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NUMERICAL INVESTIGATIONS IN THREE-DIMENSIONAL INTERNAL FLOWS

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I. BACKGROUND

NASA has an ongoing interest in supersonic and hypersonic inlet flowfield research. Their research efforts are intended to complement prospective aerospace vehicles, such as the High-Speed Civilian Transport (HSCT) and the National Aerospace Plane (NASP), as well as other variants of these vehicles intended for use with air-breathing propulsion systems. Computational Fluid Dynamics (CFD) is expected to play a large part in the design and analysis of such aircraft because experimental facilities are limited. The purpose of this Grant is to apply, evaluate and validate CFD tools for use in high-speed inlet flowfields.

In the previous progress report, a newly released full Navier-Stokes (FNS) code, OVERFLOW, was applied to the three-dimensional forebody of a hypothetical waverider designed for cruise at Mach 5 by Langley and Lewis Research Centers. The forebody flow ahead of the cowl lip station was treated with a full three-dimensional analysis and the results of that analysis were shown to agree with those previously obtained under the present Grant using the space-marched and time-marched versions of Greg Molvik's codes, STUFF and TUFF, respectively. For the flow downstream of the cowl lip and ahead of the entrance to the subsonic diffuser, a two-dimensional solution from the OVERFLOW code was shown and results were compared with those obtained from the STUFF and SCRAM2D hybrid analysis also developed under the present Grant. Nearly a two-order-of-magnitude decrease in CPU time was obtained from the OVERFLOW code compared to the hybrid.

Effort during the present reporting period continues the application of the OVERFLOW code. Solutions were obtained for the supersonic inlet and a three-dimensional subsonic diffuser with a terminal shock wave.

II. INTRODUCTION

In the present reporting period, the 3D version of the OVERFLOW code was used to solve the flow within the internal portion of the supersonic inlet. The internal portion of this inlet is bounded by an inflow plane containing of the leading edge of the sidewalls, the sidewalls, the ramp and cowl surfaces and an outflow plane just downstream of the minimum geometric area of the inlet. Boundary layer bleed was used in the two-dimensional calculations discussed in the previous progress report and that same bleed was applied in the present study. For reference, this bleed corresponds to locations designated as R2 and R3 in the Mach 5 inlet model test. Using the GRIDGEN code, a three-dimensional grid was generated that accounted for the viscous effects expected to occur on the sidewall, as well as those known to occur on the ramp and cowl surfaces. The internal flow grid size was 141 streamwise by 101 cross stream by 71 in the lateral direction between sidewalls. Since the flow entering the inlet was not symmetrical, the inlet was solved from sidewall to sidewall (without using a symmetry plane). In addition to the short sidewalls proposed in the Langley geometry database, a set of shorter sidewalls was also investigated in the present study and was shown to have beneficial effects with respect to the flow distortion exiting the supersonic inlet. In addition to these calculations, additional 3D solutions using the OVERFLOW code were obtained for the flow downstream of the throat of the supersonic inlet, including a terminal shock wave system produced by a back-pressured subsonic diffuser.

III. RESULTS AND DISCUSSION

In the previous progress report it was shown that the flow on the underside of the waverider is highly three-dimensional near the outboard module just ahead of the cowl lip station. In order to avoid difficulties with this situation pointed out in the previous report, only the flow in the inboard module was considered here. For reference, and to indicate the scale of the problem being considered, Figure 1 shows the normalized pressure obtained from the three-dimensional solution at the cruise Mach number of 5 from the leading edge of the vehicle through the throat of the inlet. Figure 1a shows the entire plane of that flow path, while Figure 1b shows the flow from the multiple ramp system and further compression through the inlet's throat. A maximum compression ratio of approximately 110 relative to the freestream atmospheric pressure is shown.

The three-dimensional flowfield internal to the supersonic inlet is shown in Figure 2. The inflow data are from the three-dimensional forebody solution presented in the previous progress report. Figure 2a shows the Mach number contours in five crossflow planes beginning at the left of the figure from the inflow plane. This figure shows the far wall and the cowl surfaces. (The ramp and near sidewall have been removed to show the solution.) The cowl shock wave and its ultimate intersection with the ramp surface near the ramp shoulder can be seen. Because of the boundary layer on the short sidewalls and the glancing cowl-shock-wave interaction, a distorted flowfield is produced that is evident in the last two computational planes shown. Figure 2b shows a detailed view of this distorted flowfield. Figure 3 shows the Mach number contours with particle traces superimposed on the figure. These particle traces are developed from particles released within the sidewall boundary layer. The behavior of these particles is similar to that observed previously, in which virtually all of the sidewall boundary layer flow was swept along the shock wave and deposited near the corner of the shock-reflecting surface.

In order to understand fully the nature of the distorted flow shown in Figures 2 and 3, a solution was run without sidewalls. This was accomplished through the use of the appropriate boundary conditions in the OVERFLOW code. These results are shown in the Mach number contours of Figure 4. As would be expected, a nearly two-dimensional flowfield is obtained. The primary features of the ramp and cowl boundary layer in the throat section of the inlet are clearly evident. No distorted features and no sidewall boundary layers are present. An inlet system using no sidewalls at all is believed to be unacceptable due to intermodular cross-talk during an unstart. Some isolation must be maintained, so another solution was run with a different set of sidewalls. This set of sidewalls, termed the very short sidewalls, was derived on the basis of the following argument. Since it is believed that the interaction of the sidewall boundary layer and any glancing shock wave will produce the distorted vortical flow features, the solution is simply to have the origin of the sidewalls downstream of the offending shock waves.

In the present inlet, the shoulder is intended to cancel the cowl shock wave and, as indicated in previous CFD efforts and the results shown in Figure 1, this is generally true. For example, a set of sidewalls that originate at the ramp shoulder and drop vertically to the cowl should eliminate the glancing interaction. These very short sidewalls could prevent some intermodular communication and they were imposed as boundary conditions. The computed flowfield results are shown in Figure 5. No substantial vortical features are shown and only nominal sidewall boundary layers, in addition to those for the ramp and cowl, are present near the inlet throat. The outflow from the supersonic diffuser under the three different sidewall conditions is summarized in Figure 6.

Because of the interest in operating certain classes of rotating machinery for one- and two-stage-to-orbit and cruise vehicles, a subsonic diffuser located downstream of the supersonic inlet flows depicted previously is of interest. In the present study, because of the dramatic decrease in required computational time that can be obtained for a full three-dimensional solution, the feasibility of using the OVERFLOW code for use in studying a terminal shock wave system and subsonic diffuser was investigated. The subsonic diffuser is assumed to be a straight diffuser having the dimensions required by the Lewis/Langley waverider aircraft. The area ratio is 7.5, while the normal length divided by diffuser inflow width has a value near 46. This produces an effective divergence angle of about 7.6 degrees. This diffuser is a transitioning diffuser from rectangular to round. The width of the diffuser remains constant. Its initial height is approximately 12.5 cm, while the circular section at the outflow is approximately one meter in diameter, consistent with known rotating machinery.

In order to reduce the computational time, a symmetry plane was assumed at the center of the diffuser. To start the computational process, supercritical solutions were obtained. The short sidewall and very short sidewall flowfields, which are those summarized in Figure 6, were used as inflow conditions to the diffuser. For supercritical cases, Figure 7 shows the three-dimensionality of the solutions portrayed as Mach number contours in several crossflow planes beginning at the inlet throat and progressing toward the circular section. The initially distorted profile associated the short sidewall inlet in Figure 6a is seen to persist throughout the duct.

Of considerable interest is the nature of these flowfields when back pressure is applied to the inlet. Back pressure was imposed at the outflow boundary and a normal shock system resulted. Bleed was employed for these calculations in order to control the boundary layer near the terminal shock region. Figure 8 shows the Mach number contours for two cases having a final pressure ratio of 170. Figure 8a shows a highly distorted flowfield resulting from the initial distortion with the short sidewalls present. Figure 8b shows a nearly uniform diffusion process for the very short sidewall case. The Mach number contours depicted in Figure 8 show the large amplification of the distortion in the throat region as a result of going through the terminal shock wave and proceeding into a subsonic diffuser. It is clear that the trade off in diffuser distortion versus intermodular separation due to the length of the sidewalls will be an important design consideration.

The total pressure recovery for the two cases shown in Figure 8 is very low because the terminal shock wave system is occurring at a relatively high Mach number. The value of the recovery is around 30% for both of these cases. It is of interest to continue to increase the back pressure and obtain a solution for which the terminal shock wave system is as far upstream as possible (lowest Mach number) in order to maintain stable operation. Figure 9 shows the result of imposing a back pressure, producing a compression ratio of 250. For this case, the Mach number and pressure distribution on the symmetry plane of the solution are shown. Figure 9a shows the Mach number contours on the symmetry plane, while Figure 9b shows the normalized pressure contours. Figure 9b indicates that the terminal shock wave occurs near the entrance to the diffuser within the region of bleed, so that the boundary layer is effectively controlled. The three-dimensional flowfield is portrayed in Figure 9c, where, again, the Mach number contours in various crossflow planes are shown. Relatively little distortion is observed with the very short sidewalls. The recovery relative to the freestream is shown in Figure 9d. To examine the distortion and recovery values in detail at the outflow plane (engine face), Figure 9e shows the normalized total pressure. The average total pressure recovery for this installed inlet system is about 46% with virtually no spatial distortion. The solution is also steady so that no temporal distortions occur.

