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REPORT NO. RN-DR-0055

MULTIPLE EXHAUST NOZZLES FOR
NUCLEAR ROCKET ENGINES

MARCH 1965

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AEROJET-GENERAL CORPORATION

SACRAMENTO, CALIFORNIA

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NUCLEAR ROCKET ENGINES

ROCKET ENGINE OPERATIONS - NUCLEAR

MARCH 1965

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ABSTRACT

This report presents the results of studies conducted to evaluate the feasibility of expanding nuclear rocket reactor exhaust gases by means of multiple exhaust nozzles. The studies were performed under company-initiated Independent Research and Development.

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C. C. Ross
Vice President - REON

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I. INTRODUCTION

The primary objective of this study was to investigate the multiple-nozzle concept for solid graphite nuclear rocket application. Analytical procedures for nozzle design and theoretical performance prediction were developed. Because of the complex flow phenomena expected with the nozzle design, these analyses were limited to the assumption of ideal isentropic flow. Technique development for determining the effects of flow discontinuities was not considered within the scope of these studies. Instead, an experimental program was conducted to evaluate the effects of the flow disturbances relative to performance degradation. The experiments were devised so as to provide data which would substantiate the adopted analytical procedures for design and performance prediction.

II. SUMMARY AND CONCLUSIONS

A. The multiple-nozzle concept investigated in this study consists of a cluster of sixty asymmetric nozzles placed in a circular pattern around a single conventional nozzle. Each of these nozzles has a circular cross-section at the throat station, whereas the geometry of the exit plane is a sector of an annulus. The nozzle array, shown in Figure 1 was adopted so as to completely utilize the available area at the multiple-nozzle exits and, more important, to provide an enclosed region between adjacent nozzles which would serve for coolant passages. In this concept, the overall nozzle area ratio at the exit station of the cluster is nearly equivalent to that of a conventional nozzle of the same general geometric dimensions. The nozzle design, moreover, incorporates a conical shroud downstream of the multiple nozzle exit to provide additional nozzle flow expansion.

B. Analytical studies were conducted to determine nozzle coordinates and theoretical specific impulse. During these studies, an approximate numerical technique was devised for determining those nozzle geometries which would minimize flow disturbances within the asymmetric nozzles. Moreover, a geometric design was adopted having a small, flow-divergence angle at the nozzle exit so as to decrease the effect of exhaust-flow impingement from adjacent nozzles.

C. An experimental program was conducted to provide verification for the analytical design method. The experimental data agreed with the theoretical analysis to approximately one percent both for single nozzle and for multiple-nozzle performance. The measured data indicated the theoretical calculations to be slightly pessimistic because of the **stronger** influence on nozzle performance from the pressure rise in oblique shock waves generated inside of and at the exit of the individual nozzles. The computer programs used for theoretical analysis contained no provision for shock wave consideration, and predicted performance losses on the basis of imperfect, or non-optimum flow expansion. Whereas for single nozzle consideration, the shock losses are considered to be associated with performance losses, the experimental data indicate that the shrouded multiple-nozzle system actually derives a measurable performance benefit from the effects of the juncture shocks at the shroud surface.

D. The theoretical predictions for nozzle performance indicated nozzle efficiencies of 96.4 percent for a representative single nozzle, 94.4 percent for the unshrouded array, and 94.3 percent for the shrouded system. Vacuum thrust coefficients were 1.592, 1.557, and 1.578, respectively. Experimental data provided respective magnitudes of 97.1, 94.9, and 95.5 percent for thrust efficiency and 1.61, 1.567, and 1.594, for optimum thrust coefficient.

E. The result of both theoretical and experimental analysis of the multiple-nozzle design concept shows that this concept will result in minimum nozzle over-all length with minor performance loss. For the NERVA engine application a weight reduction of approximately 35% is possible when compared to an equivalent conventional convergent-divergent nozzle. The overall increased engine payload based on interstage weight reduction would be about 1930 pounds.

III. TECHNICAL DISCUSSION

The multiple nozzle concept, applied to nuclear rocket engines, has a primary advantage in reducing heat transfer through the utilization of refractory nozzle materials. Refractory materials, such as tungsten, tantalum and graphite, can be fabricated in nozzle configurations with limited dimensions. Thus for large flow areas the use of many smaller refractory nozzles is feasible under the present state-of-the-art technology. The reduced heat transfer serves as a performance advantage to the nuclear engine in that less pump pressure is required and more of the heat is available for conversion to thrust.

Additional multiple nozzle advantages include a shortened engine length thus allowing a reduction in the interstage structural weight. The integration of multiple nozzles in an elliptical thrust chamber head (pressure vessel) configuration allows a significant nozzle weight savings. This latter advantage is primarily a result of the typically large convergent area ratio associated with nuclear rocket engines.

A. MECHANICAL DESIGN

1. Geometry Selection

For this study it was first necessary to determine the overall configuration of the nozzle array, and then to design each individual nozzle. There are three general configurations possible when considering the multiple nozzle concept for nuclear rocket application:

1. The use of individual nozzles opposite or attached to each reactor core fuel element (1)*.
2. The use of an individual nozzle on each gas passage in each fuel element.
3. The use of a short plenum chamber, in which the gas issuing from the reactor core exit is mixed, followed by a nozzle "plate".

*Number in parenthesis indicate references listed in Section V of this report.

There are advantages and disadvantages associated with each concept, and all three were considered in some detail for this study. Schematic diagrams of the three configurations appear in Figures 2, 3, and 4.

Configuration (a) shown in Figure 2 features the reactor support concept utilizing a cooled metal base plate and appears naturally adaptable to the multiple nozzle design. Initial base plate designs involved hot-gas-flow passages through the base plate for each core module. Two serious problems are introduced by this concept:

1. The loading on the base support plate is multiplied by a factor of five. The base plate must be reinforced by increasing its thickness an unreasonable amount resulting in an excessive weight penalty.

2. A sonic throat restriction at the exit from each core module imposes an additional hot spot factor on the reactor. The throat tolerance appears as a squared quantity since it restricts the flow but does not influence the heat generation in the core module.

The primary disadvantage of Configuration (b) shown in Figure 3 would be the sizing of individual nozzles to match each reactor core fuel module and its associated power and flow rate. Also, as in the case of Configuration (a) the sonic throat restriction imposes an additional hot spot factor on the reactor. An additional problem area is ablation or distortion of the individual nozzle throats.

To avoid sizing each fuel element flow passage it is desirable to locate a mixing chamber between the core and the nozzles as illustrated in Figure 4. However, as in Configuration (a) loading on the flat support plate becomes excessive so that extensive reinforcing by increasing the plate thickness is required.

A first approximation in avoiding the aforementioned problems associated with Configuration (c) is illustrated in Figure 5. The primary load bearing member is in the form of an elliptical head and is separated from the reactor core exit such that a mixing chamber is located upstream of the nozzle

entrance section. The axial alignment of each nozzle was retained; the perforated elliptical chamber head thickness was selected considering the radiation heating and each nozzle was standardized with respect to the entrance, throat, and exit area. The result was an excessively long entrance section at near sonic velocities and concomitant nozzle efficiency losses.

A modification of the axially oriented nozzles in an elliptical head involved reducing the entrance length to that necessary to fit the cooled chamber head dimensions. This modification is illustrated in Figure 6. The result is a variation in entrance losses such that, unless the throat and exit areas were also modified, the supersonic streams impinging at the exit would result in further losses thus negating the effect of the initial modification.

It appeared that the multiple nozzle concept could not be reasonably designed with all nozzles of the same configuration axially oriented with the engine centerline. Each annular ring of nozzles should have a different exit geometry. By orienting the nozzle throat planes tangent to the chamber head surface, all entrance configurations were made the same with a 2:1 convergence area ratio such that entrance losses are minimized. This configuration is illustrated in Figure 7.

The selected configuration consists of a cluster of sixty asymmetric nozzles placed in a circular pattern around a single conventional nozzle. These nozzles have a circular cross-section at the throat station, whereas, the geometry of the exit plane is a sector of an annulus. This configuration was selected to completely utilize the available area at the multiple nozzle exit plane, and, more important, to provide an enclosed region between adjacent nozzles for coolant flow passages. Using this concept, the overall nozzle area ratio achieved at the exit station of the cluster is nearly equivalent to that of a conventional nozzle of the same geometric dimensions. Additional flow expansion to the desired area ratio is accomplished by a conical shroud downstream of the multiple nozzle exits.

2. Thermal Design Evaluation

The multiple nozzle configuration selected for this study was considered in preliminary detail for thermal design. Figure 8 illustrates a suggested detail for thermal design of the chamber head in the region of one of the nozzles. The tungsten nozzle would have a zirconia (ZrO_2) layer on the outside surface in the region of the cooled chamber head structure. The cooled metal structure is also coated on the exterior surfaces exposed to the hot gas or radiant heat flux (σT^4) from the nozzle expansion shroud. Coolant flow is directed through an annular array of slots around each nozzle so as to insure integrity of the chamber head structure.

