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PROPULSION DIVISION

STABILITY CHARACTERIZATION OF ADVANCED INJECTORS

Contract NAS 8-20672

Quarterly Progress Report to

GEORGE C. MARSHALL SPACE FLIGHT CENTER
National Aeronautics and Space Administration

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7 October 1968

Quarterly Progress Report 20672-Q-5

STABILITY CHARACTERIZATION OF ADVANCED INJECTORS

Prepared under

Contract NAS 8-20672

for

George C. Marshall Space Flight Center
National Aeronautics and Space Administration
Huntsville, Alabama

AEROJET-GENERAL CORPORATION

Liquid Rocket Operations Sacramento, California

FOREWORD

This is the first in a series of quarterly reports submitted in partial fulfillment of contractual provisions of Phase II, Supplemental Agreement No. 7 of NAS 8-20672, and the fifth quarterly report for the basic contract.

This report includes factual data, analysis and interpretation of results, a summary of the progress of work and planned future work. The reporting period is 1 July 1968 through 1 October 1968.

Work is being conducted by the Engineering Division of Liquid Rocket Operations, Aerojet-General Corporation, Sacramento, California. This program is under the direction of Dr. C. B. McGough, Program Manager; J. M. McBride, Project Manager; W. W. Howard, Project Engineer.

Approved by:

J. M. McBride

J. M. McBride, Project Manager

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I. INTRODUCTION

The purpose of this program is to develop combustion stability design criteria for injectors using hydrogen and oxygen as propellants at high combustion chamber pressures. Analytical and experimental investigations on this program are divided into four tasks: (1) Analytical Model Development, (2) Annular Thrust Chamber Assembly Testing, (3) Transverse Excitation Chamber Assembly Testing, and (4) Formulation of Design Criteria.

II. SUMMARY

Work conducted during the period 1 July 1968 to 1 October 1968 is reported herein. Program schedule and progress of work for individual tasks are shown in Figure 1. Expenditures and budgets for the program in both man-hours and dollars are shown in Figures 2 and 3, respectively.

The model for the Task I effort has been defined as the gas tap-off cycle. Because the injection methods for this combustion cycle employ gaseous hydrogen and liquid oxygen or gaseous hydrogen and gaseous oxygen, the combustion stability problems associated with injecting propellants at a density in the order of the combustion gases have been explored.

Test hardware for the Task II effort is being prepared for testing, which is scheduled for the month of February 1969.

Design and fabrication of test hardware for Transverse Excitation Chamber testing on Task III has been completed. Modifications to the test stand to improve the start and shutdown transient pressure and mixture ratio conditions, and additional pressurization capability for increasing the testing duration of the test stand are being made. Testing is planned to start in the month of October.

The survey of literature for the formulation of design criteria on Task IV is nearly completed. An annotated bibliography of forty references is included in this report. Completion of design correlations for the coaxial injector element is planned for the next quarter.

III. PROGRESS OF WORK

A. TASK I: ANALYTICAL MODEL DEVELOPMENT

1. The following is a discussion of work in progress to develop a one-dimensional model for gas-gas rocket engine systems using H_2 and O_2 as propellants. A gas-gas rocket engine was chosen for the model because $\frac{\Delta P}{P_c}$, which is related to low frequency stability, does not depend on the operating chamber pressure for gas-gas injection. This is a distinct advantage for a throttleability engine such as the Aerospike. Figure 4 shows schematically the parts of the analytical model and the variables connecting these parts. The assumptions and limitations of the model are discussed below.

a. The following general assumptions have been made:

(1) Turbine and pump inertias are assumed to be too high to respond to pressure oscillations in the feed system. This assumption determines a lower limit of frequency at which the model will be applicable.

(2) Combustion is quasi-steady. This is mainly done for simplicity and because little is known about the dynamic response of flames.

(3) The admittance of the feed lines and gas tapoff line will be considered known. The analysis of these components is straight-forward using one-dimensional acoustics.

(4) Chamber and feed line Mach numbers are small ($M < 0.3$). This refers to the Mach number with feed lines and in the vicinity of the combustion process. The combustion equations are significantly simplified when this assumption is made. This assumption is realistic for most engines.

III, A, Task I: Analytical Model Development (cont.)

(5) Combustion will be considered to be concentrated at the injector face. At low frequencies (low compared to the chamber acoustic modes of oscillation) the location of the combustion is unimportant since the pressure amplitude is almost uniform over the entire chamber.

b. The feed system will be analyzed using one-dimensional acoustics. A computer program exists at Aerojet to calculate the admittance of a feed line once the dimensions of all lines are known as well as the density and speed of sound in the fluid. In the analysis the symbol G will be used for the feed line admittance. Normally liquids are used in the feed lines so weight flow can be used as a matching parameter. If gases are used and there are significant changes in gas properties in the lines, volume flow must be used as the matching parameter when going from one section of line to another.

c. From the feed lines the fluid is injected into the chamber. In low frequency stability analysis it is assumed there is a transport delay τ between the time of injection and the time of burning. This assumes the combustion is concentrated at one location in the chamber. Work has been done to extend this concept to the case where combustion is distributed throughout the chamber (Ref. 1). When the distributed combustion is taken into account an attenuation of the feed system oscillations occurs as well as the transport delay. The above concepts were derived for liquid injection. With gas injection there are additional affects that must be considered in the injection process. If the percent open area of the injector face is high (at least higher than for liquid injection) the gases in the zone before combustion will be mostly reactants. If they were all reactants, oscillations in injected flow would be transmitted to the combustion zone acoustically. In this case there would be no time delay in the traditional sense. The actual case is somewhere between the case just described and the

III, A, Task I: Analytical Model Development (cont.)

traditional time delay. To actually evaluate the time lags in the liquid injection model the Priem steady state combustion model is used. For the gas-gas injection process additional work is required and is planned as part of this effort.

d. The Combustion Process: The combustion process for liquid propellants is considered to be a process of gas generation; and the liquid is considered to have negligible volume. When the propellants are gaseous however, this assumption is not valid. Actually, the combustion process involves the disappearance of some species and the appearance of others with corresponding changes in molecular weight, temperature, density, etc. The dynamics of the combustion process across a combustion front are described by the continuity, momentum and energy equations which can be written as follows:

$$\dot{w}_x + \dot{w}_f = \dot{w}_p$$

$$\frac{\dot{w}_p}{g} V_p - \frac{\dot{w}_x v_x}{g} - \frac{\dot{w}_f v_f}{g} = P_x A - P_p A$$

$$\dot{w}_x h_x + 1/2 \frac{V_x^2}{g} + \dot{w}_f h_f + 1/2 \frac{V_f^2}{g} = \dot{w}_p h_p + 1/2 \frac{V_p^2}{g}$$

where the enthalpy, h, includes the chemical energy

$$h_x = h_{ox} + \int_{To}^{Tx} C_{px} \, dT$$

$$h_f = h_{of} + \int_{T_o}^{T_f} C_{pf} \, dT$$

III, A, Task I: Analytical Model Development (cont.)