The recovery value of 46% for the current inlet installed on the waverider aircraft is about the value that would be expected based on the results of the isolated inlet wind tunnel test, in which the maximum recovery for the inlet with full sidewalls was shown to be about 37%. The losses associated with the forebody for the waverider are less than those assumed for the 9-degree pre-compression body when recoveries were quoted for the wind tunnel test case.

IV. CONCLUSIONS

An investigation was carried out to examine the three-dimensional nature of the flowfield within the supersonic portion of the inlet for the hypothetical Mach 5 waverider aircraft. Solutions from two tentative sidewall configurations for the supersonic inlet were continued through a terminal shock wave system and into a long, transitioning subsonic diffuser. Large, three-dimensional flow effects were shown to exist for the short sidewalls proposed in the original Lewis/Langley waverider geometry. These three-dimensional flow effects are amplified when taken through a terminal shock wave system and allowed to develop in a subsonic diffuser that is transitioning from rectangular to round. In contrast, a set of very short sidewalls, having their origin downstream of any substantial shock waves in the supersonic inlet, produced very low distortion in the inlet throat, and, when that flowfield was taken through the terminal shock wave system and allowed to develop into the subsonic diffuser, very little distortion persisted. The case in which the terminal shock wave was moved upstream had a recovery value near 46%. This value is in good agreement with that expected from the installation on the waverider aircraft and results of the Mach 5 inlet wind tunnel test program.

During the course of the study, numerous subsonic diffuser solutions were obtained using parametric variations of the back pressure in order to examine the nature of the three-dimensional flow within the subsonic diffuser, including the terminal shock wave. This parametric study was possible through the use of the newly released OVERFLOW code. Since this code operates with nearly a two-order-of-magnitude decrease in required computer time compared to the previously-used SCRAM3D code, numerous parametric solutions could be obtained. The average run time for a solution to be developed from one parametric change to another was approximately 15 minutes of CPU time on the Cray C-90. These solutions were not run in a time-accurate mode since only developed, steady solutions are of engineering interest. Unsteady solutions were noted for some cases by observing large differences between successive "time" steps. The results presented in this report are believed to be the first calculations of the three-dimensional inlet flow for an aircraft operating at a high Mach number with a terminal shock wave and transitioning subsonic diffuser. The use of the single code from the nose of the vehicle to the compressor face reduces the engineering manpower effort required to obtain such a solution.

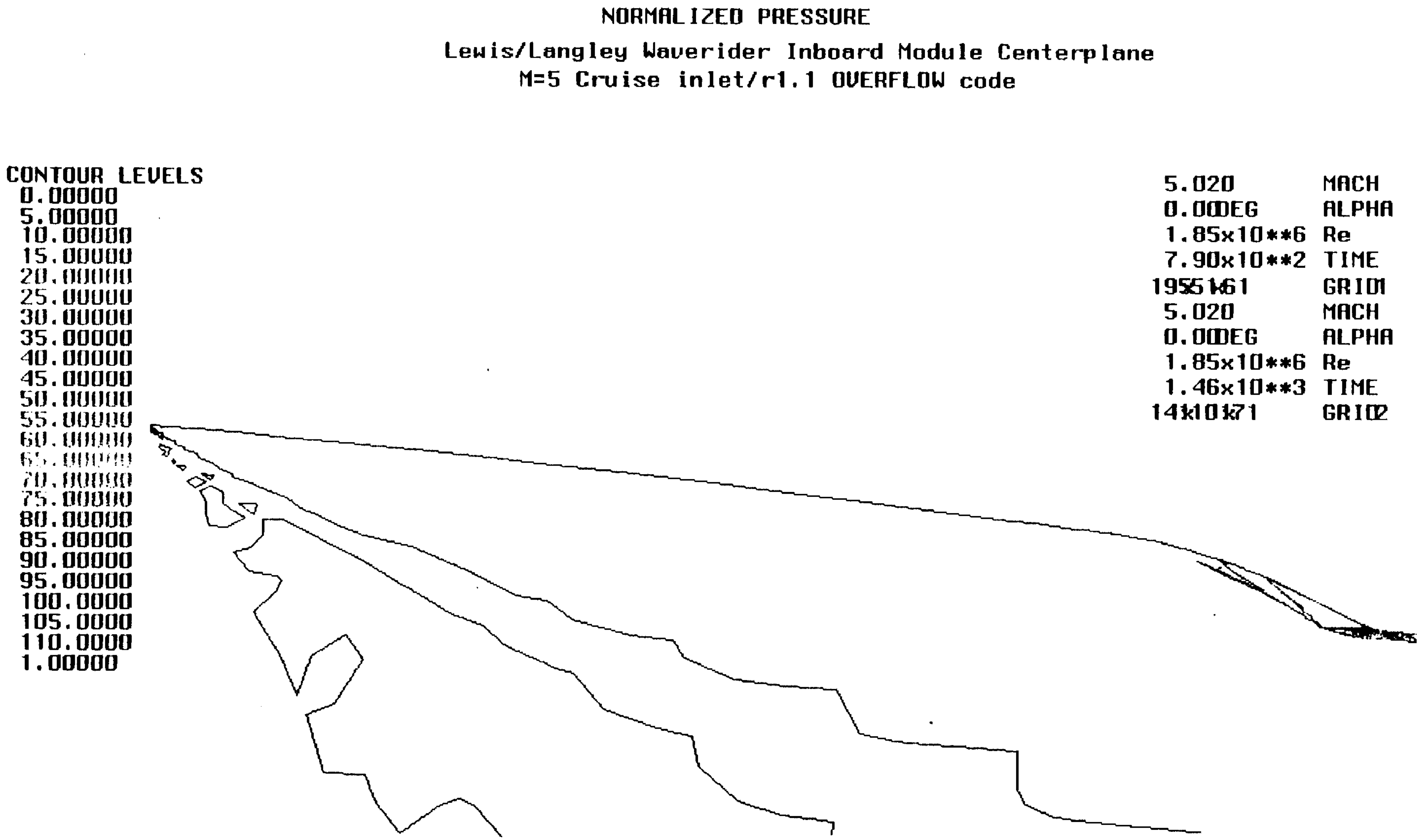

FIGURE 1 Pressure ratio contours for the plane containing the center of the inboard module from the tip of the vehicle to the inlet throat.

a) Entire computational plane

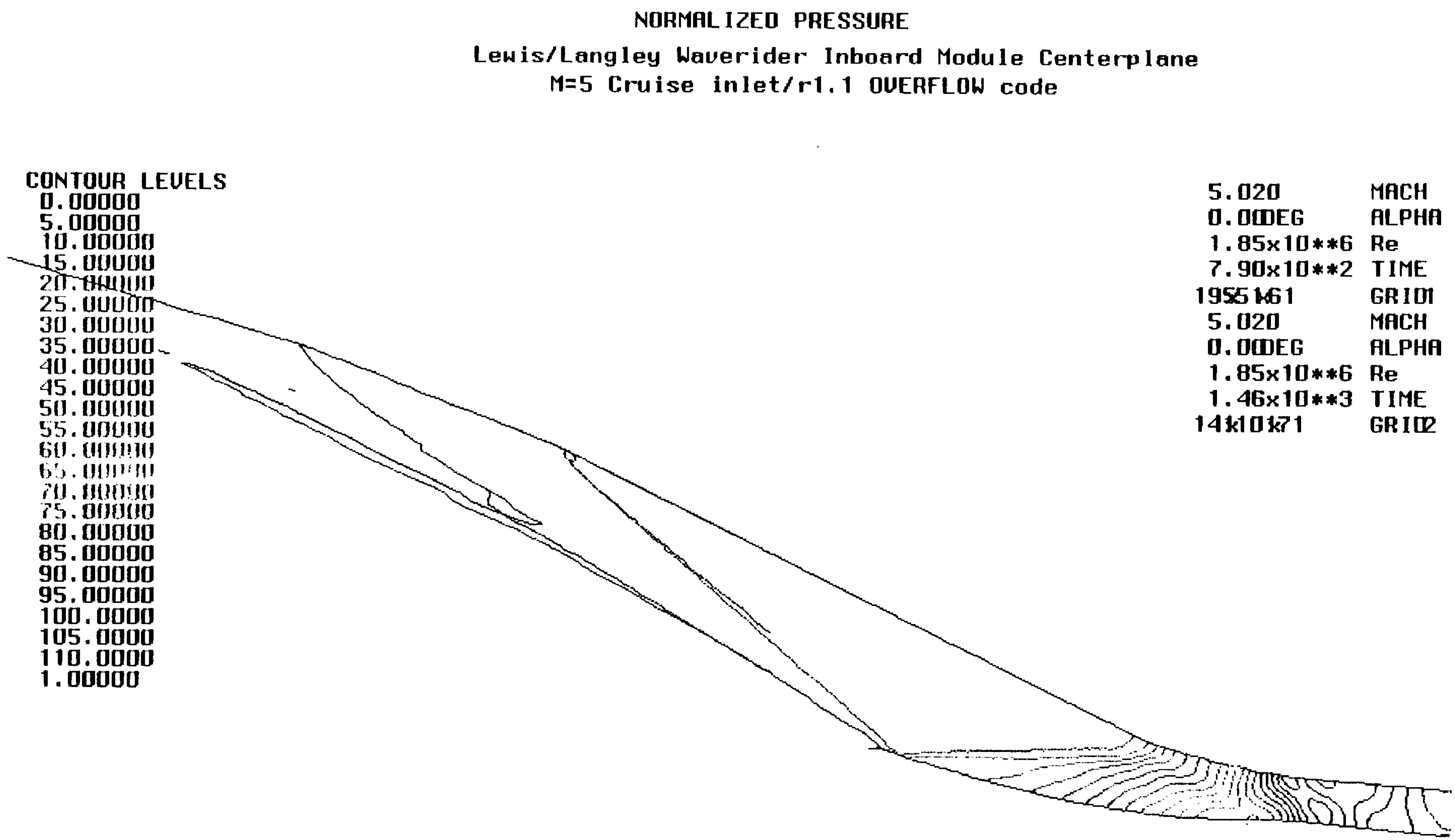

FIGURE 1 Concluded.