The results of a one-dimensional heat transfer analysis are indicated in Figure 8 along with the basic dimensions and flow parameters assumed for the system. Note that the maximum radial temperature differential within the tungsten is at the thick throat region. The two layers of ZrO_2 have reduced the heat flux to less than 3 Btu/in.²-sec and the material temperature differentials are less than 500°R for the tungsten and less than 400°R for the Inconel wall.

The tungsten temperature differential is somewhat excessive even though the thermal stress loading is compressive. Further design consideration with respect to insulation, material thicknesses and coolant flow parameters is required for this subject. The use of Ta-10W would relieve the thermal stress. Also of concern is the relative dimensional change between the cooled structure and the tungsten nozzle. As assembled, the nozzle would be loose within the chamber head. In order to insure proper thrust orientation of the individual nozzles some locking mechanism or configuration compromise must be arranged.

The total heat transfer to the coolant of the multiple nozzle is, only 1.5 Mw when considered for NERVA engine application. This is less than 10% of that of a conventional convergent-divergent nozzle. The result is that the pump discharge pressure may be reduced considerably and also the core entry temperature is much lower, 150°R into the core versus the present anticipated 220°R. The increased hydrogen density perturbs the reactivity effect considerably and also is a strong influence on the orifice sizing.

3. Assembly Concept

Initial nozzle designs involved a threaded joint in a thickened region at the throat. The circular entrance configuration allowed the two pieces to be joined, thus clamping the nozzle in the chamber head. This concept is illustrated in Figure 8.

The loose ambient fit and the alignment problem, mentioned above, suggests an alternative. The small nozzle would be fabricated in one piece with a configuration allowing the entrance section to be inserted through the chamber head opening and then be wedged into place on the chamber side. Figure 9 illustrates this concept. The tungsten expands when heated and "climbs" the slope of the shrinking metal structure upon which it rests. The exit cone, half angle of 30° serves to orient the nozzle for proper thrust alignment on assembly and also during the startup thermal transient. If friction coefficients are not excessive, the expansion of the exit cone will follow the 30° slope of the cooled surface.

The problems of friction coefficients and associated influences of extreme temperatures, radiation effects, hydrogen atmosphere, and possible use for extended times in a vacuum environment must be investigated. Also, fabrication tolerances and quality control procedures and instruments must be considered. Additional features with respect to materials for the various components are of concern. Tungsten, zirconia, graphite and Inconel, which have been mentioned, are not necessarily established for the multiple nozzle concept.

B. ANALYTICAL PROGRAM

A study was conducted to investigate the proposed nozzle concept with a primary objective being to develop analytical procedures for asymmetric nozzle design. Because of the complex flow phenomenon expected with the asymmetric nozzle design, these analysis were limited in that ideal isentropic flow was assumed.

An exact analytical procedure necessary to define flow characteristics within an asymmetric nozzle geometry was considered too complex and time consuming

for preliminary evaluation of the proposed multiple nozzle concept. Therefore, an approximate method was adopted which basically assumes that internal flow definition can be accomplished by using existing numerical techniques. Specifically, the existing methods assumes symmetric flow expansion and use of the method of characteristics solution of the potential flow equations. Using results of these calculations, the flow parameters and performance of the asymmetric nozzle were defined. Although this procedure is not considered applicable to asymmetric geometries, it does provide an analytical tool with which to obtain reasonable configurations and performance predictions for preliminary evaluation.

An experimental program was conducted to provide verification for the analytical design method. Because the experimental models differed in design from those used in the initial analysis, the flexibility of the analytical method was demonstrated in a program revision, accounting for experimental nozzle geometry that resulted in performance predictions agreeing to within approximately one percent of experimental data.

1. Nozzle Geometry

a. Axisymmetric Nozzle

The design of the axisymmetric nozzle, located in the center of the multiple nozzle array, was established using an existing digital-7090 computer program. This program computes the "optimum" thrust nozzle geometry and performance using Rao's technique (2). The nozzle exit flow angle was restricted to be less than 5 degrees so as to maintain a low flow impingement angle with the exhaust of the first row of asymmetric nozzles. The coordinates of the nozzle and parametric nomenclature is given in Table I.

The nozzle expansion ratio of 11.3:1 was determined by geometric dimensions specified in the nozzle layout. Because the nozzle design was restricted to have a low divergence angle at the exit, near maximum performance was achieved for this expansion ratio. That is, the specific impulse is nearly equal to that of a parallel flow nozzle having a flow direction parallel to the nozzle axis. Theoretical specific impulse calculated is 757.7 lb-sec/lb.

(b) Asymmetric Nozzle

The coordinates of the asymmetric nozzle were obtained by assuming that nozzle internal flow properties are similar to those defined by symmetrical expansion about a centerline. In an axisymmetric nozzle, the flow can be considered a family of stream-tubes, as illustrated in Figure 10, each containing a portion of the total mass being exhausted. Because of continuity, therefore, the outermost boundary of each stream tube will represent coordinates of the physical wall containing the appropriate mass.

Supposing the asymmetric nozzle to be contained within a symmetric nozzle, shown schematically in Figure 10, it can be assumed that the total mass exhausting in the asymmetric nozzle will be enclosed by that family of streamlines defined by the radial coordinates of the asymmetrical nozzle at the exit plane. The basic procedure for individual nozzle design was to compute streamlines within an axisymmetric nozzle having the same expansion ratio as the desired asymmetric nozzle. The analytical method used to define flow streamlines, by use of the method of characteristics solution of equations of fluid motion, is described in Reference (3). The exit area of the axisymmetric nozzle was determined by the maximum radial distance (r_m) existing at the exit as the asymmetric nozzle. Along each radial plane, the asymmetric nozzle coordinates were established as the coordinates of the streamline passing through the point on the periphery at the exit. Using this procedure as a basis, coordinates at various orientation angles (ϕ) were determined with consequent definition of the entire nozzle configuration.

If the streamlines used in defining wall contours of the asymmetric nozzle are followed into the throat region, the cross section at the throat station would have a geometric shape similar to that of the exit plane. It was necessary, therefore, to deform the streamline coordinates in the throat region so as to obtain the desired circular cross section. This was accomplished by considering source flow conditions to exist in the vicinity of the throat and determining a common source point which would result in source flow lines tangent to both the streamline and the external throat curvature (r_d).

From geometric considerations (Figure 11), the following equations are pertinent:

$$X_t = \frac{R}{\tan \theta_i} - l \quad (1)$$

$$r_d = \frac{l}{\tan \theta_i} + (r - r_t) \quad (2)$$

$$r_d = \frac{l^2 + (r - r_t)^2}{2 (r - r_t)} \quad (3)$$

where X_t = axial distance from source point to throat section
 R = radial coordinate of inflection point
 l = axial coordinate measured from the throat
 r_t = throat radius
 r_d = external throat radius
 r = radial coordinate along the source flow line
 θ_i = local wall angle at the inflection point or angle of source flow line.

The first expression is determined from geometric considerations and conditions along the streamline. By an iterative procedure, the common source point was established to yield a reasonable range of external throat radius for each nozzle geometry. External throat radius was determined along each radial plane by simultaneous solution of equations (2) and (3). To expedite computations, a digital computer program (GE 225) was devised to accomplish the detailed calculations.

Using the method described, wall coordinates for the four asymmetric nozzles were computed and are presented in Table II. The contour coordinates are presented as a function of angular orientation (ϕ) about the nozzle axis.

2. Estimated Performance

The thrust of a nozzle can be obtained by computing the momentum of the flow across the exit area. Normally, the momentum equation for exhaust flow that is not parallel to the nozzle centerline is expressed as

$$F = A (\rho \mu^2 \cos \theta + P) \quad (4)$$

This expression can be converted to an impulse function by substituting the following gas-dynamic relation

$$\rho \mu^2 = \gamma P M^2 \quad (5)$$

Therefore, local thrust can be computed by use of the expression

$$F = \int_{A_{\text{exit}}} P (1 + \gamma M^2 \cos^2 \theta) dA \quad (6)$$

where F = Axial thrust component
 ρ = Local gas density
 P = Local static pressure
 θ = Local flow angle
 γ = Ratio of specific heats
 A = Axial projected area
 M = Local Mach number
 μ = Local velocity

The procedure used to evaluate the thrust produced by an asymmetric nozzle consisted of integration of equation (6) across the exit plane. Properties of the exhaust, such as pressure, Mach number, and flow angle, were determined from exit flow data of the axisymmetric nozzle. The numerical integration, programmed on the GE 225 Computer, consisted of a summation of thrust contributions from incremental areas of the asymmetric nozzle plane. Individual nozzle thrust

was, subsequently, corrected for angular displacement of the nozzle centerline relative to engine centerline to obtain total axial force produced by the multiple nozzle array. Specific impulse was determined by dividing total axial thrust by total mass flow as calculated using Sauer's (4) technique for evaluating throat flow conditions. A complete listing of operating conditions and overall nozzle geometric parameters is given in Table III.

C. EXPERIMENTAL PROGRAM

A series of cold-flow experiments was conducted to evaluate actual performance of the proposed multiple-nozzle concept. The program was designed to obtain quantitative and qualitative data defining both performance of the nozzle system and the effects of asymmetric expansion and adjacent nozzle interaction on the characteristics of nozzle flow. The nature of the resulting experimental data was planned to parallel the procedures and results of the analytical approach and so allow comparison of intermediate data. Because of the established validity of existing methods for design and performance prediction for axisymmetric nozzles, only the asymmetric system components were experimentally evaluated.