$$h_p = h_{op} + \int_{T_o}^{T_p} C_{pp}\, dT$$

The symbol h_o is the heat of formation at the temperature T_o. The products are likely to be made up of both hydrogen and water since the maximum performance from hydrogen oxygen system occurs at a mixture ratio of 5 instead of at the stoichiometric mixture ratio, 8. The variables h_{op} and C_{pp} are determined by the mass fractions Y of fuel and water in the products.

$$h_{op} = Y_f\, h_{of} + Y_{H_2O}\, h_{oH_2O}$$

$$C_{pp} = Y_f\, C_{pf} + Y_{H_2O}\, C_{p\,H_2O}$$

The mass fractions Y are determined by the mixture ratio. From the overall chemical reaction

$$2H_2 + O_2 \rightarrow 2H_2O$$

the number of moles of H_2O formed is twice the number of moles of O_2 present.

$$2\,\frac{\dot{w}_x}{MW_x} = \frac{\dot{w}_{H_2O}}{MW_{H_2O}}$$

The number of moles of fuel in the products is the number of moles of fuel in the reactants minus twice the number of moles of oxidizer in the reactants.

$$\frac{w_f}{MW_f} - \frac{2\,w_x}{MW_x}$$

III, A, Task I: Analytical Model Development (cont.)

The mass fractions of fuel and water in the products can then be written:

$$Y_f = \frac{MW_f \left(\frac{\dot{w}_f}{MW_f} - \frac{2\,\dot{w}_x}{MW_x} \right)}{\dot{w}_p}$$

$$Y_{H_2O} = \frac{MW_{H_2O} \left(\frac{2\,\dot{w}_x}{MW_x} \right)}{\dot{w}_p}$$

In order to determine the relative importance of the different terms in the momentum and energy equations the equations can be divided by an appropriate non-dimensionalizing factor. When this is done it is found that if the Mach number, based on the speed of sound of a stoichiometric mixture, is small the velocity terms in the energy and momentum equations can be neglected. The simplified equations across the combustion front then become:

$$\dot{w}_f + \dot{w}_x = \dot{w}_p$$

$$P_x = P_v = P_p$$

$$\dot{w}_x h_x + \dot{w}_f h_f = \dot{w}_p h_p$$

It is required to obtain equations for the oscillations of all variables about their steady state value. To do this each variable must be written as a steady part plus an oscillatory part. When these variables are used in the above equations and the steady state equations are subtracted out, equations relating the oscillating variables are obtained.

III, A, Task I: Analytical Model Development (cont.)

e. Exhaust Nozzle: The dynamics of a sonic exhaust nozzle have been analyzed previously. An admittance relationship exists between oscillations of pressure, velocity and entropy at the entrance to the nozzle. This relationship is:

$$\gamma P + \alpha U + \beta S = 0$$

where γ is the ratio of specific heats and α and β are functions of frequency and nozzle geometry. A computer program is available at Aerojet to evaluate α and β. It is necessary then to relate P, U, and S to the variables downstream of the combustion zone.

f. System Analysis: The overall system stability analysis consists of forming a system characteristic equation and evaluating the stability of that equation. The characteristic equation can be formed by reducing the number of system equations by substitution. This approach is cumbersome when a large number of equations and variables exist. The characteristic equation can also be found by writing all the system equations in matrix form. The characteristic equation is obtained by setting the determinant of the square matrix equal to zero. In the analysis of the nozzle, feed lines and gas tap off line only frequency response data is available. The characteristic equation is then in terms of frequency response instead of the Laplace variable. To determine stability from frequency response data a plot is made of amplitude and phase of the reciprocal of the characteristic function. System resonances occur near the peaks of the amplitude curve. The stability is determined by the behavior of the phase plot near the resonance. The band width of the resonant curve is a measure of the degree of stability or instability. Further description of this analysis is found in Reference 2.

III, A, Task I: Analytical Model Development (cont.)

Future Work

(1) The only part of the analytical model that is incomplete is the time lag. To clarify this area an analysis of the injection process will be performed. The pre-combustion zone will be divided into layers of oxidizer, fuel and water. The boundaries between layers are flexible but an increase in area of one layer means a decrease in area of another layer. The momentum and continuity equations will be written for each layer. The properties in each layer can be different except for pressure which must be the same. The flow in each layer will be considered to be isentropic.

(2) A steady state combustion model is needed for gas-gas injection. The limiting process in this case is mixing. A Spalding mixing law may be applicable; an investigation will be made to determine its suitability to the problem.

(3) Details necessary to connect all the variables for the system analysis will be formulated.

B. TASK II: ANNULAR THRUST CHAMBER ASSEMBLY TESTING

This task consists of a stability evaluation of a triplet injector in an annular combustion chamber. The injector will be rated for combustion stability by pulse testing at design and operating conditions duplicated in the Transverse Excitation Chamber. These tests are intended to provide correlations between Transverse Excitation Chambers test results and conventional combustion chamber results. The hardware for this testing, fabricated on Phase I of this program, is being modified to mount the four Photocon pressure transducers nearest the injector, flush with the inside diameter of the chamber wall. The remaining components needed for conducting the tests have been obtained. Testing is scheduled for the month of February 1969.

III, Progress of Work (cont.)

C. TASK III: TRANSVERSE EXCITATION CHAMBER TESTING

1. Design

The triplet injector previously tested on the Phase I program was modified by the addition of a copper face to increase the heat transfer to the propellant and to decrease the problems of oxidation of the injector material during the transient at shutdown. The injectors were repaired by machining off the damaged face and replacing it with a section. The entire injection pattern and oxidizer manifold was electric discharge machined into the replacement copper face and entrance of the fuel orifice contoured to decrease the pressure drop. These copper modules were then furnace brazed to the injector's steel body. A sketch of this modification is shown in Figure 5.

HIPERTHIN injectors are also being designed for testing in the Transverse Excitation Chamber. These injectors will be patterned after those designed on the "Development of Advanced Injectors Program" (Contract NAS 8-21052). Detailed design of these injectors for the excitation chamber has not been started, and will be delayed until the designs for "Development of Advanced Injectors" program are completed. Schematic representation of these injectors is shown ir Figure 6.

Replacement parts for the Excitation Chamber expended during the Phase I program have been procured, and repair of damage to the excitation chamber during testing has been completed. A photograph of the excitation chamber and its major components is shown in Figure 7.

2. Fabrication

Fabrication of the copper face triplet injectors for the Transverse Excitation Chamber has been completed. A total of eleven injectors

III, C, Task III: Transverse Excitation Chamber Testing (cont.)

containing the fine orifice pattern are available for testing and twelve replacement copper faces have been machined. Replacement copper faces have also been fabricated for the eight coarse pattern injectors. A hydroflow pattern check and flow to determine orifice coefficients has been conducted on the eleven fine pattern injectors, improved orifice coefficients have been observed for the fuel circuit as a result of the contoured orifice entrance.

The repair of the Transverse Excitation Chambers which were damaged during testing on the Phase I program has been completed and the chambers have been delivered to the test area for installation on the test stand.

Fabrication of the HIPERTHIN injectors for the Transverse Excitation Chamber has been delayed at customer's request until 1 December 1968.