b) Detail of ramp shock waves and internal flow compression

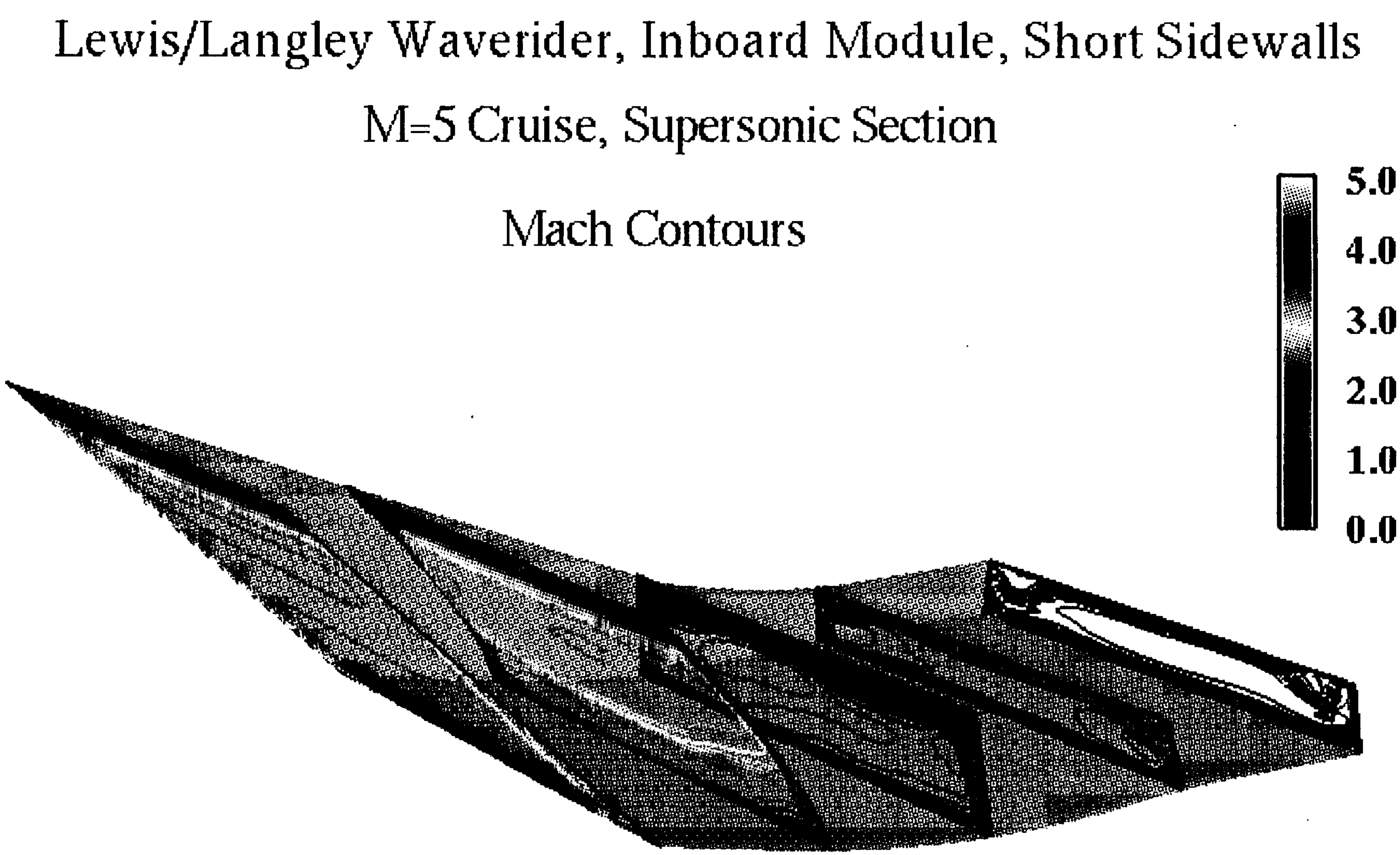

FIGURE 2 Mach number contours shown in representative crossflow planes for the internal flow portion of the supersonic inlet with short sidewalls.

a) Sidewall leading-edge plane to exit

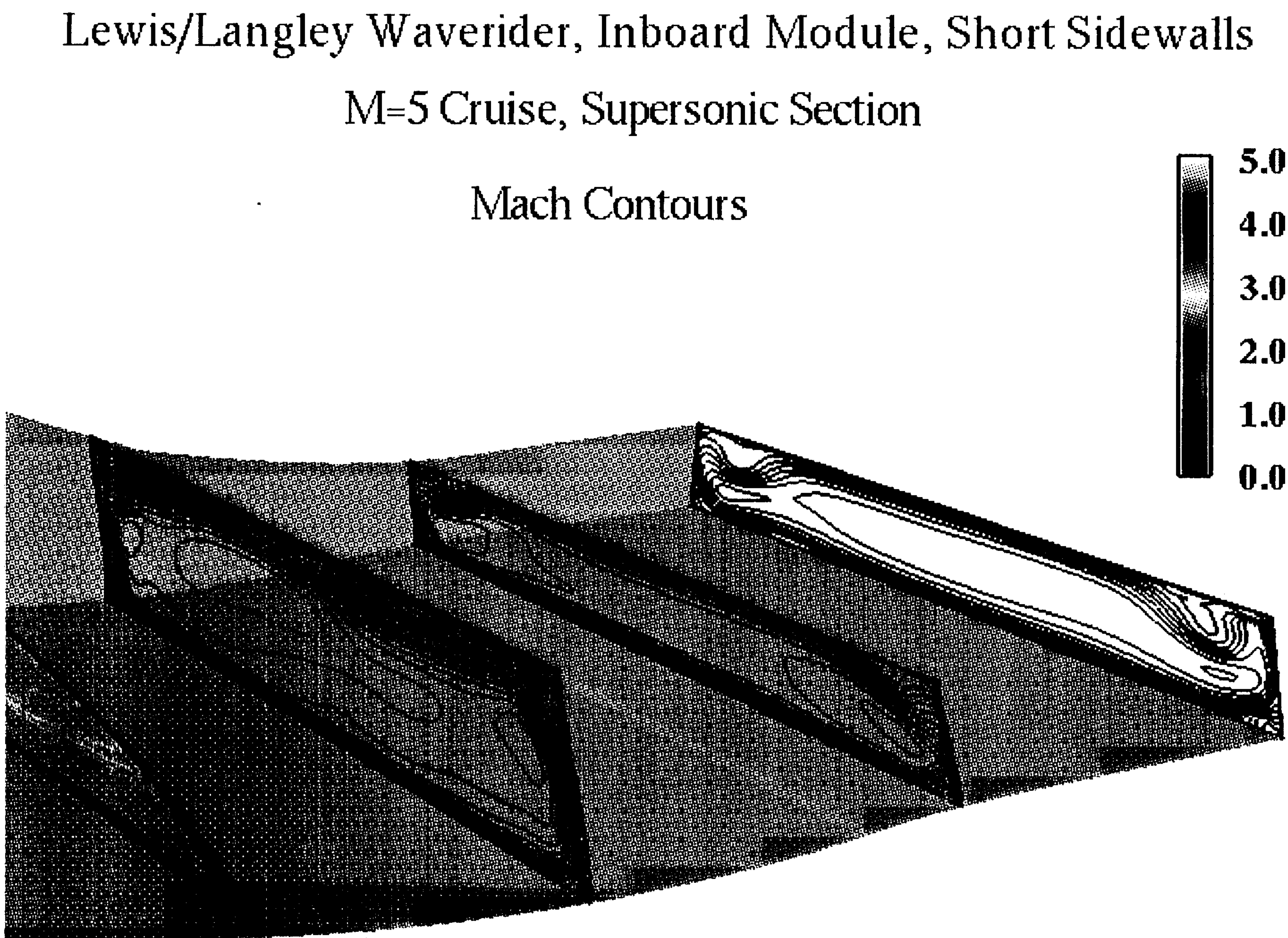

FIGURE 2 Concluded.