All experiments for the multiple nozzle study were conducted in the Elk River facility of the Fluidyne Engineering Corporation, Minneapolis, Minnesota. Force and pressure data and shadowgraph pictures were obtained during cold flow tests of portions of the test article.

1. Facility Description

a. Static Test Stands

Two different static test stands, Figure 12, were used to accommodate the test requirements of the subject study. The major differences between the two test stands are their size and consequent capability of accurate control of various air flow rates.

Both static test stands were designed specifically for high area ratio nozzle testing with direct force measurement. The larger of the two test stands, Channel 7, was used in testing the 10 nozzle configuration over the range of pressure ratios (model total pressure to ambient pressure) from about 100 to 210. The smaller test stand, Channel 8, was used to obtain shadowgraphs of the nozzle exhaust from the 3-nozzle configuration and to obtain three-component force data and pressure data for the single nozzle configuration at pressure ratios from approximately 60 to 270.

Supply air to the static test stands was obtained from a 500 psi storage source of 2370 cu. ft. capacity. The inlet controls and flow metering station for the test stands have interchangeable components (valves and meters) for proper matching of the facility to the test article. The low test chamber pressures that can be achieved with these facilities are obtained by three stages of exhaust ejection. The first stage decelerates the exhaust stream from the test article in a carefully matched diffuser section. The second and third pumping stages are through specially designed ejectors using air and steam respectively as primary fluids (no air ejector was used in channel 7 for this program). Gross control of test chamber pressure (ambient to the test model) is exercised by regulation of the primary air and steam pressures to the ejectors and by throttling of the flow from the first stage exhaust diffuser. Fine control of test pressure ratio is achieved by a shielded inbleed of atmospheric air into the test section. This fine control can be quickly adjusted during a run when test conditions are being established.

b. Force Balances and Force Readout System

The 10-nozzle configuration was mounted in the test chamber of Channel 7 on a single component axial force balance. The 3-nozzle configuration was mounted to the test chamber of Channel 8 without any force balance since no force data were to be obtained. The single nozzle configuration was mounted in the test chamber of Channel 8 on a three-component force balance designed specifically for vectored tests.

The components of force measurement are defined relative to the axis of the inlet supply air by the sketch of Figure 13. Structural isolation from the inlet air ducting is through a thin, rubber, membrane seal. Calibration of balance and seal combination is closely associated with the data reduction procedure described in Section C.2, below.

The outputs from the balance strain gauge elements were measured by a multi-module data system. Up to ten separate sources of electrical inputs from strain elements, thermocouples, or pressure transducers can be measured with a single module of this system with the outputs printed in digital form on sensitized tape. Either one or three of the measurement channels were used for the components of force resolution and an additional channel was used to monitor a known input signal to preclude the possibility of any unexpected shift in system adjustment being undetected.

c. Other Instrumentation

Other instrumentation committed to the program included 120 tubes of manometry and several precision pressure gauges in the 0.1% accuracy range. All static pressure measurements required on aerodynamic boundaries within the model or force balance system were recorded on mercury or silicone oil manometry. The manometer tubes using silicone oil were gated to preserve the integrity of the outgassed fluid for measurement at low pressures. All other pressure measurements were taken on either high pressure Heise gauges or Wallace and Tiernan vacuum gauges. Standard laboratory instrumentation was used for temperature and alignment measurement when required. Shadowgraphs of the model exhaust were taken to aid in the analysis of the data.

2. Balance Calibration

Two basic types of calibrations are required for complete definition of the axial and the vertical readout obtained during a test. The first calibration provides a relation between the horizontal force sensed by the balance and its corresponding readout. The second calibration gives a similar relationship between vertical force and location and the readout of the two vertical strain elements.

a. Ten-Nozzle Configuration

The ten-nozzle configuration was calibrated along the horizontal axis of a single component force balance to determine a set of calibration constants for calculation of the overall thrust coefficient from test measurements.

The seal used to separate the nozzle and balance assembly from the inlet duct is a simple pressure-tight expansion joint consisting of a thin rubber membrane lying over a small gap between two metal plates. Since the balance deflection under load is low, the stretch in the thin rubber seal is also low, and the tensile forces across the seal can be incorporated into the balance calibration.

To incorporate the seal tensile forces into the horizontal balance calibration, the balance is calibrated under simulated operating conditions as follows. The inlet duct is pressurized to test values of seal station differential pressure. A series of upstream loads are then applied to the balance by means of the weight-pulley arrangement, shown schematically in Figure 14, to simulate the test range of resultant loads and seal gaps. As the upstream load is applied, the digital output, C_2 is read. The apparent load on the balance $H_{20} = \Delta P A_3 - R$ is divided by the readout C_2 to determine the balance calibration constant, K_{20} . Values of K_{20} are plotted against their respective counts. During actual tests, the appropriate value of K_{20} was selected for the ΔP and C_2 values of the test. Thus the effect of seal tension (as a function of seal gap and pressure differential) is included in the calibration constants.

b. Single-Nozzle Configuration

Since the thrust vector for the single nozzle was not along the tunnel axis, the calibration required both a horizontal and vertical calibration. The horizontal calibration was similar to that for the ten nozzle configuration but performed over the appropriate load range.

The vertical calibration consisted of applying a series of vertical loads at a constant horizontal loading. This was done with horizontal loads which covered the balance loads during testing. The final results of the calibration were values of K_1 and K_3 , vertical force constants, which, when multiplied by the force readout in counts, yield the vertical forces V_1 and V_3 . Figure 15 shows the calibration setup and gives critical dimensions and distances applicable for the calibration calculations.

Summation of moments from Figure 15 for the single nozzle gives:

$$(V.L.)d_3 + (H.L.)d_2 - V_1 c_1 = 0$$

or
$$K_1 = \frac{(V.L.) (d_3/d_1) + (H.L.) (d_2/d_1)}{c_1}$$

also,
$$(H.L.)d_2 + (V.L.) (d_1 + d_3) - V_3 d_1 = 0$$

or
$$K_3 = \frac{(H.L.) (d_2/d_1) + (V.L.) (1-d_3/d_1)}{c_3}$$

where
$$V_1 = K_1 c_1 \text{ and } V_3 = K_3 c_3.$$

3. Test Procedures

The test program included evaluation of a typical single nozzle by itself and measurement of data for the multiple nozzle array. In addition, a special test installation of a three-nozzle group allowed shadowgraph definition of the interactive shock systems generated by adjacent nozzle flows. Data obtained in the single-nozzle experiment allow a direct comparison with the predictions of the nozzle design analysis; multiple-nozzle tests define the losses that can be anticipated from nozzle interaction.

The general procedure for conducting the test program consisted of opening the primary air supply valve to establish the desired nozzle total pressure. At the same time the ejector system lowered the nozzle ambient pressure. The desired nozzle pressure ratio is set by throttling a shielded inbleed from atmosphere into the test chamber. When steady-state conditions were sustained, force and pressure were taken simultaneously.

a. Single-Nozzle Experiments

A typical asymmetric nozzle (Nozzle E of Figure 16) was removed from the multiple array, instrumented with forty static pressure orifices, and installed on a three-component force balance in Channel 8 of the facility. Photographs of the instrumented nozzle and of the nozzle installation are shown in Figure 17; Figure 18 is a drawing of the test setup. Four test runs were made at pressure ratios (chamber to exhaust) of 101.14, 143.36, 187.97, and 263.21.

b. Three-Nozzle Shadowgraph Grouping

Nozzles B, C, and F (of Figure 16) were grouped in the same respective positions as they would be in the ten-nozzle array and mounted on a modified support flange in Channel 8 of the facility (Figures 19 and 20). Shadowgraphs of the exhaust flow interactions along nozzle junction planes in three views were obtained at a pressure ratio of approximately 130.

c. Ten-Nozzle Experiments

The multiple array representing a ten-nozzle sector grouping was tested with and without an exhaust expansion shroud in Channel 7 of the facility. The model was mounted on a single-component thrust balance with the balance centerline aligned with the effective motor centerline. The single axisymmetric nozzle (in the center of the motor) was omitted for these tests. Figure 21 shows the system with the shroud installed and Figure 22, the method of installation relative to the force balance and diffuser-ejector section. Tests, made for pressure ratios of 97.14, 129.62, 163.55, 208.49, 264.35, and 269.88 included measurements of monitoring shroud static pressures and flow visualization photographs of oil traces on the shroud flow surfaces.

4. Discussion of Test Data

A direct comparison of experimental model performance with the analytical predictions was not possible. Whereas the analytical approach resulted in the specification of a theoretically optimum design within restricted envelopes, the models evaluated experimentally were fabricated to substantially different specifications, generally resulting in shorter lengths and greater expansion ratios than considered in the analytical results. Actual measured parameters are listed in Table IV. For example, the expansion ratio for the single nozzle used in the experimental program appears to be 8.296, whereas for the experimental program the model had an actual expansion ratio of 10.3, a substantially larger value.