3. Test Plan

A test plan for the evaluation of design variables affecting combustion stability of the triplet injector has been prepared. The test plan consists of forty tests to evaluate the effect of the following design and operation variables:

a. Chamber Pressure
b. Mixture ratio
c. Velocity ratio of injected propellants
d. Absolute velocities of injected propellants
e. Hydrogen temperature
f. Thrust per element

The effect of these variables will be explored over a range of combustion chamber acoustic frequencies from 2000 to 7000 Hz

III, C, Task III: Transverse Excitation Chamber Testing (cont.)

A complete listing of the planned tests are given in Table I and the test matrix showing the coverage of design variables is included in Table II.

Following is a brief description of the type of comparisons which will be made to evaluate the design and operation characteristics mentioned above. Evaluation of the operating parameters of chamber pressure, mixture ratio, and velocity ratio will be done by establishing a base set of data for the fine pattern injector at P_c = 1500 psia, MR = 6 and VR = 3.5. Each of these parameters are then perturbed from this base condition within the range of P_c from 1000 to 2500 psia, MR from 4 to 6 and VR from 3.5 to 7. Once the effect of the operating parameters is established on the fine pattern then variations of the injector design will be tested over the range of operation characteristics and compared to the results from the fine pattern tests. Design variations will include enlarging the fuel orifice for the fine pattern, enlarging both fuel and oxidizer orifices for the fine pattern, and increase the thrust per element by reducing the number of elements and maintain constant flow rate and velocity ratio. A test by test presentation of how these comparisons will be made is illustrated in Table II.

4. Test Stand Buildup

Test Stand J-1 used on the Phase I program is being modified for this testing. Smaller flow control valves are being installed in both the oxidizer and fuel system to provide better control of mixture ratio during start and shutdown. This should produce smoother starts and minimize hardware damage resulting from oxidizer rich shutdowns. A high pressure gaseous nitrogen purge system is being installed to reduce purge time for the oxidizer manifold. Additional pressurization capacity is being added to the present system to increase the run capability of the stand.

III, C, Task III: Transverse Excitation Chamber Testing (cont.)

5. Future Work

Testing of triplet injectors in the Excitation Chamber is planned for the next three months. Test stand modifications will be completed and testing is scheduled for mid-October. Design of the HIPERTHIN injector will also be completed this quarter.

D. TASK IV: FORMULATION OF DESIGN CRITERIA

1. Literature Survey

The purpose of this sub-task is to evaluate combustion stability information pertaining to hydrogen/oxygen systems, and compile these data in a comprehensive design document. The goal is to determine combustion stability criteria which is easily interpretable by an engine designer. To this end analytical theories, experimental test data, and combinations of both are being examined.

Bibliographical searches have been conducted from Chemical Propulsion Information Agency, Aerojet's information retrieval systems, Defense Document Center information retrieval system, National Aeronautics and Space Administration (NASA) information retrieval system, and Aerojet's Corporate Technical Information Center.

Most of the data obtained to date has been through the Chemical Propulsion Information Center, the Defense Document Center, and from the reference sections of the individual documents received.

All subsystems of the engine have been examined (e.g., injector, feed system, combustor, and nozzle). A large majority of injection elements

III, D, Task IV: Formulation of Design Criteria (cont.)

were of the concentric tube or coaxial design. Most of NASA LeRC test data were taken from their 20K lb thrust combustor (chamber diameter equals 10.77 in.) with the hydrogen density (temperature) the stability rating parameter.

The following projects provided most of the experimental data; (1) high chamber pressure programs conducted by Aerojet and Pratt and Whitney, (2) NASA LeRC research programs, (3) RL-10, J-2, and M-1 engines, and (4) Aerospike system.

The bibliography of this report is arranged by subject matter; the following five subject categories were selected to identify the pertinent information of each report: (1) stability criterion, (2) analytical, (3) experimental test data, (4) component design studies, and (5) damping devices. Some documents also may logically fall under more than one of the above categories, however, the dominant one (if definable) has been used.

2. Future Work

The foremost task for the coming quarter is the correlation of coaxial injector data. Non-coaxial data from the published literature will also be included; however, the quantity of data obtained to date has been small. These correlations will include data from the literature survey and previous combustion stability programs at Aerojet.

The literature survey from NASA's Scientific and Technical Information Facility has been resubmitted and is the only remaining formal literature search to be reviewed. Several other documents have been ordered which were found as a result of other literature searches, these will also be reviewed when received.

IV. PROGRAM MANAGEMENT

Work has been progressing satisfactorily. A schedule showing progress of work is shown in Figure 1. For the Task I effort, the gas tap-off cycle was selected as a result of a meeting between Aerojet and MSFC on 23 August 1968. No work was planned for the Task II effort in the first quarter with the exception of modifying one combustion chamber for flush mounted pressure transducers. Most of the activity this quarter has been concentrated in the Task III and IV efforts. In Task III, the design of HIPERTHIN injectors for the excitation chamber has been started and the repair of test hardware and fabrication of modified injectors completed. The literature survey is near completion and the coaxial injector test result correlations will continue on Task IV.

The following expenditures have been committed for the quarter and are shown versus the budget for both manhours and dollars in Figures 2 and 3.

	Manhours	Dollars
July	212	$ 3,364
August	1,197	28,607
September	1,572	27,706
Totals	2,981	$59,677

LIST OF REFERENCES

1. Waugh, R. C., McCalister, D. A., McBride, J. M., "A Mathematical Model for Transverse Mode Instability with Feed System Coupling for Titan IIIM," 5th ICRPG Combustion Conference, Chemical Propulsion Information Agency, October 1968.

2. Gemini Stability Improvement Program, Contract AF 04(695)-517, Final Report Vol. 3, "Analytical Model," Report GEMSIP FR-1, Aerojet-General Corp., 31 August 1965.

BIBLIOGRAPHY (ANNOTATED)

A. STABILITY CRITERION

Included under this first category are those reports which treat parameters suspected to be of prime importance in the development of a combustion stability correlation equation.

1. Wanhainen, J. P., Feiler, C. E., and Morgan, J. C., Effect of Chamber Pressure, Flow per Element, Contraction Ratio on Acoustic-Mode Instability in Hydrogen-Oxygen Rockets, NASA TN D-4733, August 1968.

Parametric studies were made examining the effect of chamber pressure, flow/element and contraction ratio on acoustic mode combustion instability. A stability correlation equation was generated from the test results.

2. Hersch, M. and Rice, E. J., Gaseous Hydrogen/Liquid Oxygen Rocket Combustion at Supercritical Chamber Pressures, NASA TN D-4172, September 1967.

Performance and to a lesser degree combustion stability data were sought as a function of the interchangeability of hydrogen and oxygen flowing through a coaxial injection element. The experimental data yielded that a somewhat higher performance (and instability) trend existed when the liquid oxygen flowed annularly around the rapidly diffusing gaseous hydrogen.

3. Wanhainen, J. P., Parish, H. C., and Conrad, E. W., Effect of Propellant Injection Velocity on Screech in 20,000-Pound Hydrogen-Oxygen Rocket Engine, NASA TN D-3373, April 1966.