b) Mach number contours near the throat section of the supersonic inlet

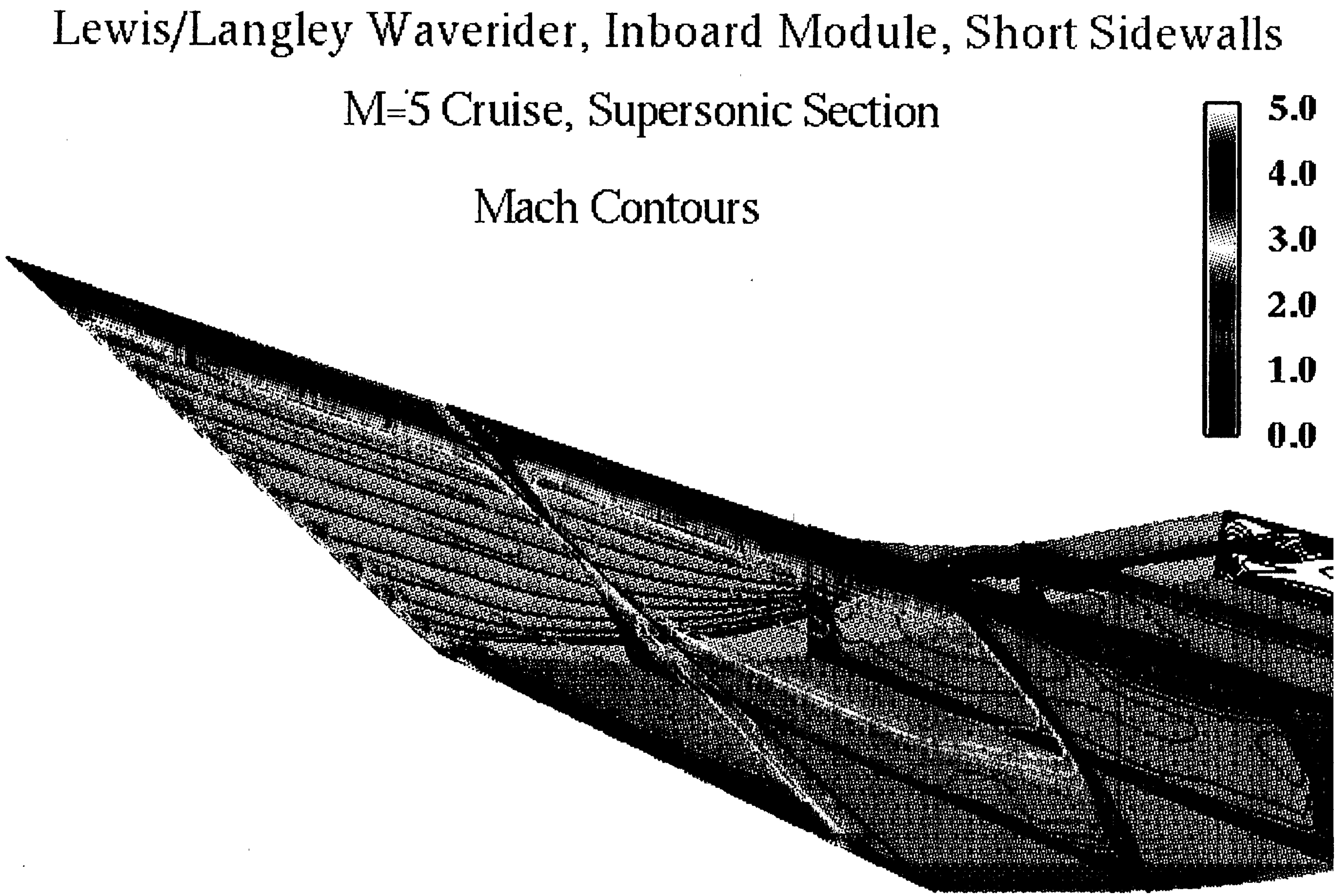

FIGURE 3 Mach number contours and particle traces showing the origin and evolution of the vortical flow feature in the supersonic inlet.

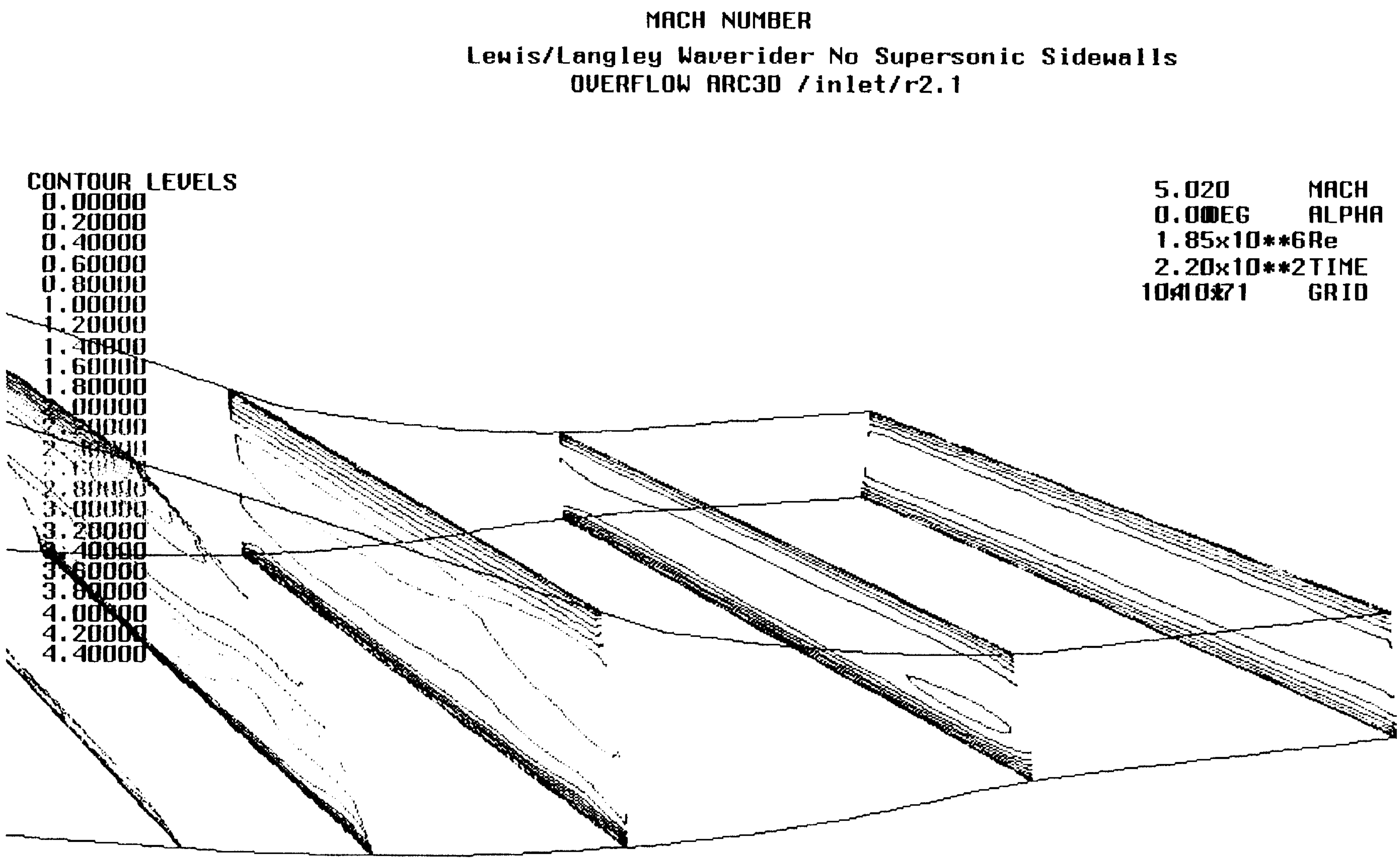

FIGURE 4 Mach number contours near the throat of the supersonic inlet for no sidewalls.

