photograph of Jet Exhaust Plume with Superimposed Theoretical Free Boundaries and Shock Waves.

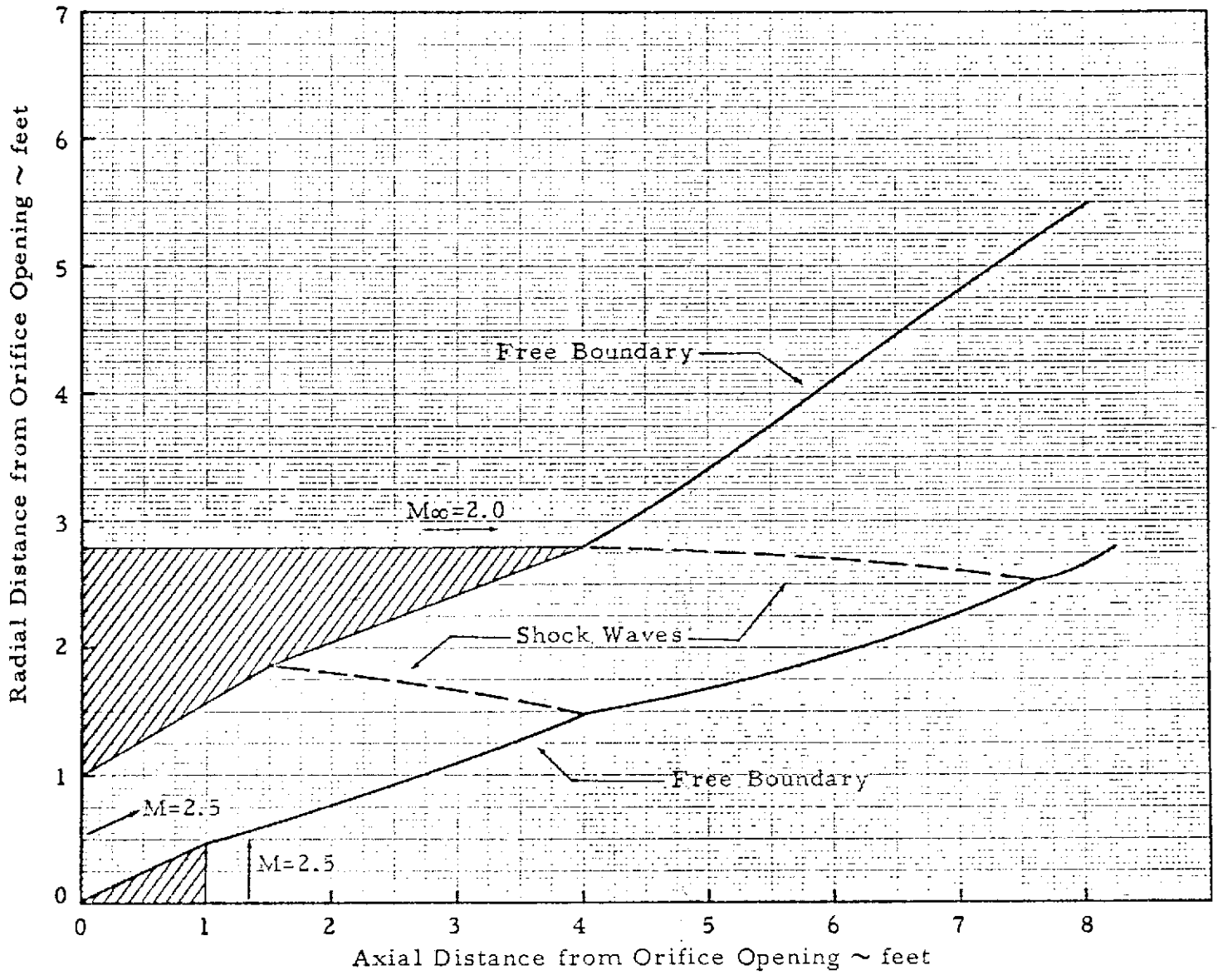

Figure 22 - Demonstration of Program