The importance of velocity ratio (V_f/V_o) as a stability parameter was experimentally investigated and verified. Combustion stability was attained when the ratio exceeded 6.5.

4. Heidmann, M. F. and Weiber, P. R., Analysis of Frequency Response Characteristics of Propellant Vaporization, NASA TN D-3749, December 1966.

A linear analysis has been derived which yields the dynamic response characteristics of droplet vaporization excited by transverse waves. Approximation equations are given to solve for the pressure sensitivity and time lag as formulated by the Sensitive Time Lag Theory.

BIBLIOGRAPHY (ANNOTATED) (cont.)

5. Bloomer, H. E., Wanhainen, J. P., and Vincent D. W., "Chamber Shape Effects on Combustion Instability," 4th ICRPG Combustion Conference, Vol. 1, pages 39-55, October 1967.

Chamber wall contours (tapering), contraction ratio, nozzle shape, and injection mass distribution effects were experimentally evaluated. Nozzle variations had essentially negligible effects while physical changes in the region of the injection zone showed substantial stabilizing (destabilizing) trends. Contraction ratio (A_c/A_t) changes were important providing the throat area (A_t) was the varying parameter.

6. Feiler, C. E., "Effect of Combustor Parameters on the Stability of Gaseous Hydrogen-Liquid Oxygen Engine," 4th ICRPG, Vol. 1, pages 83-92, October 1967.

An experimental evaluation of the stability criteria relationship analytically advanced in Reference B3 was made to determine the response variations for chamber pressure, flow rate, throat area, and oscillation frequency.

7. Wanhainen, J. P., Hannum, N. P., and Russell, L. M., Evaluation of Screech Suppression Concepts in a 20,000-Pound Thrust Hydrogen-Oxygen Rocket, NASA TM X-1435, August 1967.

Various high frequency oscillation suppression concepts were reviewed: (1) injection mass distribution, (2) fluorine additive to LOX, (3) oxygen injection tube length, (4) porous injector face plate, (5) nozzle design, and (6) combustor wall cooling were studied. Items 1, 3, 4 and 5 gave stabilizing (destabilizing) trends, 2 and 6 had negligible stability effects though did affect performance.

8. Heidmann, M. F., Sokolowski, D. E., and Diehl, L. A., Study of Chugging Instability with Liquid Oxygen and Gaseous Hydrogen Combustors, NASA TN D-4005, June 1967.

An experimental research combustor was developed to study some of the "classic" design parameters found in low frequency, chugging theory. L*, injector length to diameter ratio, feed system dynamics, pressure differentials, and atomization were studied.

BIBLIOGRAPHY (ANNOTATED) (cont.)

9. Hannum, N. P. and Conrad, E. W., Performance and Screech of a Series of 2,500-Pound Element Liquid Oxygen/Hydrogen Rocket Engines (u), NASA TM X-1253, July 1966 (Confidential).

Ten different large thrust per element designs were experimentally evaluated. None exhibited high frequency oscillations though seven designs did have low frequency chugging. Performance, always a problem in large thrust/element designs, varied from 65 to 95% as the individual designs became "sophisticated."

B. ANALYTICAL

These reports detail various facets of combustion theory related to the technical understanding of stability parameters.

1. Conrad, E. W., Hannum, N. P., and Bloomer, H. E., Photographic Study of Liquid-Oxygen Boiling and Gas Injection in the Injector of a Chugging Rocket Engine (u), NASA TM X-948, 1964 (Confidential).

A photographic study was made of the effects of introducing helium gas into the oxidizer and fuel flow passages of an injector. This action had a stabilizing trend on low frequency chugging with flow into the oxidizer circuit exhibiting the greater effect.

2. Heidmann, M. F., Oxygen - Jet Behavior During Combustion Instability in a Two-Dimensional Combustor, NASA TN D-2725, March 1965.

Photographic studies were made through the walls of a two-dimensional "pancake" motor to examine the behavior of liquid oxygen breakup in a gaseous hydrogen field. Instabilities were triggered by GN_2 injection. An oscillatory combustion front together with atomization variations were observed during both the high and low frequency wave patterns.

3. Feiler, C. E. and Heidmann, M. F., Dynamic Response of Gaseous-Hydrogen Flow System and Its Application to High Frequency Combustion Instability, NASA TN D-4040, June 1967.

An investigation was made to determine if high frequency coupling could occur between gaseous hydrogen flow oscillations and chamber pressure. An analytical model was derived using injector design variables and evaluated against experimental data.

BIBLIOGRAPHY (ANNOTATED) (cont.)

4. Priem, R. J. and Guentert, D. C., Combustion Instability Limits Determined by a Non-Linear Theory and One-Dimensional Model, NASA TN D-1409, July 1962.

This report details the classical Priem/Guentert theory. The basic energy, conservation, continuity equations are summarized and the effect of propellant vaporization as a controlling factor on burning rate is given.

5. Priem, R. J. and Heidmann, M. F., Propellant Vaporization as a Design Criterion for Rocket-Engine Combustion Chambers, NASA TR R-67, 1960.

Preliminary development of the vaporization model and its relationship to combustion.

6. Phillips, B. and Morgen, J. C., Mechanical Absorption of Acoustic Oscillations in Simulated Rocket Combustion Chambers, NASA TN D-3792, January 1967.

Analytical treatise on acoustic liner theory, the importance of cavity gas temperature, and the degree of combustor wall coverage required for sufficient resonant suppression.

C. EXPERIMENTAL

To ascertain the validity of particular correlation equations, test data are required. The references listed below describe those test programs where data can be obtained for evaluation of correlating equations.

1. Ito, J. I., Development of Liquid Oxygen/Liquid Hydrogen Gas Generators for the M-1 Engine, NASA CR 5-4812, June 1966.

An extensive experimental and analytical report giving the design philosophy behind the M-1 gas generator development. Large thrust/element, baffling, and effects of velocity ratio are discussed in the stabilizing of the combustion. Since this report covers gas generators, combustion is in the low (1 to 2) mixture ratio region.

BIBLIOGRAPHY (ANNOTATED) (cont.)

2. Advanced Engineering Program System and Dynamics Investigation (Aerospike) (u), Vol. 2 of Rocketdyne Engineering Final Report on Contract No. NAS 8-19, 1 January 1966 to 15 August 1967 (Confidential).

This report summarizes eighteen months of the Aerospike Program including design considerations, test data, etc., with a major emphasis placed on combustion stability.

3. 9th Liquid Propulsion Symposium, Chemical Propulsion Information Agency Pub. No. 155, Vol. 1, September 1967, pages 21-47 ("Results of the 1st Phase of an Aerospike Engine Test Program" (u)) and pages 49-83 ("Full-Scale Component Testing for a High Performance Oxygen/Hydrogen Rocket Engine" (u)) (Confidential).

Two articles in this publication are of special interest. One covers the results of Phase I Aerospike Test Program, while the second deals with staged combustion in a LOX/H_2 engine system.

4. Bastress, E. K., Harris, G. H., and Miller, I., Statistical Derivation of Design Criteria for Liquid Rocket Combustion Instability, NASA CR-72370, December 1967.