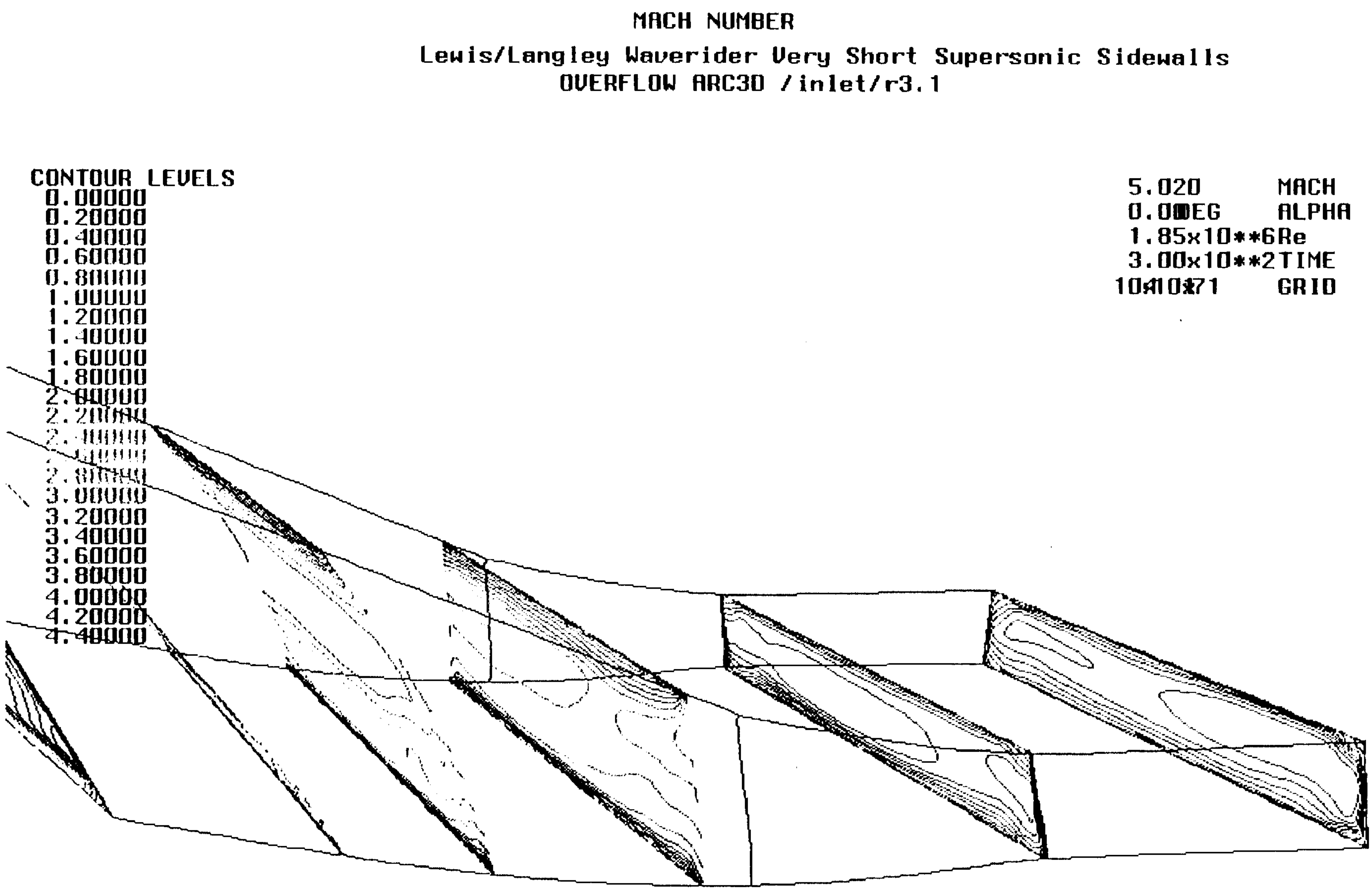

FIGURE 5 Mach number contours near the throat of the supersonic inlet for very short sidewalls.

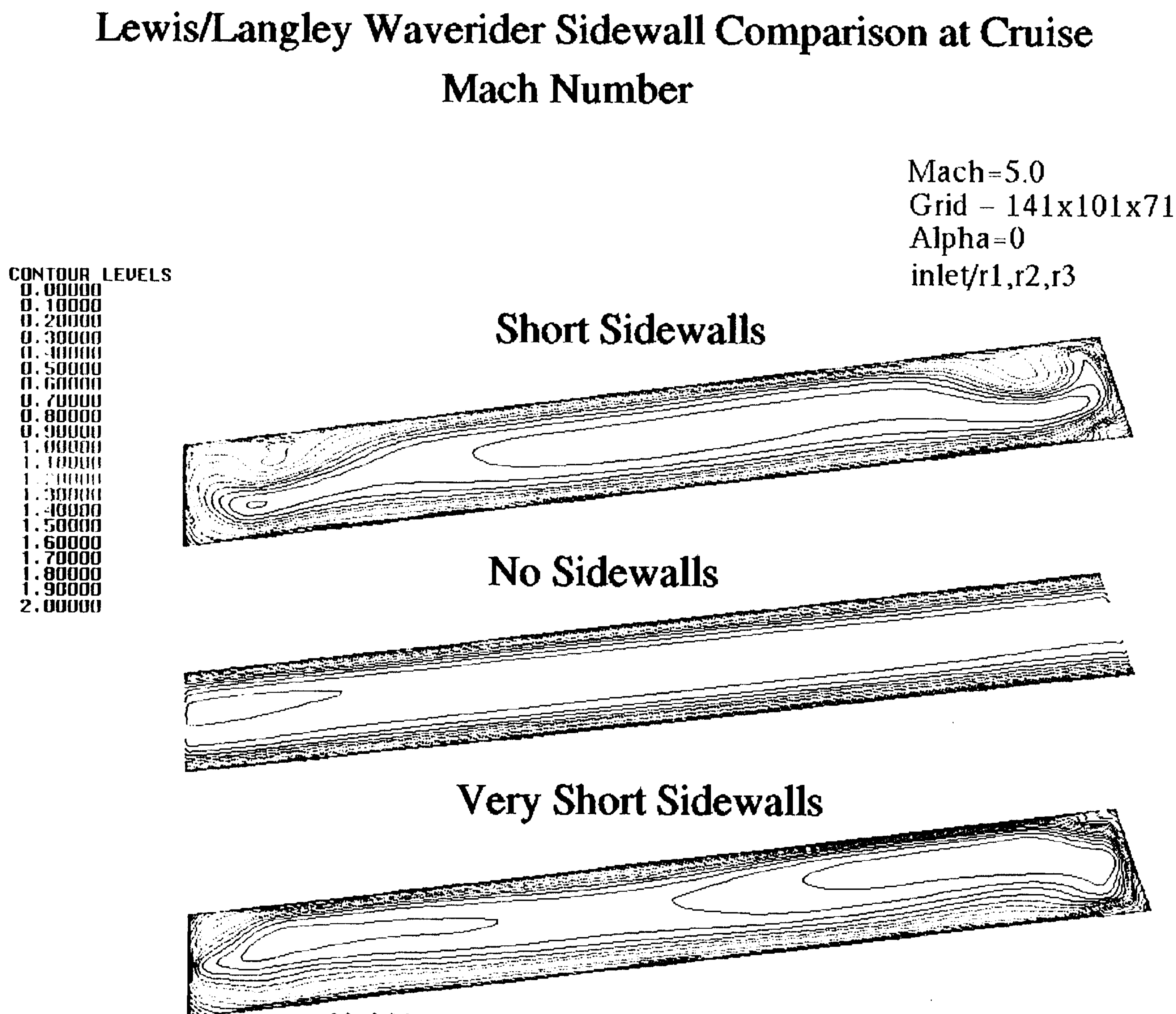

FIGURE 6 Mach number contours for short sidewalls, no sidewalls and very short sidewalls shown at the exit of the supersonic inlet.

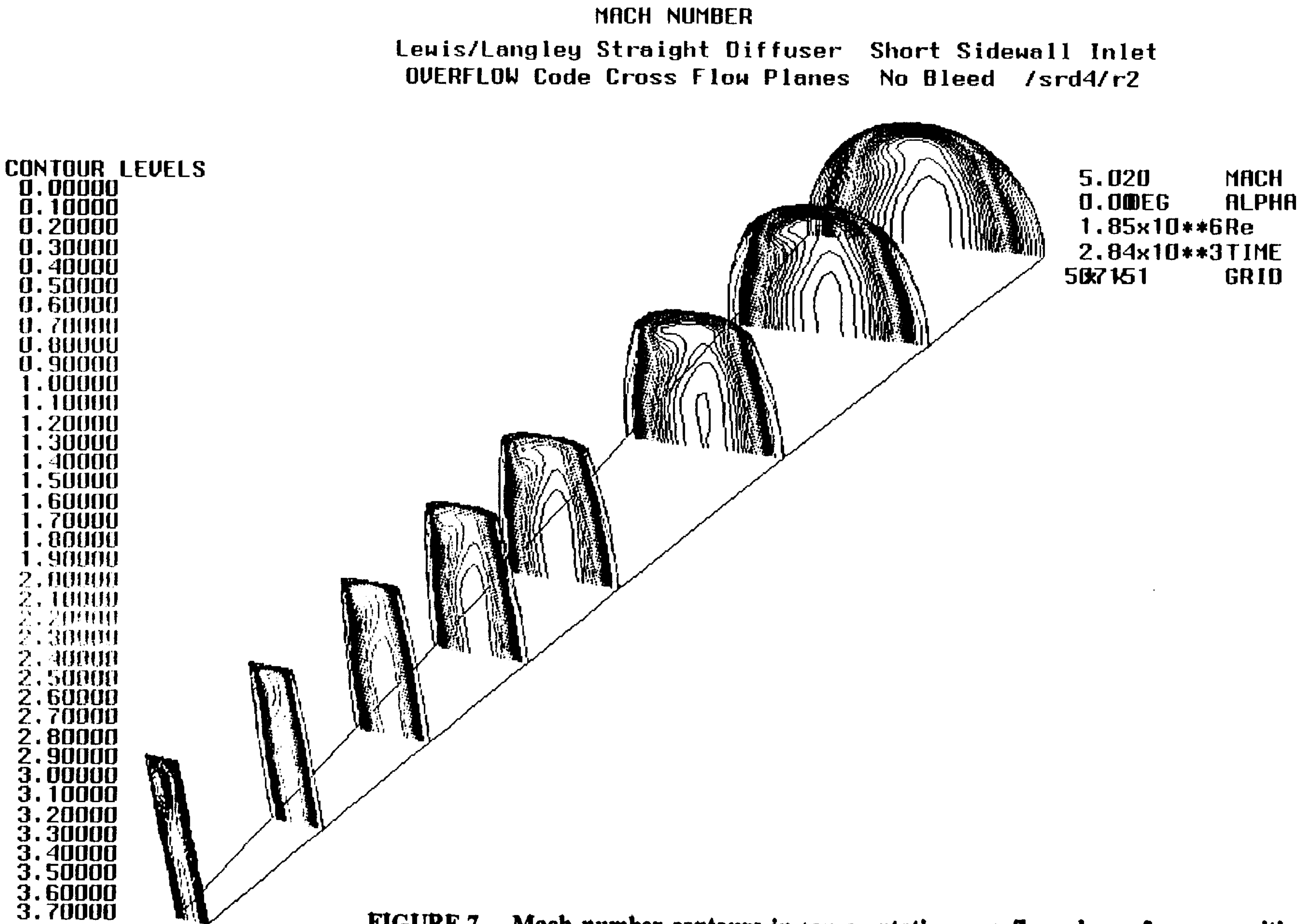

FIGURE 7 Mach number contours in representative crossflow planes for supercritical operation throughout the diffuser.