Revised theoretical nozzle performance data applicable to the test nozzles, are listed in Table V. Calculations were actually made only for Nozzle E; data for the other nozzles and for the 10-nozzle array were obtained from simple extrapolation on the basis of geometric area, and nozzle orientation angle. This was done by determining the incremental performance attributable to differences in expansion ratio for the individual nozzles. Theoretical performance for the ten-nozzle system was determined by an area-weighted average of the single nozzle data corrected for the individual nozzle thrust angles relative to the motor axial centerline.

a. Experimental Test Results

(1) Thrust Measurements

Nozzle experimental performance is presented in Figures 23 and 24 in the form of thrust efficiency and vacuum thrust coefficient respectively, as functions of nozzle pressure ratio. The coefficients are defined as follows:

$$C_{F_{vac}} = \frac{F}{P_c A_t} + \epsilon \frac{P_a}{P_c}, \text{ vacuum thrust coefficient}$$

and $\eta I = C_F / C_{Fopt}$, thrust efficiency

where F = Measured nozzle thrust

P_c = Measured nozzle total pressure

A_t = Nozzle throat area

ϵ = Nozzle expansion ratio

P_a = Nozzle exhaust ambient pressure

C_F = Thrust coefficient at pressure P_a

$= C_{Fvac} - (P_a / P_c)$

C_{Fopt} = Thrust coefficient based on ideal exit flow momentum at the nozzle pressure ratio and discharge coefficient.

The results of the theoretical calculations are presented as tagged symbols for the respective nozzle configurations on the same plot.

The experimental curves for thrust efficiency are observed to have maximum values at the theoretically optimum nozzle pressure ratio for each configuration and are within approximately one percent agreement with the theoretical data at this condition. This is a clear indication of the validity of the theoretical method for determining performance of asymmetrical nozzles of this category.

The experimental thrust efficiency of the single nozzle is approximately two-percent lower than that conventionally determined for a conical nozzle with a 15-degree half angle. This results primarily from the asymmetric local expansion and compression zones which are subject to description by pressure and visual data measurements. The agreement between theoretical and experimental performance is not consistent for the three different configurations, although the general agreement is approximately one percent. The theoretical prediction is pessimistic to this extent. Reasons for this condition will be developed in subsequent discussions of pressure and visual data. The major significance of the experimental data is in verification that although nozzle efficiency with and without shroud for the ten-nozzle array is relatively low because of the thrust vector cosine effect of the individual nozzles, vacuum performance is nearly recovered relative to the single nozzle by increasing overall expansion ratio through use of a shroud. Moreover, the

improvement through use of a shroud is experimentally shown to be approximately fifty percent greater than is theoretically predictable on the basis of expansion ratio increase. This is emphasized in the experimental improvement in thrust efficiency of the shrouded system over the unshrouded system.

In summary, two major conclusions can be reached from the foregoing discussion. The first is that there is no appreciable performance loss because of shock losses at the nozzle exit station when the nozzles are grouped in an unshrouded array such as that evaluated. The second conclusion is that thrust efficiency for the shrouded array is increased over the unshrouded system by 0.6 percent; the corresponding improvement in vacuum thrust coefficient (2 percent) is approximately 1-1/2 that which would be predicted theoretically on the basis of expansion ratio increase and constant thrust efficiency.

(2) Shadowgraph Data

The purpose of obtaining shadowgraph pictures of the nozzle exhaust flow was to define the strength of the jet interaction shock waves at the juncture of adjacent nozzles. The theoretical nozzle design resulted in a wedge angle (or half-angle) of approximately 5 degrees at the juncture station. It was expected that the shock strength would be defined by this wedge angle and the local flow Mach number of the shadowgraphs would give visual verification of the shock characteristics. However, several factors of actual model geometric design considerably altered conditions: shape and area relationships for the models severely differed from the theoretical designs. The most severe condition was in the existence of effectively zero wedge angle, for the last inch of nozzle length, preceded by a conic expansion section.

Figure 25 shows (conceptually) the nozzle shape midway between corners and the associated shock pattern for internal nozzle flow. The sketch is presented to illustrate the sources of the various shock patterns evident in the shadowgraph pictures of Figures 26 through 29. Referring to Figure 25, the only major shock wave having an appreciable effect on the performance of the individual

nozzles in the array is that generated at the transition from the conic cross-section near the throat to the asymmetric expansion section. This disturbance is strongest midway between the "corners" of the expansion section and does not exist in the corner regions. Similarly, the compression at the transition between angled and flat surface near the exit also occurs midway between corners, this latter shock system has but little area of influence inside the nozzle itself. The juncture shock, however, is strongest near the corners and weakest near the midway points, primarily because the flat surfaces become wedge surfaces in the corner regions and the flow turning required generates a substantial shock wave. The juncture shock system, therefore, has a strong influence for the configuration that includes an expansion shroud, for the shock static pressure increase has an effect at the shroud surface.

The shadowgraph pictures of Figures 26 through 29 show shock patterns for the single nozzle (effectively a tangential view of Nozzle E) and for a representative three-nozzle grouping of nozzles B, C, and F. The views of the three-nozzle system are:

Figure 27 - A view along the juncture of nozzles B and C for $\omega = -90^\circ$, where ω is an arbitrary view angle designation

Figure 28 - A view normal to the juncture line of nozzles B and C ($\omega = 0^\circ$)

Figure 29 - A view along the juncture of nozzles C and F ($\omega = +10^\circ$)

(3) Pressure Data

Static pressure measurements were made at orifices located in five axial rows in the expansion section of nozzle E and at selected locations on the internal shroud surface of the ten-nozzle array. The data were obtained primarily to define the flow characteristics along nozzle and shroud surfaces by detection of the presence and strength of shock waves.

Pressure distributions for the single asymmetric nozzle are presented in the form of Mach number in Figure 30. These include both measured data (symbols) and computed magnitudes for the theoretical analysis. The major

difference between theoretical and experimental distributions is caused by inability to follow, in theoretical analysis, the local irregularities in the actual experimental nozzle contours. The best agreement is reached in the nozzle "corners" where the experimental flow expansion in the exit cone is more uniform. For the theoretical case, the procedure forces nozzle flow expansion in the exit region to be nearly uniform. In the actual nozzle, strong changes in curvature existed at the transition between throat component and the exit section as shown in Figure 25. The Mach number distributions for the measured data show a distinct "dip" in each of the curves in this region. This "dip" is associated with a compression shock wave of a varying strength depending on location. In the worst example on the inboard side, the indicated Mach number change is of the approximate magnitude of 0.5. Actually, downstream of this location, the Mach number magnitudes are somewhat false, for the presence of the flow disturbance effects a loss in both local total and static pressure. Computed experimental Mach numbers as presented are based on upstream total pressure and, as a consequence, are pessimistically low. From conical shock theory, the total pressure loss across a shock wave of the maximum strength indicated is approximately two percent, whereas the static pressure rise is 70 percent. Because the shock wave pattern at the nozzle exit plane (Figure 26) indicates that approximately 50 percent of the nozzle wall area flow is affected by this loss, its overall impact on nozzle performance can be substantial.

It has been shown that the multiple nozzle array is subject to the effect of individual nozzle juncture shocks when the array is shrouded. Quantitatively, it is estimated that approximately 50 percent additional advantage expected from an increased expansion ratio was gained because of the existence of these shocks. Concept of the shock strength is possible by means of the monitoring pressure data measured at the shroud surface. The pressure orifices were randomly located so that some were in line with individual nozzle juncture planes and others were at in-between locations. The pressure orifice system and measured magnitudes of pressure ratio (to upstream total pressure) are shown in Figure 31. For clarity, the shroud is shown unfolded about a line in the exit plane of the individual nozzles and the shock wave patterns as interpreted from the shadowgraph pictures are shown for each perirent nozzle juncture. Three orifices locations not shown on Figure 31 are actually upstream of the individual nozzle exit stations in the dead-air region between nozzles and shroud; as a consequence, the pressure data

for these orifices represent the general magnitude of pressure at the exit junctures. Pressures greater than these three reference pressures must of necessity be shock-generated for a full-flowing nozzle system.

The diagram of Figure 31 shows that orifice 8 is located to sense the full effect of a juncture, or interaction, shock, and that orifices 7 and 13 are located in a unique position in a shroud corner and are theoretically displaced only slightly from the effective position of orifice 8. The data show that the pressure coefficients for orifices 8, 13, 5, and 7, in descending order of magnitude, are all greater than the coefficients of the three pressure references in the dead-air region. Moreover, the pressure coefficient for orifice 8 is approximately five times the coefficient of the comparable positioned orifice 6. An average pressure increase of this magnitude in as little as 12.5 percent of the flow field could result in the incremental performance gain being 50 percent greater than that predicted analytically.

D. MATERIALS EVALUATION

In order to take full advantage of the multiple nozzle concept, as applied to the nuclear rocket engine, refractory materials are necessary to reduce the heat transfer which reduces the pump pressure requirements. Refractory materials, e.g., tungsten, tantalum, molybdenum, niobium, and graphite can be fabricated in nozzle configurations of limited dimensions. For large flow areas the use of many smaller refractory nozzles is feasible under present state-of-the-art technology.