A statistical approach was taken to ascertain if a design criteria equation could be derived to predict stable combustion--no equation was found. This report gives a comprehensive listing of numerous design parameters, test histories, etc.

5. Tomazic, W. A., et al., M-1 Injector Development - Philosophy - Implementation, AIAA Paper No. 67-461.

Injection element design philosophy is discussed as well as factors relating to the design of the RL-10 and J-2 systems.

6. Scott, H. E., Bloomer, H. E., and Mansour, A. H., M-1 Engine Subscale Injector Tests, NASA TN D-4053, July 1967.

Several injection element configurations are evaluated to establish design criteria which satisfactorily meets both the stability and performance requirements. Large injection pressure differentials and large velocity ratios (V_f/V_{ox}) were found to act as deterrents to high and low frequency instability.

BIBLIOGRAPHY (ANNOTATED) (cont.)

7. Hoehn, F. W. and Lawhead, R. B., "Combustion Instability on Liquid Oxygen (LOX)/Hydrogen Rocket Systems (U)," 5th Liquid Propulsion Symposium 1968 (Confidential).

The design philosophy of the J-2 injection system is discussed. Two-dimensional motor tests, coaxial element design, mixture ratio and hydrogen temperature as stability correlating parameters are discussed. Streak film studies were made.

8. Atherton, R. R., Advanced Cryogenic Rocket Engine Program, Staged-Combustion Concept (u), Final Report AFRPL-TR-67-298, Vol. 1, December 1967 (Confidential).

Comprehensive description of a staged combustion engine system. No stability correlations are made.

9. Summary Report on an Investigation of Combustion Instability for Liquid Oxygen and Liquid and Cold Gaseous Hydrogen Propellants, Pratt & Whitney, Report FR-1005, June 1964.

Extensive report covering combustion stability experimental results from two-dimensional and RL-10 combustors. Acoustic liners evaluated, thorough test and design details given, together with an analytical combustion stability theory.

10. Anonymous, High Chamber Pressure Operation for Launch Vehicle Engine Program, Final Report, Aerojet-General Corporation under NASA Contract NAS 8-4008, June 1964.

11. Anonymous, An Experimental Investigation of Combustion Stability Characteristics at High Chamber Pressure, Final Report, Phase 2, Aerojet-General Corporation under NASA Contract NAS 8-11741, August 1966.

D. DESIGN DETAILS

Certain references yielded a considerable amount of design information, giving detail descriptions of the various system components (e.g., injector, nozzle, etc.).

BIBLIOGRAPHY (ANNOTATED) (cont.)

1. Goelz, R. R., Combustion Instability in Steel and Ablative Rocket Chambers, NASA TM X-1511, February 1968.

Combustor wall material selection (steel vs ablative) was evaluated to determine its effect on combustion stability. Seventeen tests yielded no particular trend for either material.

2. Hersch, M., Effect of Interchanging Propellants on Rocket Combustion Performance with Coaxial Injection, NASA TN D-2169, February 1964.

Performance and to a lesser degree combustion stability data were sought as a function of the interchangeability of hydrogen and oxygen flowing through a coaxial injection element. The experimental data yielded that a somewhat higher performance trend existed when the liquid oxygen flowed annularly around the rapidly diffusing gaseous hydrogen.

3. Hannum, N. P. and Conrad, E. W., "Some Injector Element Detail Effects on Screech in Hydrogen/Liquid Oxygen Rockets," 4th ICRPG, pages 31-38, October 1967.

A parametric study was experimentally performed on design variations of the concentric injection element. Injection angle; inner tube thickness, concentricity, recess, or extension; and thrust per element were evaluated. All parameters except concentricity had an effect on combustion stability.

4. Rocketdyne 3rd and 4th Quarterly Progress Reports on Advanced Engine Design Study (Aerospike) (u), Contract NAS 8-203849 (Confidential).

System design philosophy, experimental test data, etc., are covered in various detail in these reports.

5. See also References A5, A6, C6, A7, and A8.

E. DAMPING DEVICES

References dealing with the design of acoustic damping mechanisms (liners, baffles, particulate injection, and propellant mass distribution) as they are related to LOX/H_2 system are given.

BIBLIOGRAPHY (ANNOTATED) (cont.)

1. Wanhainen, J. P., Bloomer, H. E., Vincent, D. W., and Curley, J. K., Experimental Investigation of Acoustic Liners to Suppress Screech in Hydrogen-Oxygen Rockets, NASA TN D-3822, February 1967.

Acoustic liner theory and design were used successfully in suppressing high frequency instability. From a design viewpoint, liners need not extend the entire chamber length but are most effective when they cover approximately the first 20% downstream from the injector.

2. Absorbing Liners for Rocket Combustion Chambers - Theory and Design Techniques (u), AFRPL-TR-66-234 (Confidential).

Comprehensive acoustic liner design theory is presented in this document. Mathematical descriptions, acoustical/mechanical analogies, engine design details, etc., are discussed.

3. Heidmann, M. F. and Povinelli, L. A., "An Experiment on Particulate Damping in a Two-Dimensional Hydrogen/Oxygen Combustor," 4th ICRPG, pages 311-320, October 1967.

The introduction of aluminum powder (5 micron) into the combustion process was evaluated using the two-dimensional "pancake" motor research combustor. Mass additions above 0.02% had a stabilizing effect, whereas below that figure destabilizing occurred.

4. Hannum, N. P. and Scott, H. E., "The Effect of Several Injector Face Baffle Configurations on Screech in a 20,000-Pound Thrust Hydrogen/Oxygen Rocket," 3rd ICRPG, pages 587-595, October 1966.

Seventeen different baffle configurations were evaluated in a 94 test series. Compartment size and baffle length were both important design parameters.

5. Hannum, N. P., Bloomer, H. E., and Goelz, R. R., Stability Effects of Several Injector Face Baffle Configurations on Screech in a 20,000-Pound Hydrogen/Oxygen Rocket Engine, NASA TN D-4515, April 1968.

Baffle design concepts are tested for their stabilizing effects. Length and compartment dimensions evaluated for this particular engine.

6. See also References A7 and B7.

BIBLIOGRAPHY (ANNOTATED) (cont.)

F. MISCELLANEOUS

These reports did not directly fall into any of the previous five categories; however, they do contain information important to the overall understanding of combustion stability in LOX/H_2 rocket engines.

1. Main, H. V., George, D., Mahugh, V. L., and Tepe, L. E., Cellular Combustion Chamber Program (Project Scorpio), Phase 2 Final Report, "Oxygen/Hydrogen Injector Thrust per Element Size Maximization" (u), AFRPL-TR-65-199, December 1965 (Confidential).

   This report dealt primarily with the performance aspects of large thrust/element injection. One dynamic response pressure transducer located through the combustor walls registered low amplitude chugging oscillations.

2. Unconventional Injector Concepts Development (u), Unicode Quarterly, 1962 (Confidential).

   A brief treatise on the feasibility of large thrust-per-element injection is covered. Some test and design data are given which could possibly be used to validate a stability correlation equation.

3. Unconventional Injector Feasibility Investigation (u), Rocketdyne Final Report R-3969, March 1963, pages 125-141 (Confidential).