a) Short sidewalls

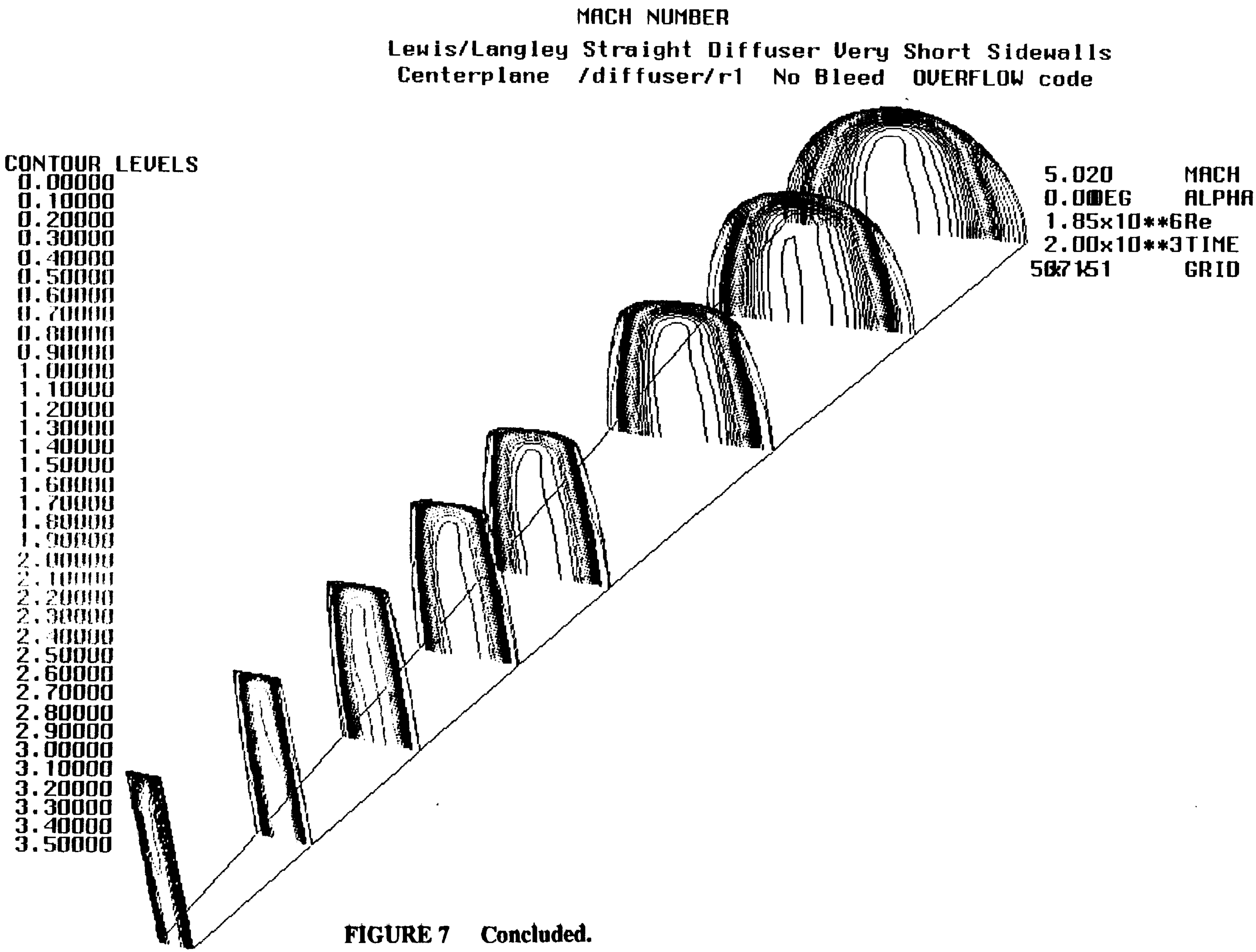

FIGURE 7 **Concluded.**

b) Very short sidewalls

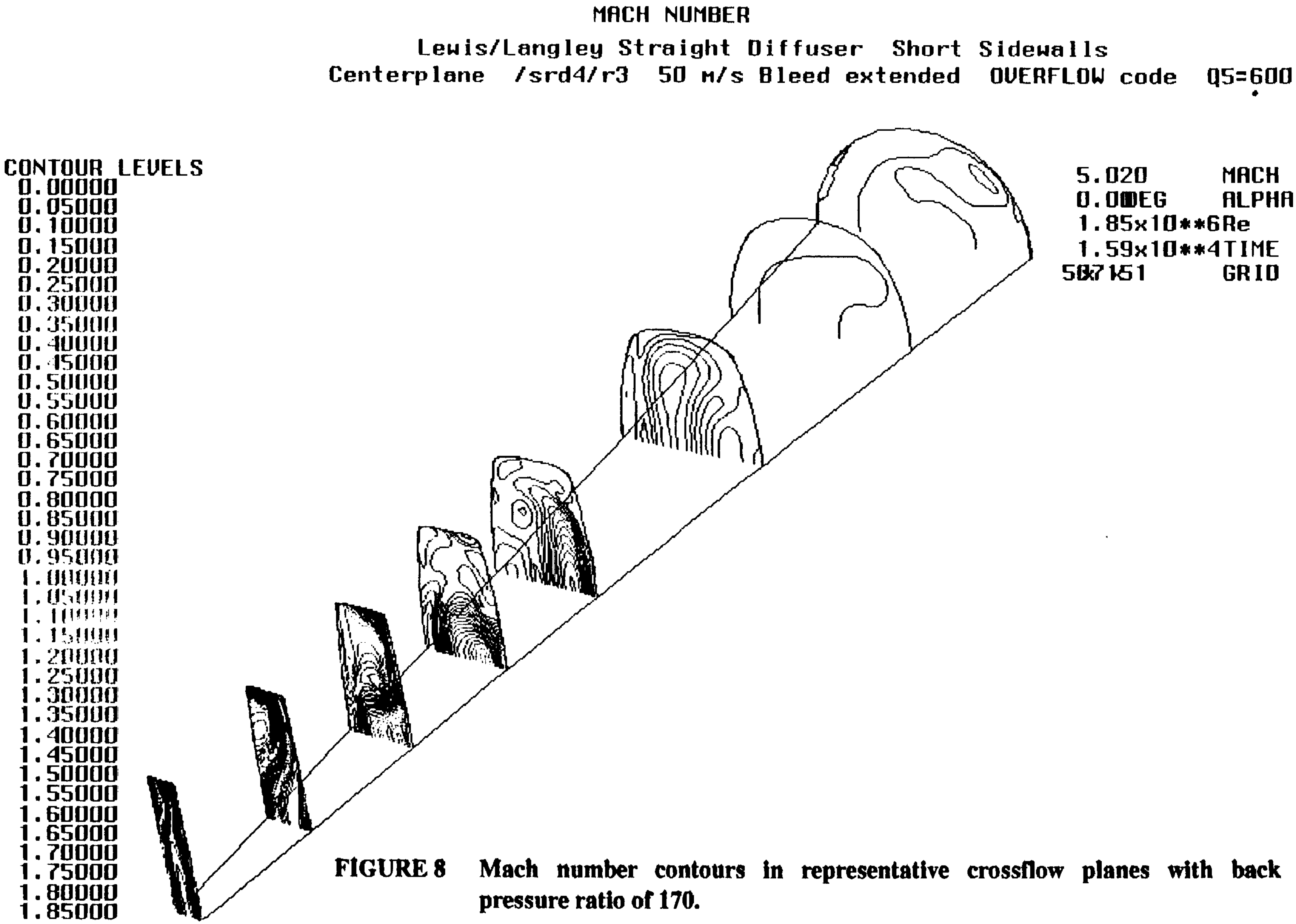

FIGURE 8 Mach number contours in representative crossflow planes with back pressure ratio of 170.