An investigation of the state-of-the-art technology of refractory materials and fabrication methods suitable to the multiple nozzle concept was conducted and the results of these studies along with recommendations of material, fabrication methods and costs are presented herein.

The 4000°R operating temperature at the inside surface of the nozzle throat is sufficiently low to enable consideration of most refractory alloys. However, since the throat region is cooled by liquid hydrogen (65°R at steady state), the material should possess sufficient ductility and notch toughness over a wide

range of cryogenic temperatures. In general, tungsten, molybdenum and niobium become brittle while tantalum retains ductility. The requirement for ductility at room and cryogenic temperature suggests a tantalum alloy. Fabrication of tantalum alloys by machining, threading, forming and welding is readily accomplished.

The material selection for the multiple nozzle application that satisfied ductility requirements at room and cryogenic temperatures along with high temperature exposure is an alloy containing 90% tantalum and 10% tungsten. This alloy has been completely evaluated and is obtainable as a "shelf item".

Although tantalum is one of the prime candidates, it will chemically react with hydrogen in the temperature range of 800°F - 1300°F, forming a brittle tantalum hydride. Thus, the dwell time in this temperature range is an important consideration.

Several methods of fabrication have been considered to obtain adequate mechanical properties at minimum cost. Among these are: plasmaspray, powder compaction, spinning, die forming, welding, machining, electroforming, vapor deposition, and casting. A flow diagram is shown in Figure 32 giving the details of each fabrication approach. A survey of fabrication techniques and suppliers' costs for a single part or several parts is shown in Table VI. The two most economical approaches are;

1. Spin, die-form weld
2. Plasmaspray, sinter

Either method would be applicable to multi-nozzle production at reduced costs. The first-choice fabrication approach is spinning the entrance section, die-forming and welding the exit section. Threads can be readily machined after fabrication. This method makes use of wrought sheet materials; the highest strength form of the alloy. Fabrication and welding techniques are also well established for this material.

The plasmaspray approach provides the lowest cost product, but to date, there are insufficient data on mechanical properties.

1V. COMPARATIVE EVALUATION

Two important considerations (engine weight and specific impulse) that affect engine performance are presented here for comparative evaluation. The first, engine weight, can be examined in terms of component weight and interstage weight. The application of the multiple nozzle concept to the NERVA engine results in weight reductions in both categories. Figure 33 illustrates the multiple nozzle attached to the NERVA engine pressure vessel aft interface flange. The hot bleed port has been included to show feasibility of this application. Axial distance requirements utilizing a 2:1 elliptical nozzle support plate were determined when applying an optimum thrust angle of 45° between the nozzle and the pressure vessel aft flange interfaces. The multiple nozzle will weigh approximately 65% of the comparable "U" tube steel jacketed NERVA nozzle, or 430 pounds less.

The most attractive weight saving occurs when interstage rocket weight is examined. This weight is a function of the stage axial length. Figure 34 shows a comparison between the NERVA engine axial length utilizing the converging - diverging nozzle and the multiple nozzle concept. The multiple nozzle engine is 37.5 inches shorter, and results in an inter-stage weight reduction of approximately 1500 pounds based on a lunar type mission, a 33 ft dia propellant tank, and a stage weight estimated to be 40 lb/in. Therefore an engine utilizing the multiple nozzle concept can reduce engine and interstage weight by 1930 pounds which can be converted directly to increased payload.

The second engine performance consideration is specific impulse (I_{sp}) or the measure of thrust per unit of working fluid weight flow rate. The multiple nozzle specific impulse was determined from thrust coefficients (C_F) based on experimental test data. Since optimum nozzle pressure ratios were used in the test program, the resulting calculated thrust coefficients are also optimum (C_{FOPT}). A comparison of conventional converging - diverging nozzle optimum thrust coefficients and resulting performance data with that of the multiple nozzle performance data is shown in Table VII. It can be seen that the multiple nozzle I_{sp} for a 13.65 area ratio is 5 seconds less than an equivalent conventional converging - diverging nozzle.

However, when considering an engine application the difference is reduced to 4.2 seconds because of the lower coolant pressure drop in the multiple nozzle concept. The pressure drop for cooling the multiple nozzle assembly is expected to be considerably lower than an equivalent conventional converging - diverging nozzle. This reduction in pressure drop, approximately 125 psia, will reduce pump requirements considerably. Performance data presented in Table VII are based on pump discharge pressures of 915 psia for the conventional nozzle engine application and 790 psia for the multiple nozzle assembly engine. The turbine drive gas requirements are subsequently reduced resulting in a nominal increase in overall engine specific impulse.

Another factor to consider when evaluating the multiple nozzle concept is the size of conventional converging - diverging nozzles as the physical size of nuclear rockets increase. Techniques presently employed in the construction of tube-and-shell type nozzles may be affected by the capability and capacity of existing fabrication facilities. The use of many small nozzles fabricated of refractory materials is feasible under present state-of-the-art technology. For larger reactor core diameters the number of small nozzles could be increased or the nozzle size modified without affecting fabrication capabilities.

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3. Computing Services Report No. 15, "Numerical Solution of Supersonic Flow by the Method of Characteristics," by R.D. Glauz, 18 January 1961.
4. Sauer, R., "General Characteristics of the Flow Through Nozzles at Near Critical Speeds", NACA TM 1147, June 1946.

TABLE I
 AXISYMMETRIC NOZZLE WALL COORDINATES
 NOZZLE CONFIGURATION NO. 5

$$\theta_i = 19.000^\circ \quad R_d = 0.760 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.247	0.591	TANGENT AND INFLECTION POINT
0.765	0.785	
1.302	0.973	
1.851	1.138	
2.373	1.274	
2.947	1.400	
3.463	1.496	
4.011	1.584	
4.598	1.662	
5.090	1.717	
5.602	1.765	EXIT
6.272	1.816	
6.825	1.850	

TABLE II
 ASSYMMETRIC NOZZLE WALL COORDINATES
 NOZZLE CONFIGURATION NO. 1

$$\emptyset = 0^{\circ}$$

$$\theta_i = 11.841^{\circ} \quad R_d = 9.952 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
2.042	0.762	TANGENT POINT
2.998	0.962	INFLECTION POINT
3.336	1.046	
3.658	1.105	
3.821	1.132	
4.362	1.214	
4.751	1.265	
4.976	1.291	
5.400	1.335	
5.656	1.358	
6.086	1.392	
6.410	1.414	
6.825	1.416	EXIT

TABLE II (CON'T.)
ASYMMETRIC NOZZLE WALL COORDINATES
NOZZLE CONFIGURATION NO. 1

$$\emptyset = 12.2^{\circ}, 98.8^{\circ}$$

$$\theta_i = 12.212^{\circ} \quad R_d = 8.881 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.879	0.751	TANGENT POINT
2.918	0.976	INFLECTION POINT
3.246	1.044	
3.377	1.069	
3.798	1.144	
3.866	1.155	
4.387	1.234	
4.412	1.238	
5.031	1.314	
5.676	1.378	
5.717	1.382	
6.477	1.437	
6.825	1.455	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 1

$$\emptyset = 24.7^{\circ}$$

$$\theta_i = 13.069^{\circ}$$

$$R_d = 6.794 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.536	0.726	TANGENT POINT
2.735	1.004	INFLECTION POINT
3.110	1.087	
3.568	1.178	
3.847	1.227	
4.083	1.266	
4.432	1.319	
4.652	1.349	
5.054	1.399	
5.297	1.426	
5.712	1.468	
6.010	1.494	
6.825	1.550	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 1

$$\emptyset = 35.4^{\circ}, 146.2^{\circ}$$

$$\theta_i = 14.485^{\circ} \quad R_d = 4.232 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.059	0.685	TANGENT POINT
2.416	1.035	INFLECTION POINT
2.559	1.072	
2.882	1.150	
3.221	1.225	
3.404	1.263	
3.898	1.356	
4.442	1.446	
5.048	1.530	
5.571	1.591	
5.735	1.608	
6.495	1.676	
6.825	1.699	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 1

$$\emptyset = 44.2^{\circ}$$

$$\theta_i = 16.372^{\circ}$$

$$R_d = 1.952 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.550	0.629	TANGENT POINT
1.938	1.037	INFLECTION POINT
2.353	1.154	
2.822	1.275	
3.068	1.333	
3.312	1.388	
3.787	1.485	
4.321	1.582	
4.908	1.673	
5.328	1.730	
5.972	1.806	
6.304	1.839	
6.825	1.883	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 1

$$\emptyset = 52.1^{\circ}$$

$$\theta_i = 18.920^{\circ}$$

$$R_d = 0.080 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.026	0.554	TANGENT POINT
0.975	0.880	INFLECTION POINT
1.557	1.074	
2.100	1.241	
2.623	1.386	
3.102	1.507	
3.611	1.622	
4.149	1.730	
4.718	1.831	
5.317	1.922	
5.734	1.978	
6.547	2.023	
6.825	2.100	EXIT

TABLE II (CON'T.)