   A unique injection technique was demonstrated yielding some preliminary data of value for the Aerospike Engine System. Effect of hydrogen temperature, divergence rings, plus a broad coverage of injector design details are given though experimental stability data are somewhat limited by a lack of high frequency instrumentation.

4. Combs, L. P., et al., Steady-State Rocket Combustion of Gaseous Hydrogen and Liquid Oxygen, Part 2, "Analysis for Coaxial Injection," Rocketdyne Report 64, 29 March 1965.

   As this report covers steady-state combustion, it is not of direct value to instability studies. It is important from the aspect of detailing some of the design philosophy behind coaxial injectors.

BIBLIOGRAPHY (ANNOTATED) (cont.)

5. Thomas, J. P., Transient Pressure Measuring Methods Research - Final Summary Technical Report, Princeton University, March 1967.

Since the attainment of high frequency information is vital to the establishment of a design criteria/stable combustion guide, this report dealing with the virtues of high response pressure measuring instrumentation is included.

6. Multi-Tech Inc., Final Report, Combustion Processes in a Bipropellant Liquid Rocket Engine, under Subcontract JPL 950926, from NASA Contract NAS 7-100.

An extensive literature search covering all phases of bipropellant liquid combustion.

7. Development of LOX/H_2 and LOX/RP-1 Engines for Saturn/Apollo Launch Vehicle, AIAA Papers 68-568(9).

Brief discussions of the LOX/H_2 (LOX/RP-1) engines are made in these papers.

TABLE I

TEST PLAN FOR EVALUATION OF TRIPLET INJECTORS IN THE TRANSVERSE EXCITATION CHAMBER

Test No.	No. of Injectors	Injector** Dash No.	P_c (psia)	$\dot{w}_{f\ total}$ (lb/sec)	$\dot{w}_{ox\ total}$ (lb/sec)	H_2 temp* (°R)	A_{t_2} (in.^{2})
1	3	1	1500	1.8	7.2	80	1.42
2	3	1	1500	1.29	7.7	80	1.42
3	3	1	2500	1.8	7.2	80	0.85
4	3	1	2500	1.29	7.7	80	0.85
5	4	1	1500	2.4	9.6	80	1.94
6	4	1	1500	1.72	10.3	80	1.94
7	4	1	2500	1.72	10.3	80	1.16
8	2	1	1500	1.2	4.8	80	0.97
9	2	1	1500	0.85	5.14	80	0.97
10	2	1	2500	0.85	5.14	80	0.58
11	2	5	1000	1.29	7.7	120	2.14
12	3	5	1000	1.29	7.7	175	2.14
13	4	5	1000	1.72	10.3	120	2.9
14	4	5	1000	1.72	10.3	175	2.9
15	4	1	1000	1.72	10.3	120	2.9
16	4	1	1500	1.72	10.3	200	1.94
17	3	1	1000	1.29	7.7	120	2.14
18	3	1	1500	1.29	7.7	200	1.42
19	2	1	1000	0.85	5.14	120	1.45
20	2	1	1500	0.85	5.14	200	0.97
21	2	3	1000	0.85	5.14	220	1.45
22	2	3	1500	0.85	5.14	140	0.97

TABLE I (cont.)

Test No.	No. of Injectors	Injector** Dash No.	P_c (psia)	$\dot{w}_f$ total (lb/sec)	$\dot{w}_{ox}$ total (lb/sec)	H_2 temp* (°R)	A_{t_2} (in.^{2})
23	3	3	1500	1.8	7.2	90	1.42
24	3	3	1000	1.29	7.7	220	2.14
25	3	3	1500	1.29	7.7	140	1.42
26	3	3	1500	1.29	7.7	270	1.42
27	3	3	1500	1.29	7.7	155	0.85
28	3	3	1000	1.72	10.3	220	2.9
29	4	3	1000	1.72	10.3	400	2.9
30	4	3	1500	1.72	10.3	140	1.94
31	4	3	1500	1.72	10.3	270	1.94
32	4	7	1500	1.72	10.3	240	.94
33	3	7	1500	1.29	7.7	240	1.42
34	3	7	1000	1.29	7.7	190	2.14
35	2	7	1000	0.85	5.14	190	1.45
36	2	7	1500	0.85	5.14	240	0.97
37	7	3	1500	3.0	18.0	140	3.39
38	7	1	1500	3.0	18.0	80	3.39
39	7	1	1500	4.2	16.8	80	3.39
40	7	1	[illegible]	3.0	18.0	80	2.03

*Tank temperature determined using the required propellant density at P_c + 1000 psi

**-1 = fine pattern triplet injector insert

-3 = modified fine pattern triplet injector insert

-5 = coarse pattern triplet injector insert

-7 = modified fine pattern triplet injector insert with enlarged ox orifice

TABLE II

EXCITATION CHAMBER TEST DATA COVERAGE

Operation Parameters	P_c (psia)	1000		1500				2500	
	MR	6		4	6			4	6
	VR.(**)	7	12	3.5	3.5	7	12	3.5	3.5
Chamber Acoustic Frequency (Hz)									
2000				x(*)	xy				x
3500		wxy	wy	x	xy	xy	z		x
4500		wxy	wz	xy	xy	xy	z	x	xy
7000		xy	z	x	x	x	z		x

NOTE: *The letter designations refer to the following injector designs:

w - Coarse pattern, x - fine pattern, y - fine pattern with fuel orifice enlarged.
z - Fine pattern with both fuel and oxidizer orifices enlarged.