Capability for Handling Unusual Flow Fields

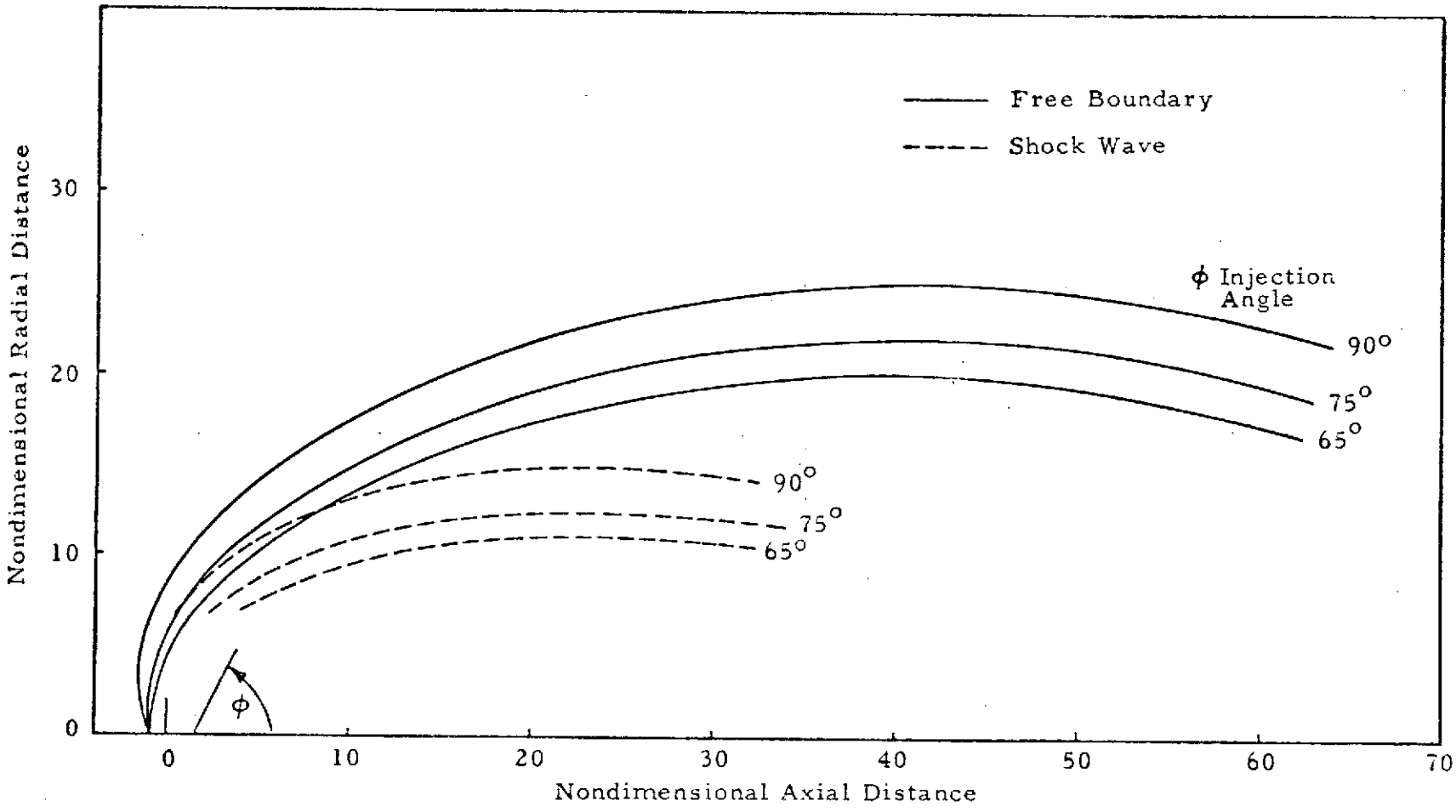

Figure 23 - Penetration of Jet Exhaust Stream into Free Stream at Various Injection Angles