NOZZLE CONFIGURATION NO. 1

$$\emptyset = 61.2^{\circ}$$

$$\theta_i = 15.812^{\circ}$$

$$R_d = 2.527 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.689	0.646	TANGENT POINT
2.089	1.042	INFLECTION POINT
2.418	1.132	
2.771	1.222	
3.186	1.318	
3.445	1.374	
3.988	1.479	
4.564	1.576	
5.173	1.662	
5.411	1.692	
6.136	1.771	
6.603	1.813	
6.825	1.831	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 1

$$\emptyset = 72.1^{\circ}, 124.3^{\circ}$$

$$\theta_i = 13.714^{\circ}$$

$$R_d = 5.510 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.306	0.707	TANGENT POINT
2.593	1.021	INFLECTION POINT
2.820	1.076	
3.244	1.168	
3.677	1.252	
3.957	1.302	
4.247	1.349	
4.545	1.393	
4.833	1.432	
5.495	1.510	
6.170	1.572	
6.232	1.577	
6.825	1.619	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 1

$$\theta = 84.6^{\circ}, 112.1^{\circ}, 156.3^{\circ}$$

$$\theta_i = 12.551^{\circ}$$

$$R_d = 7.995 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.737	0.741	TANGENT POINT
2.848	0.988	INFLECTION POINT
3.335	1.091	
3.702	1.158	
3.954	1.201	
4.281	1.251	
4.510	1.283	
4.898	1.333	
5.140	1.360	
5.551	1.402	
5.837	1.428	
6.601	1.482	
6.825	1.494	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 1

$$\emptyset = 132.7^{\circ}$$

$$\theta_i = 15.089^{\circ}$$

$$R_d = 3.387 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.882	0.667	TANGENT POINT
2.271	1.041	INFLECTION POINT
2.596	1.126	
2.920	1.204	
3.345	1.298	
3.886	1.403	
4.461	1.499	
5.070	1.586	
5.213	1.604	
5.711	1.661	
5.918	1.683	
6.385	1.726	
6.825	1.759	EXIT

TABLE II (CON'T.)

NOZZLE CONFIGURATION NO. 1

$$\emptyset = 139.5^{\circ}$$

$$\theta_i = 16.794^{\circ}$$

$$R_d = 1.566 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.452	0.617	TANGENT POINT
1.811	1.027	INFLECTION POINT
2.214	1.144	
2.566	1.239	
2.958	1.337	
3.387	1.434	
3.761	1.511	
4.318	1.614	
4.907	1.707	
5.527	1.791	
6.291	1.875	
6.427	1.889	
6.825	1.922	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 1

$$\emptyset = 168.8^{01}$$

$$\theta_i = 11.650^0$$

$$R_d = 10.550 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
2.130	0.767	TANGENT POINT
3.040	0.955	INFLECTION POINT
3.209	0.990	
3.252	0.998	
3.572	1.057	
3.728	1.084	
4.250	1.165	
4.555	1.206	
4.863	1.243	
5.196	1.278	
5.530	1.310	
6.271	1.365	
6.825	1.395	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 1

$$\emptyset = 180^{\circ}$$

$$\theta_i = 11.278^{\circ}$$

$$R_d = 11.816 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
2.311	0.778	TANGENT POINT
3.114	0.938	INFLECTION POINT
3.491	1.010	
3.632	1.034	
4.066	1.103	
4.154	1.116	
4.678	1.184	
4.746	1.192	
5.330	1.253	
6.019	1.308	
6.128	1.315	
6.745	1.349	
6.825	1.353	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 2

$$\theta = 0^\circ$$

$$\theta_i = 10.629^\circ$$

$$R_d = 13.577 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
2.504	0.783	TANGENT POINT
3.379	0.947	INFLECTION POINT
3.685	1.004	
4.189	1.083	
4.257	1.093	
4.744	1.156	
4.894	1.173	
5.333	1.219	
5.606	1.243	
6.000	1.271	
6.397	1.299	
6.600	1.309	
6.825	1.321	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 2

$$\emptyset = 14.0^{\circ}, 171.5^{\circ}$$

$$\theta_i = 11.020^{\circ}$$

$$R_d = 11.992 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
2.292	0.771	TANGENT POINT
3.293	0.966	INFLECTION POINT
3.550	1.014	
3.783	1.055	
4.343	1.141	
4.365	1.144	
4.907	1.213	
5.015	1.225	
5.503	1.274	
5.740	1.295	
6.455	1.345	
6.546	1.350	
6.825	1.364	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 2

$$\emptyset = 27.7^{\circ}, 99.4^{\circ}$$

$$\theta_i = 12.062^{\circ}$$

$$R_d = 8.578 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.792	0.739	TANGENT POINT
3.068	1.012	INFLECTION POINT
3.269	1.054	
3.511	1.101	
4.027	1.192	
4.565	1.272	
4.670	1.286	
5.136	1.344	
5.353	1.367	
6.047	1.431	
6.117	1.437	
6.690	1.476	
6.825	1.484	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 2

$\emptyset = 37.2^{\circ}, 73.5^{\circ}, 125.2^{\circ}, 148.1^{\circ}$

$\theta_i = 13.320$

$R_d = 5.613 \text{ IN.}$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.293	0.701	TANGENT POINT
2.786	1.054	INFLECTION POINT
3.194	1.146	
3.290	1.166	
3.678	1.243	
3.809	1.268	
4.190	1.333	
4.388	1.364	
5.039	1.455	
5.597	1.520	
5.764	1.537	
6.572	1.606	
6.825	1.623	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 2

$$\emptyset = 46.5^{\circ}, 62.5^{\circ}$$

$$\theta_i = 15.283^{\circ}$$

$$R_d = 2.573 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.678	0.641	TANGENT POINT
2.311	1.087	INFLECTION POINT
2.752	1.202	
3.201	1.308	
3.676	1.409	
4.178	1.503	
4.274	1.520	
4.705	1.589	
4.905	1.619	
5.615	1.712	
6.143	1.770	
6.402	1.795	
6.825	1.830	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 2

$$\emptyset = 54.7^{\circ}$$

$$\theta_i = 18.000^{\circ}$$

$$R_d = 0.159 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.049	0.558	TANGENT POINT
1.093	0.897	INFLECTION POINT
1.515	1.035	
2.062	1.204	
2.459	1.318	
3.101	1.484	
3.560	1.590	
4.043	1.690	
4.552	1.784	
5.085	1.871	
5.642	1.951	
6.225	2.022	
6.825	2.086	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 2

$$\theta = 86.0^{\circ}, 112.7^{\circ}$$

$$\theta_i = 12.229^{\circ}$$

$$R_d = 8.121 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.720	0.734	TANGENT POINT
3.031	1.018	INFLECTION POINT
3.472	1.108	
3.550	1.123	
3.978	1.199	
4.100	1.219	
4.514	1.281	
4.719	1.309	
5.376	1.386	
5.407	1.390	
5.984	1.444	
6.176	1.459	
6.825	1.502	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 2

$$\emptyset = 132.5^{\circ}$$

$$\theta_i = 14.608^{\circ}$$

$$R_d = 3.454 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.871	0.662	TANGENT POINT
2.482	1.081	INFLECTION POINT
2.647	1.124	
2.896	1.185	
3.356	1.288	
3.565	1.331	
4.097	1.431	
4.627	1.518	
4.728	1.533	
5.181	1.595	
5.418	1.625	
6.184	1.707	
6.825	1.760	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 2

$$\emptyset = 138.7^{\circ}$$

$$\theta_i = 16.074^{\circ}$$

$$R_d = 1.708 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.473	0.617	TANGENT POINT
2.093	1.084	INFLECTION POINT
2.387	1.166	
2.810	1.276	
3.257	1.382	
3.732	1.483	
4.235	1.579	
4.457	1.617	
5.109	1.718	
5.599	1.783	
5.841	1.812	
6.498	1.881	
6.825	1.911	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 2

$$\theta = 157.3^\circ$$

$$\theta_i = 11.891^\circ$$

$$R_d = 9.067 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.868	0.745	TANGENT POINT
3.105	1.005	INFLECTION POINT
3.313	1.048	
3.472	1.079	
3.815	1.141	
4.013	1.174	
4.619	1.263	
5.193	1.333	
5.299	1.345	
5.794	1.392	
6.056	1.414	
6.758	1.462	
6.825	1.465	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 2

$$\emptyset = 180^{\circ}$$

$$\theta_i = 10.827^{\circ}$$

$$R_d = 12.751$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
2.395	0.777	TANGENT POINT
3.336	0.957	INFLECTION POINT
3.603	1.007	
3.734	1.029	
4.129	1.092	
4.312	1.119	
4.680	1.167	
4.955	1.199	
5.572	1.261	
5.674	1.269	
6.201	1.309	
6.472	1.325	
6.825	1.342	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 3

$$\emptyset = 0^\circ$$

$$\theta_i = 11.336^\circ$$

$$R_d = 12.312 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
2.420	0.790	TANGENT POINT
3.446	0.996	INFLECTION POINT
3.625	1.032	
3.721	1.050	
4.283	1.145	
4.287	1.146	
4.885	1.231	
4.921	1.236	
5.523	1.305	
5.629	1.316	
6.197	1.366	
6.421	1.383	
6.825	1.407	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 3