a) Short sidewalls

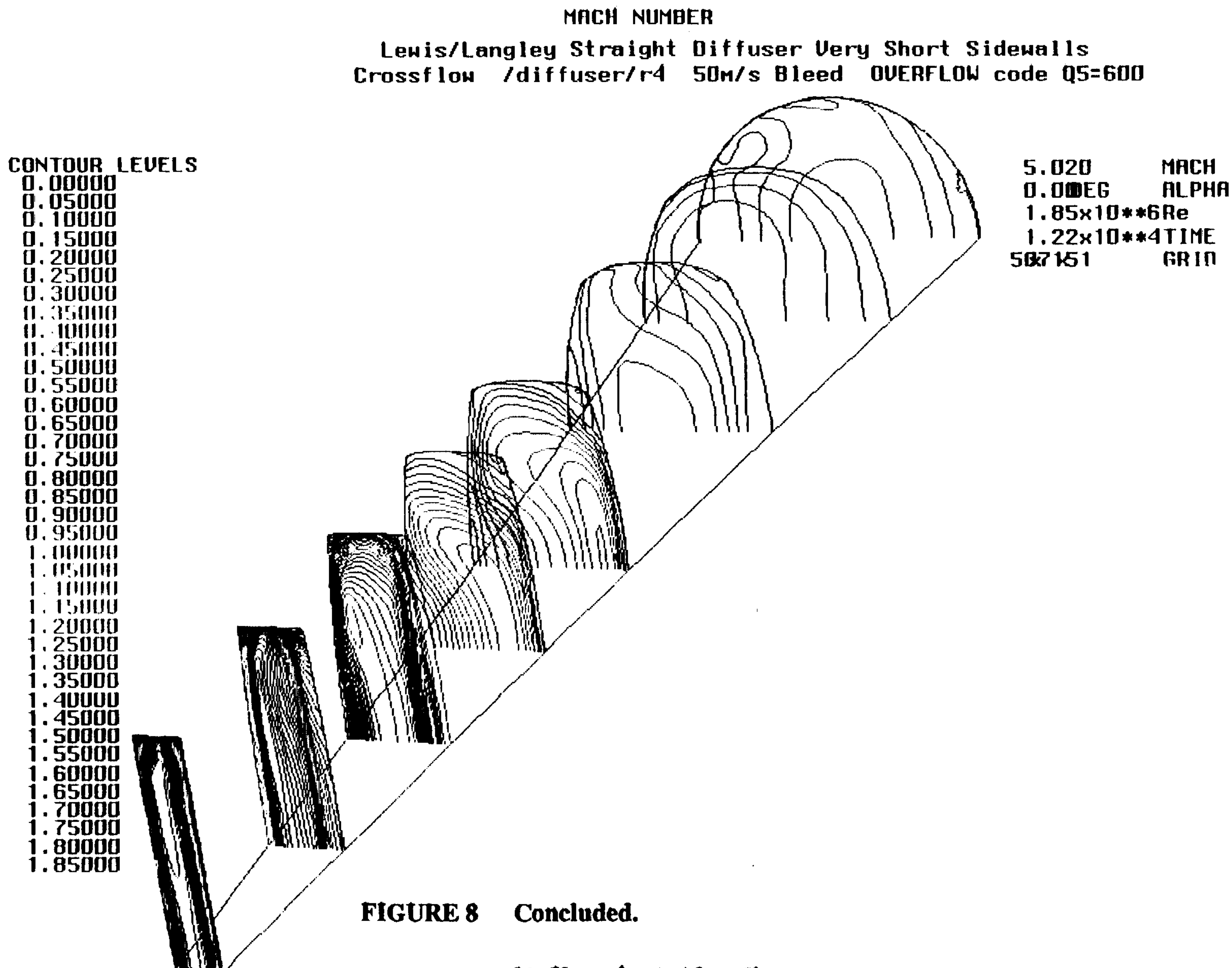

FIGURE 8 Concluded.

b) Very short sidewalls

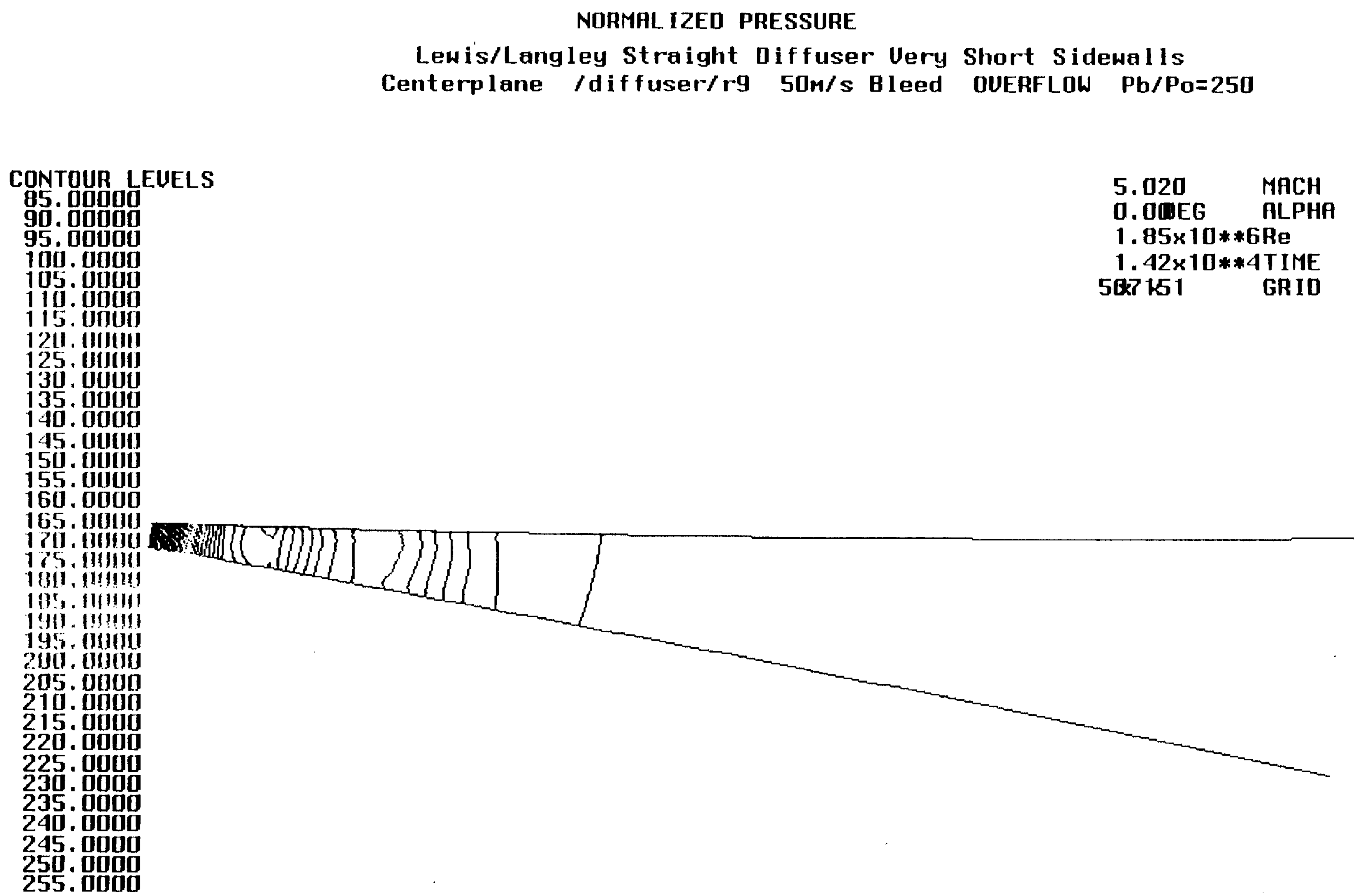

FIGURE 9 Continued.