**VR refers to the ratio of velocities of fuel to oxidizer of injected propellants.

PROGRAM SCHEDULE
STABILITY CHARACTERIZATION OF ADVANCED INJECTORS
CONTRACT NAS 3-20672

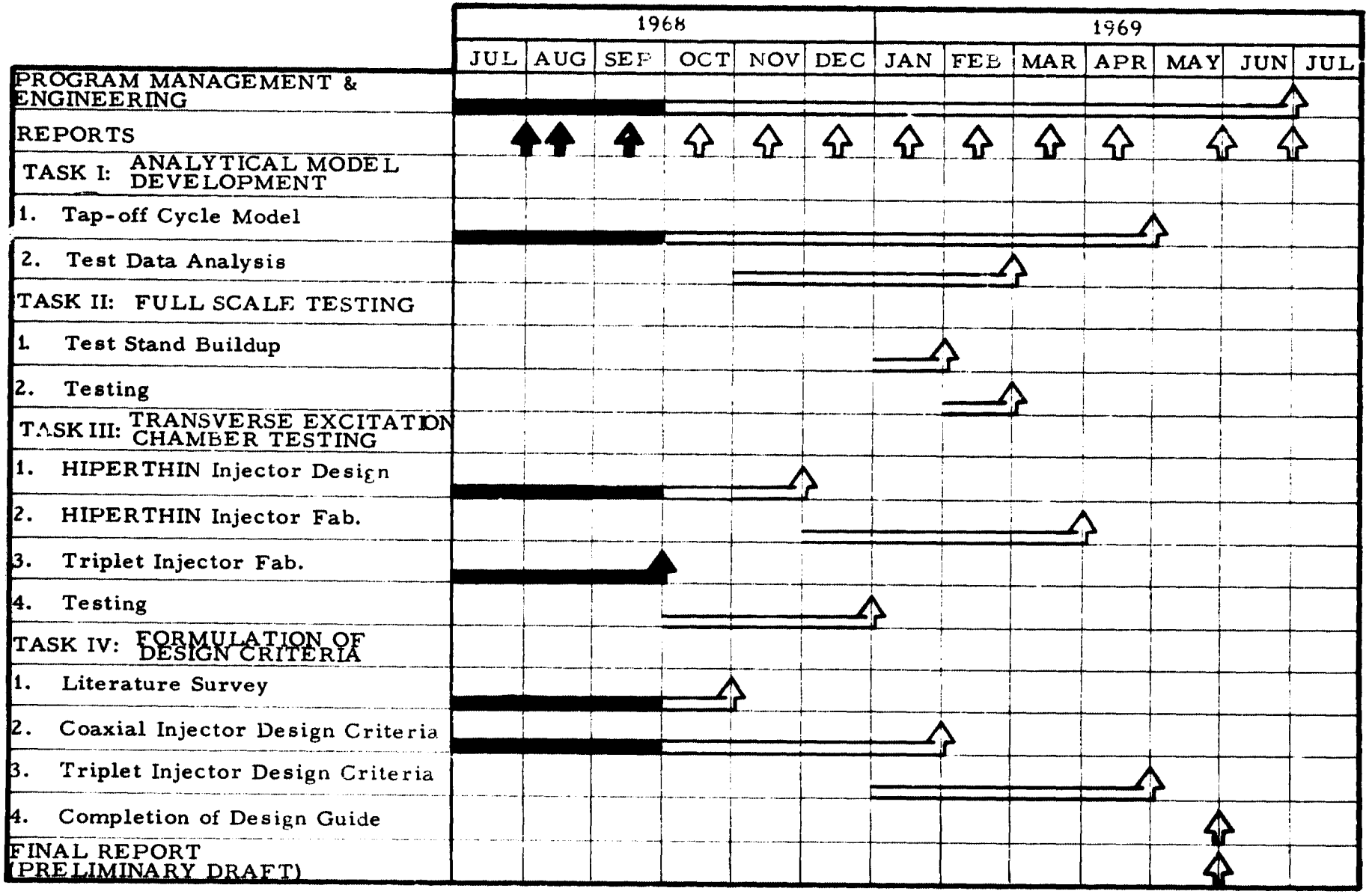


Figure 1

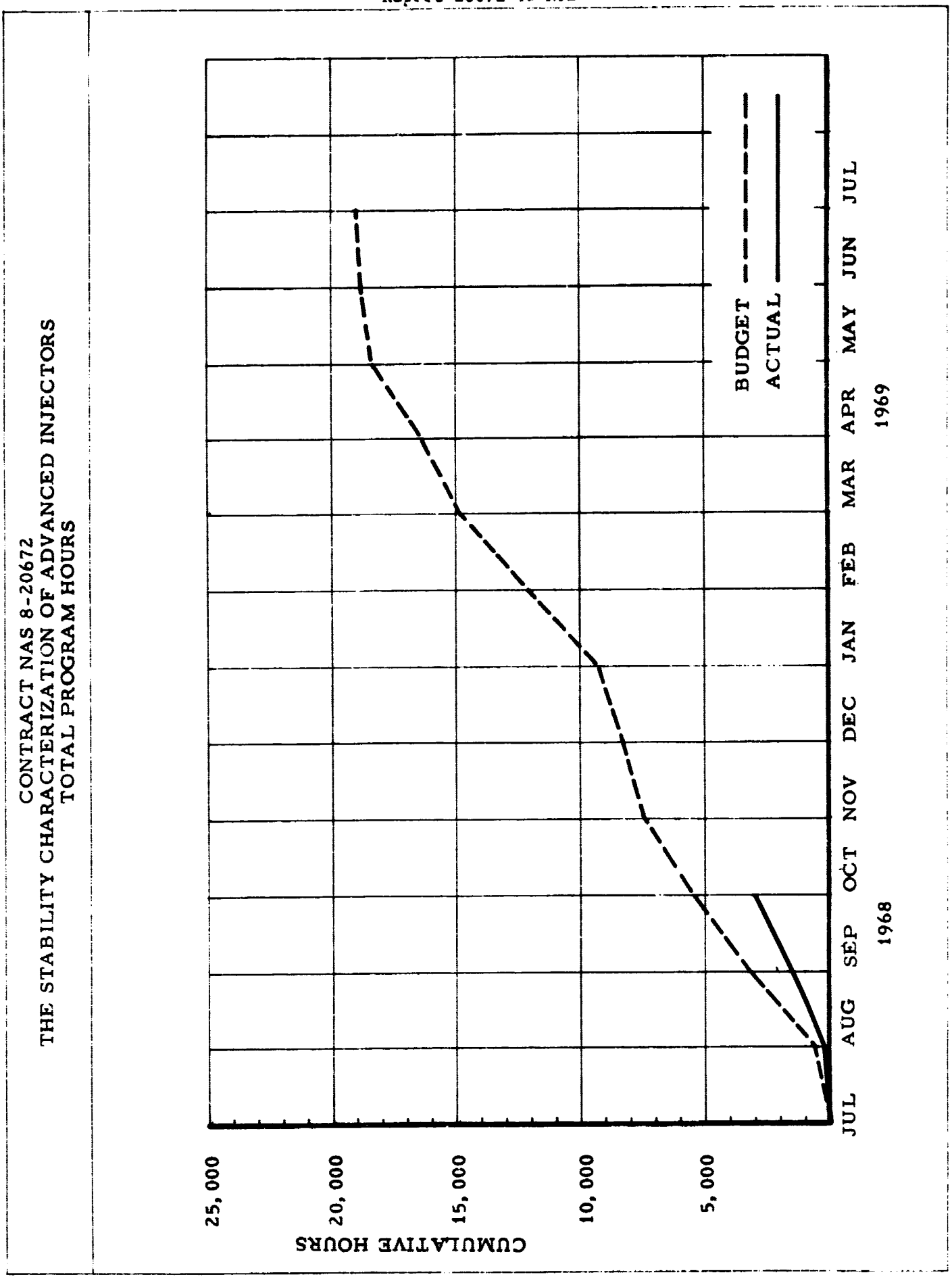

CONTRACT NAS 8-20672
THE STABILITY CHARACTERIZATION OF ADVANCED INJECTORS
TOTAL PROGRAM HOURS
CUMULATIVE HOURS
25,000
20,000
15,000
10,000
5,000
BUDGET
ACTUAL
JUL AUG SEP OCT NOV DEC JAN FEB MAR APR MAY JUN JUL
1968
1969