$$\emptyset = 16.8^{\circ}$$

$$\theta_i = 11.947^{\circ}$$

$$R_d = 10.302 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
2.133	0.773	TANGENT POINT
3.302	1.020	INFLECTION POINT
3.817	1.122	
3.877	1.133	
4.387	1.218	
4.460	1.229	
4.992	1.304	
5.112	1.319	
5.631	1.378	
5.840	1.398	
6.300	1.439	
6.655	1.465	
6.825	1.475	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 3

$$\emptyset = 37.2^{\circ}, 75.3^{\circ}, 133.6^{\circ}$$

$$\theta_i = 14.068^{\circ}$$

$$R_d = 5.458 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.327	0.714	TANGENT POINT
2.807	1.085	INFLECTION POINT
3.252	1.191	
3.343	1.211	
3.776	1.301	
4.331	1.403	
4.431	1.420	
4.918	1.496	
5.074	1.518	
5.538	1.578	
5.793	1.608	
6.532	1.682	
6.825	1.705	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 3

$$\emptyset = 47.6^{\circ}, 65.4^{\circ}, 143.0^{\circ}$$

$$\theta_i = 16.250^{\circ}$$

$$R_d = 2.560 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.717	0.652	TANGENT POINT
2.254	1.100	INFLECTION POINT
2.463	1.161	
2.865	1.269	
3.307	1.377	
3.804	1.487	
4.302	1.584	
4.874	1.683	
4.996	1.702	
5.476	1.772	
5.700	1.802	
6.487	1.893	
6.825	1.925	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 3

$$\emptyset = 57.4^{\circ}$$

$$\theta_i = 19.450^{\circ}$$

$$R_d = 0.210 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.070	0.562	TANGENT POINT
1.093	0.923	INFLECTION POINT
1.682	1.125	
2.083	1.253	
2.662	1.423	
3.128	1.548	
3.622	1.668	
4.143	1.781	
4.692	1.888	
5.268	1.987	
5.870	2.079	
6.500	2.162	
6.825	2.200	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 3

$$\emptyset = 87.7^{\circ}, 122.0^{\circ}$$

$$\theta_i = 12.782^{\circ}$$

$$R_d = 8.062 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.784	0.750	TANGENT POINT
3.110	1.051	INFLECTION POINT
3.406	1.116	
3.551	1.146	
3.853	1.204	
4.096	1.248	
4.420	1.302	
4.702	1.345	
5.336	1.429	
5.988	1.499	
6.138	1.513	
6.673	1.558	
6.825	1.568	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 3

$$\emptyset = 105.2^{\circ}$$

$$\theta_i = 12.221^{\circ}$$

$$R_d = 9.510 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
2.013	0.765	TANGENT POINT
3.242	1.032	INFLECTION POINT
3.421	1.071	
3.734	1.133	
3.952	1.172	
4.298	1.231	
4.543	1.269	
4.895	1.319	
5.204	1.358	
5.859	1.430	
5.942	1.438	
6.542	1.488	
6.825	1.508	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 3

$$\emptyset = 153.0^{\circ}$$

$$\theta_i = 13.051^{\circ}$$

$$R_d = 7.441 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.680	0.742	TANGENT POINT
3.047	1.059	INFLECTION POINT
3.127	1.078	
3.511	1.160	
3.614	1.181	
4.055	1.265	
4.165	1.284	
4.633	1.359	
4.780	1.381	
5.245	1.444	
5.467	1.471	
6.233	1.550	
6.825	1.597	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 3

$$\theta = 164.3^\circ$$

$$\theta_i = 11.643^\circ$$

$$R_d = 11.259 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
2.272	0.782	TANGENT POINT
3.374	1.009	INFLECTION POINT
3.719	1.077	
3.800	1.092	
4.187	1.159	
4.375	1.189	
4.782	1.247	
5.018	1.278	
5.412	1.324	
5.736	1.358	
6.424	1.416	
6.540	1.425	
6.825	1.442	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 3

$$\emptyset = 180^{\circ}$$

$$\theta_i = 11.035^{\circ}$$

$$R_d = 13.449 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
2.574	0.799	TANGENT POINT
3.512	0.982	INFLECTION POINT
3.533	0.986	
3.640	1.006	
4.091	1.085	
4.198	1.103	
4.683	1.174	
4.822	1.192	
5.311	1.250	
5.519	1.272	
5.975	1.314	
6.299	1.340	
6.825	1.372	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\theta = 0^\circ$$

$$\theta_i = 9.139^\circ$$

$$R_d = 24.009 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
3.813	0.855	TANGENT POINT
4.288	0.931	INFLECTION POINT
4.295	0.932	
4.450	0.955	
4.600	0.977	
4.759	1.001	
4.953	1.027	
5.383	1.077	
5.690	1.109	
6.000	1.134	
6.395	1.167	
6.511	1.175	
6.825	1.193	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\emptyset = 22.8^{\circ}$$

$$\theta_i = 9.932^{\circ}$$

$$R_d = 18.887 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
3.258	0.833	TANGENT POINT
4.086	0.978	INFLECTION POINT
4.464	1.042	
4.560	1.056	
4.750	1.082	
5.069	1.127	
5.244	1.148	
5.500	1.177	
5.711	1.200	
6.010	1.228	
6.370	1.255	
6.744	1.284	
6.825	1.289	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\theta = 40.3^{\circ}, 167.3^{\circ}$$

$$\theta_i = 11.859^{\circ}$$

$$R_d = 13.084 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
2.538	0.799	TANGENT POINT
3.789	1.046	INFLECTION POINT
3.830	1.085	
4.220	1.158	
4.417	1.191	
4.806	1.252	
5.076	1.290	
5.350	1.324	
5.749	1.373	
5.814	1.380	
6.419	1.438	
6.640	1.456	
6.825	1.468	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\emptyset = 52.7^{\circ}, 132.8^{\circ}$$

$$\theta_i = 13.284^{\circ}$$

$$R_d = 7.125 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.637	0.741	TANGENT POINT
3.245	1.120	INFLECTION POINT
3.689	1.220	
3.736	1.230	
4.236	1.328	
4.309	1.341	
4.812	1.426	
4.947	1.447	
5.420	1.514	
5.663	1.546	
6.396	1.627	
6.463	1.634	
6.825	1.665	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\theta = 63.6^\circ$$

$$\theta_1 = 15.907^\circ$$

$$R_d = 3.077 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.843	0.668	TANGENT POINT
2.575	1.161	INFLECTION POINT
2.853	1.239	
3.300	1.354	
3.801	1.470	
4.321	1.577	
4.887	1.680	
5.017	1.702	
5.483	1.774	
5.730	1.810	
6.436	1.899	
6.530	1.909	
6.825	1.940	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\emptyset = 74.6^{\circ}$$

$$\theta_i = 20.001^{\circ}$$

$$R_d = 0.064 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.022	0.554	TANGENT POINT
1.142	0.962	INFLECTION POINT
1.428	1.066	
1.937	1.240	
2.498	1.417	
2.947	1.547	
3.671	1.735	
4.185	1.854	
4.725	1.966	
5.290	2.072	
5.888	2.168	
6.444	2.175	
6.825	2.300	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\theta = 83.6^\circ$$

$$\theta_i = 16.831^\circ$$

$$R_d = 2.150 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.623	0.642	TANGENT POINT
2.319	1.155	INFLECTION POINT
2.592	1.236	
2.833	1.304	
3.015	1.353	
3.312	1.429	
3.479	1.469	
3.819	1.548	
4.354	1.660	
4.593	1.706	
5.211	1.814	
5.996	1.930	
6.825	2.030	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\emptyset = 94.0^{\circ}$$

$$\theta_i = 14.715^{\circ}$$

$$R_d = 4.603 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.169	0.701	TANGENT POINT
2.885	1.152	INFLECTION POINT
3.063	1.197	
3.313	1.259	
3.539	1.311	
3.829	1.375	
4.085	1.427	
4.661	1.534	
5.253	1.629	
5.374	1.647	
5.875	1.714	
6.136	1.746	
6.825	1.819	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\emptyset = 104.5^{\circ}$$

$$\theta_i = 13.556^{\circ}$$

$$R_d = 6.572 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.541	0.733	TANGENT POINT
3.177	1.128	
3.283	1.153	
3.614	1.228	
3.803	1.268	
4.154	1.337	
4.383	1.379	
5.028	1.485	
5.642	1.570	
5.753	1.584	
6.291	1.645	
6.563	1.672	
6.825	1.694	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\emptyset = 119.7^{\circ}$$

$$\theta_i = 13.001^{\circ}$$

$$R_d = 7.742 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.742	0.748	TANGENT POINT
3.316	1.112	INFLECTION POINT
3.669	1.191	
4.039	1.265	
4.234	1.302	
4.607	1.366	
4.864	1.408	
5.200	1.455	
5.571	1.506	
6.166	1.575	
6.362	1.594	
6.600	1.615	
6.825	1.634	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\emptyset = 142.0^{\circ}$$

$$\theta_i = 14.077^{\circ}$$

$$R_d = 5.616 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.366	0.719	TANGENT POINT
3.048	1.140	INFLECTION POINT
3.222	1.184	
3.400	1.226	
3.375	1.300	
3.931	1.341	
4.279	1.408	
4.525	1.453	
5.154	1.555	
5.775	1.639	
5.927	1.658	
6.428	1.714	
6.825	1.752	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\emptyset = 150.9^{\circ}$$

$$\theta_i = 15.678^{\circ}$$

$$R_d = 3.338 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
0.902	0.674	TANGENT POINT
2.635	1.161	INFLECTION POINT
2.811	1.210	
3.083	1.281	
3.254	1.324	
3.582	1.402	
3.750	1.439	
4.319	1.557	
4.955	1.671	
5.557	1.764	
6.187	1.847	
6.454	1.878	
6.825	1.916	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\varnothing = 158.7^{\circ}$$

$$\theta_i = 12.709^{\circ}$$

$$R_d = 8.431 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
1.855	0.757	TANGENT POINT
3.390	1.103	INFLECTION POINT
3.599	1.150	
4.125	1.255	
4.699	1.354	
4.779	1.367	
5.000	1.390	
5.304	1.443	
5.477	1.466	
5.943	1.521	
6.257	1.554	
6.500	1.575	
6.825	1.604	EXIT