b) Pressure contours on the symmetry plane

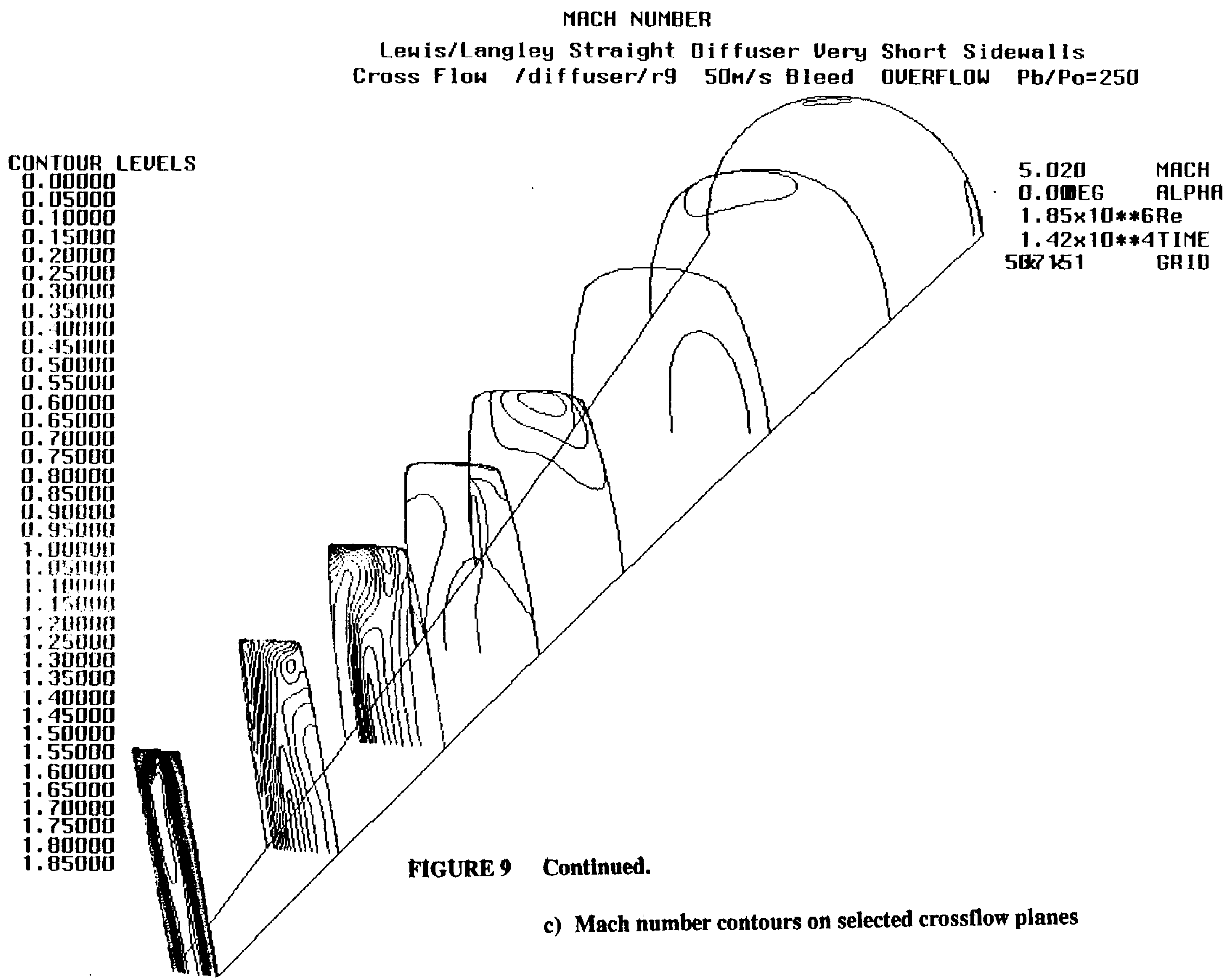

FIGURE 9 Continued.

c) Mach number contours on selected crossflow planes

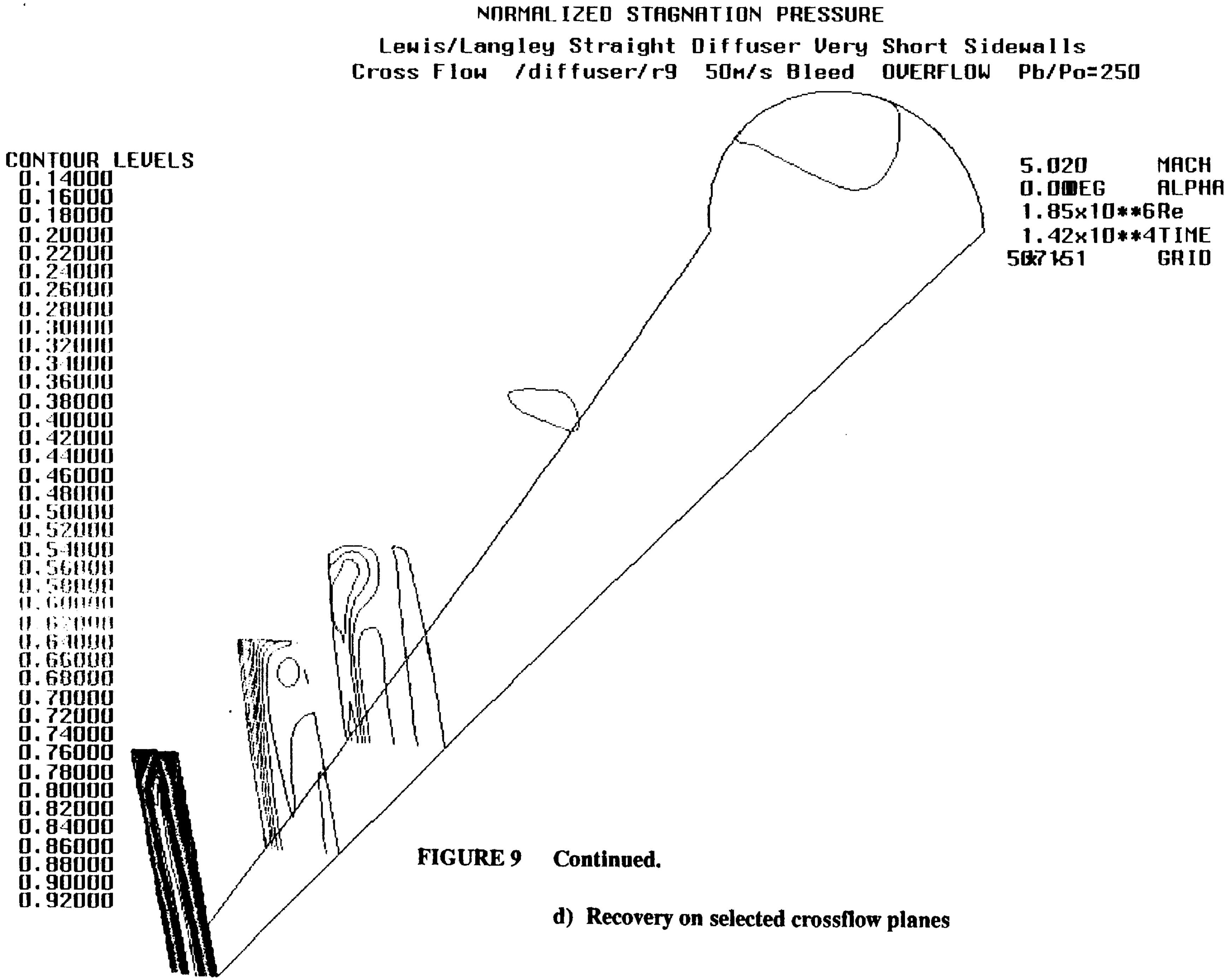

FIGURE 9 **Continued.**

d) Recovery on selected crossflow planes

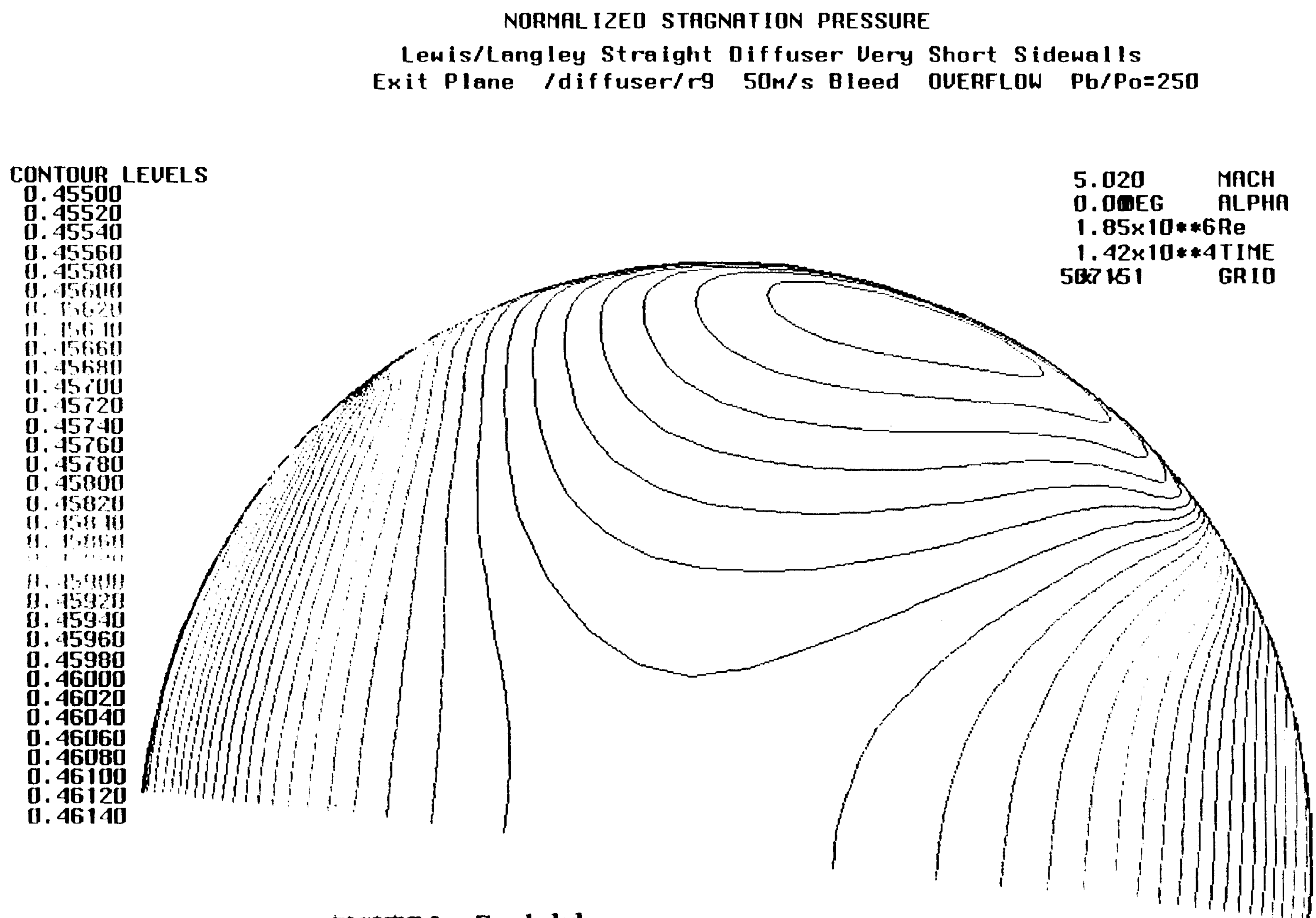

FIGURE 9 Concluded.

e) Contours of recovery at the diffuser exit (engine face)