TABLE II (CON'T.)
NOZZLE CONFIGURATION NO. 4

$$\emptyset = 180^{\circ}$$

$$\theta_i = 10.668^{\circ}$$

$$R_d = 15.209 \text{ IN.}$$

<u>L (INCHES)</u>	<u>R (INCHES)</u>	
2.815	0.813	TANGENT POINT
3.897	1.017	INFLECTION POINT
4.023	1.041	
4.166	1.066	
4.499	1.121	
4.800	1.166	
5.100	1.205	
5.424	1.247	
5.509	1.257	
6.081	1.315	
6.303	1.335	
6.500	1.356	
6.825	1.392	EXIT

TABLE III

NOZZLE OPERATING CONDITIONS AND DESIGN PARAMETERS

THROAT RADIUS	$R_t = 0.55 \text{ IN.}$
SPECIFIC HEAT RATIO	1.40
NOZZLE LENGTH	$L = 6.825 \text{ IN.}$
UPSTREAM EXTERNAL RADIUS	$R_u = 0.76 \text{ IN.}$
CHAMBER PRESSURE	$P_c = 550 \text{ PSIA}$
CHAMBER TEMPERATURE	$T = 4090^\circ\text{R}$
GAS CONSTANT	$R = 772.5 \text{ FT} / ^\circ\text{R}$
AMBIENT PRESSURE	$P_a = 0$

TABLE IV
EXPERIMENTAL NOZZLE DIMENSIONAL DATA

<u>NOZZLE</u>	<u>THROAT AREA, IN.²</u>	<u>EXPANSION RATIO</u>
A	.9808	10.363
B	.9852	10.317
C	.9825	10.345
D	.9721	10.456
E	.9755	10.281
F	1.0029	10.000
G	.9931	10.098
H	.9931	8.349
I	.9733	8.519
J	.9896	10.081
10 - NOZZLE WITHOUT SHROUD	9.848	9.89
10 - NOZZLE WITH SHROUD	9.848	13.65

TABLE V

THEORETICAL AND EXPERIMENTAL PERFORMANCE DATA
FOR
THE EXPERIMENTAL NOZZLE SYSTEM

1. EXPERIMENTAL

CONFIGURATION	η_T %	$C_{F_{vac}}$
SINGLE NOZZLE	97.1	1.610
10 - NOZZLE WITH SHROUD	95.5	1.595
10 - NOZZLE WITHOUT SHROUD	94.9	1.565

2. THEORETICAL

CONFIGURATION	η_T %	$C_{F_{vac}}$
SINGLE NOZZLE	96.4	1.592
10 - NOZZLE WITH SHROUD	94.3	1.578
10 - NOZZLE WITHOUT SHROUD	94.4	1.557

TABLE VI

FABRICATION TECHNIQUE COMPARISON

TECHNIQUE	DENSITY	PROCESSING DEVELOPED		SUPPLIERS	COST FOR ONE PART		PRODUCTION COST FOR SEVERAL PARTS	
		W	Ta - 10W		TUNGSTEN	Ta - 10 W	TUNGSTEN	Ta - 10 W
PLASMASPRAY	90%	OK	SOME REQD.	WESTERN GEAR	\$ 1500	\$1700	\$800	\$900
COMPACTION	95%			AGC - DOWNEY	10108 ①			
SPIN + DIE FORM	100%	OK	OK	SUPERTEMP	6400	8300	630	950
		SOME REQD	OK	AGC - SRP		3055 (INCL 1410 TOOLING)	(PRODUCTION OF 100 PARTS)	1100
MACHINE	100%	OK	OK	AGC - SRP		4500		1050 (BASED ON 100 PARTS)
ELECTROFORM	100%			UNION CARBIDE	PRICE NOT AVAILABLE BUT ESTIMATE \$3000 BASED ON PROCESS			

① TEN PARTS REQD. TO GUARANTEE SATISFACTORY PART

TABLE VII
MULTIPLE NOZZLE AND CONVENTIONAL NOZZLE PERFORMANCE

<u>AREA RATIO</u>	<u>C_F OPTIMUM CONVENTIONAL NOZZLE</u>	<u>C_F OPTIMUM EXPERIMENTAL DATA MULTIPLE NOZZLES</u>	<u>I_{sp} NOZZLE</u>	<u>I_{sp} ENGINE</u>
13.65	1.6048		750.2	736.0
13.65		1.594	745.2	731.8
9.88	1.5727		735.2	721.8
9.88		1.567	732.5	719.7

Figure 1
Multiple Nozzle Configuration

**MULTIPLE NOZZLE CONFIGURATION (a), USING
INDIVIDUAL NOZZLES ON EACH FUEL ELEMENT**

Figure 2
Multiple Nozzle, Configuration A

MULTIPLE NOZZLE CONFIGURATION (b), USING
INDIVIDUAL NOZZLES ON EACH GAS PASSAGE

Figure 3

Multiple Nozzle, Configuration B

**MULTIPLE NOZZLE CONFIGURATION (c) USING
PLENUM CHAMBER**

Figure 4
Multiple Nozzle, Configuration C

Figure 5
Axially Aligned Nozzles, Standard Entrance Sections

Figure 6
Axially Aligned Nozzles, Standard Entrance Sections

Figure 7
Axially Aligned Nozzles, Modified Entrance Sections

Figure 8
Thermal and Fluid Flow Design, Refractory Nozzle

Figure 9
Alternate Assembly Method

Figure 10
Superposition of Asymmetric Nozzle

NOZZLE NOMENCLATURE

SCHEMATIC OF ASYMMETRIC NOZZLE CONTOUR ALONG TWO RADIAL PLANES

Figure 11
Analytical Nozzle Geometry

CHANNEL 7

CHANNEL 8

Figure 12
Static Test Stands

Figure 14
Horizontal Calibration Force Diagram

$$V_3 > V_1 > (V.L.)$$

$$d_1 = 17.978$$

$$\sum F_V = 0 = V_3 - V_1 - (V.L.)$$

$$d_2 = 7.464$$

$$\sum M = 0 = (V.L.)d_3 + (H.L.)d_2 - V_1d_1$$

$$d_3 = 4.130$$

$$S = 5.770$$

Figure 15
Force Diagram for Vertical Calibration

Figure 16
Multiple Nozzle Schematic

Figure 17
Photographs of Single Nozzle

Figure 18
Single Nozzle Test Setup

Figure 19
Photographs of 3 Nozzle Configuration

Figure 20
Shadowgraph Setup

Figure 21
Photographs of Ten-Nozzle Configuration With Shroud

Figure 22
Facility Setup for Ten Nozzle Tests

- SINGLE NOZZLE "E"
- △ 10 - NOZZLE WITH SHROUD
- 10 - NOZZLE WITHOUT SHROUD
- □ △ REPRESENT THEORETICAL DATA

Figure 23
Thrust Efficiency vs Pressure Ratio

Figure 24
Vacuum Thrust Coefficient vs Pressure Ratio

Figure 25
Shock Wave Schematic Diagram

Figure 26
Single Nozzle Exhaust Flow Shadowgraph

Figure 27
Exhaust Flow Shadowgraph ($\Theta = -90^\circ$)

Figure 28
Exhaust Flow Shadowgraph ($\Theta = 0^\circ$)

Figure 29
Exhaust Flow Shadowgraph ($\Theta = +10^\circ$)

Figure 30
Single Nozzle Internal Pressure Distribution

ORIFICE	PRESSURE RATIO	ORIFICE	PRESSURE RATIO
1	.00679	9	.00499
2	.00410	10	.00782
3	.00475	11	.00384
4	.00553	12	.00801
5	.0116	13	.0129
6	.00384	14	.00470
7	.00906	15	.00777
8	.0187

Figure 31
Shroud Pressure Distribution

APPROACHES:

Figure 32
Fabrication Technique

Figure 33
Multiple Nozzle NERVA Engine Application

C-D NOZZLE AND MULTIPLE NOZZLE ENGINE

(10:1 AREA RATIO)

Figure 34
Pictorial Nozzle Comparison