Figure 2

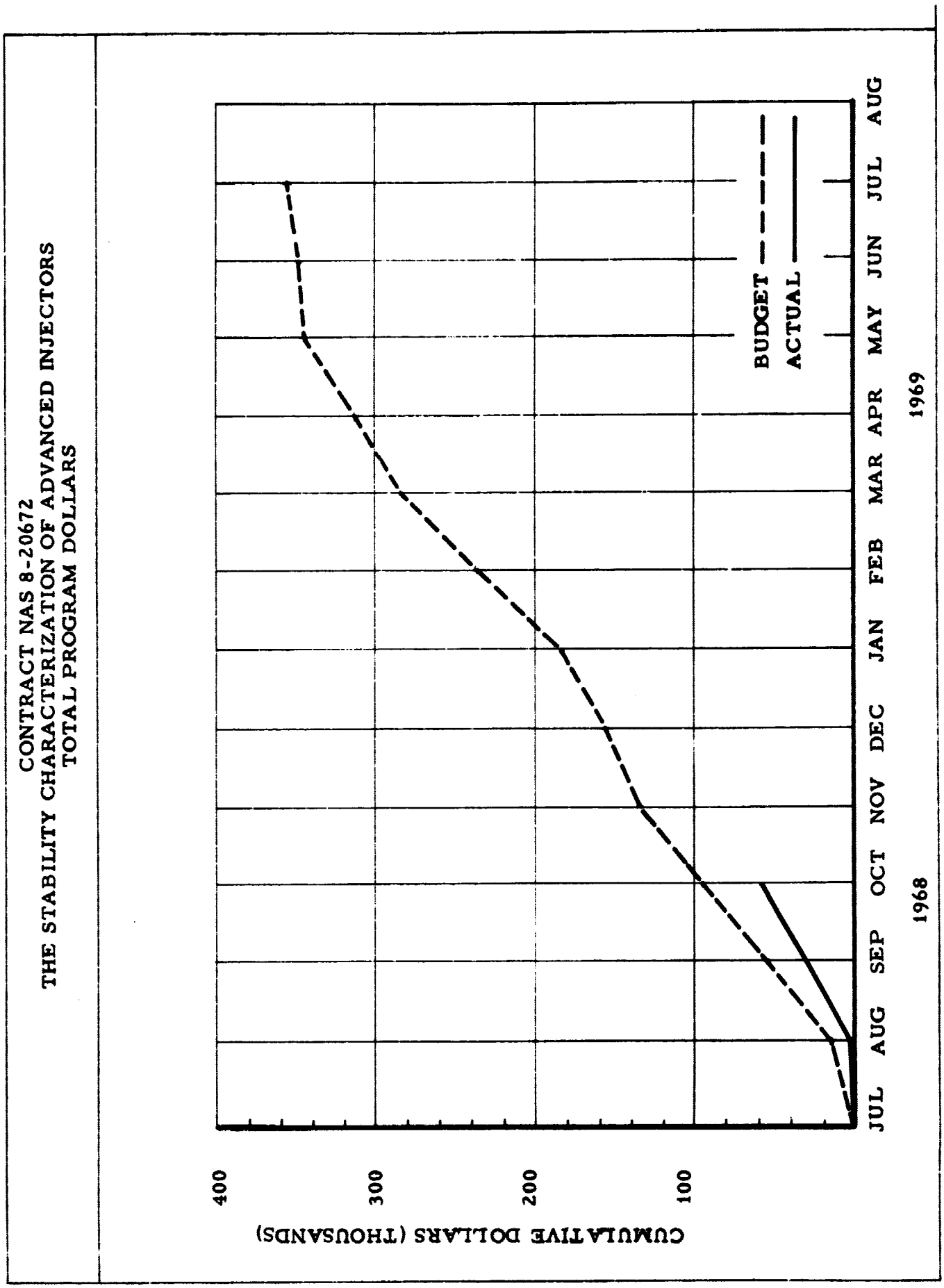
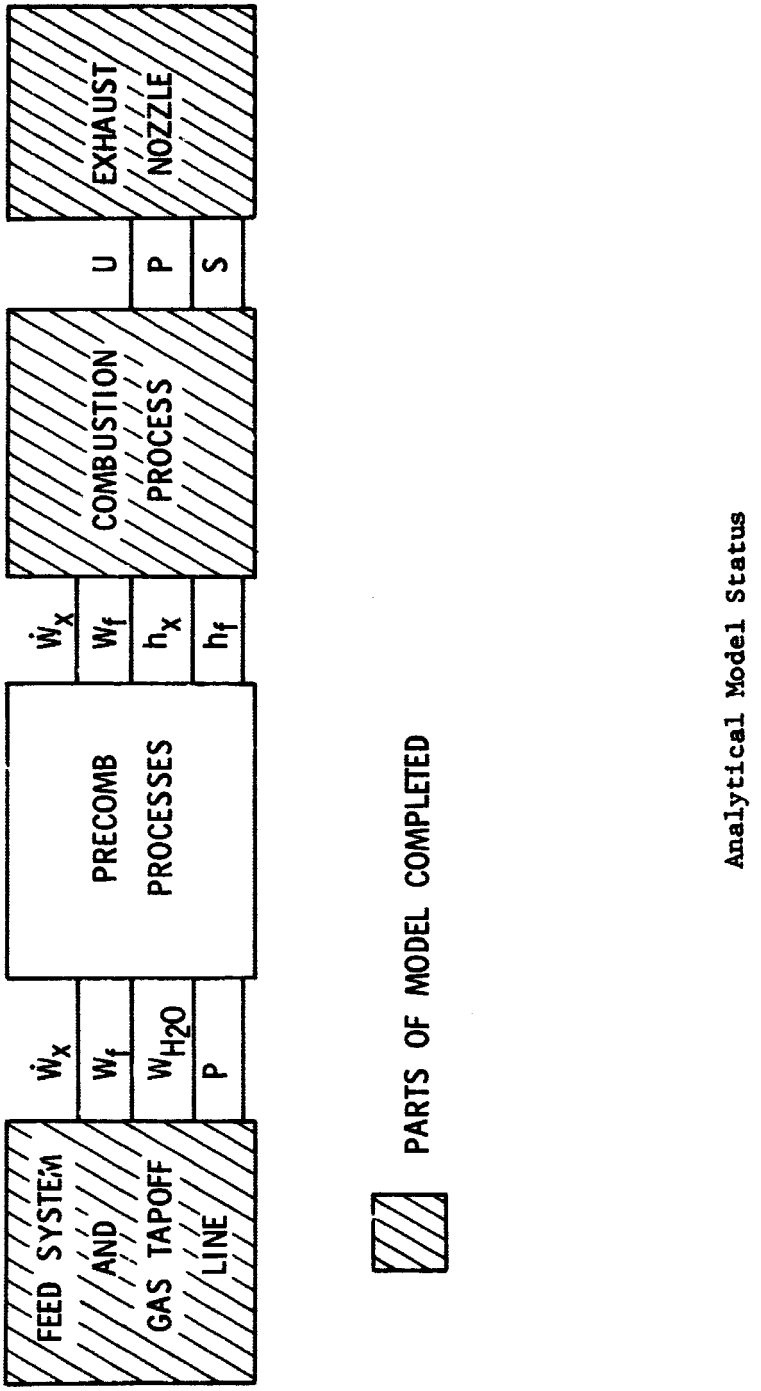
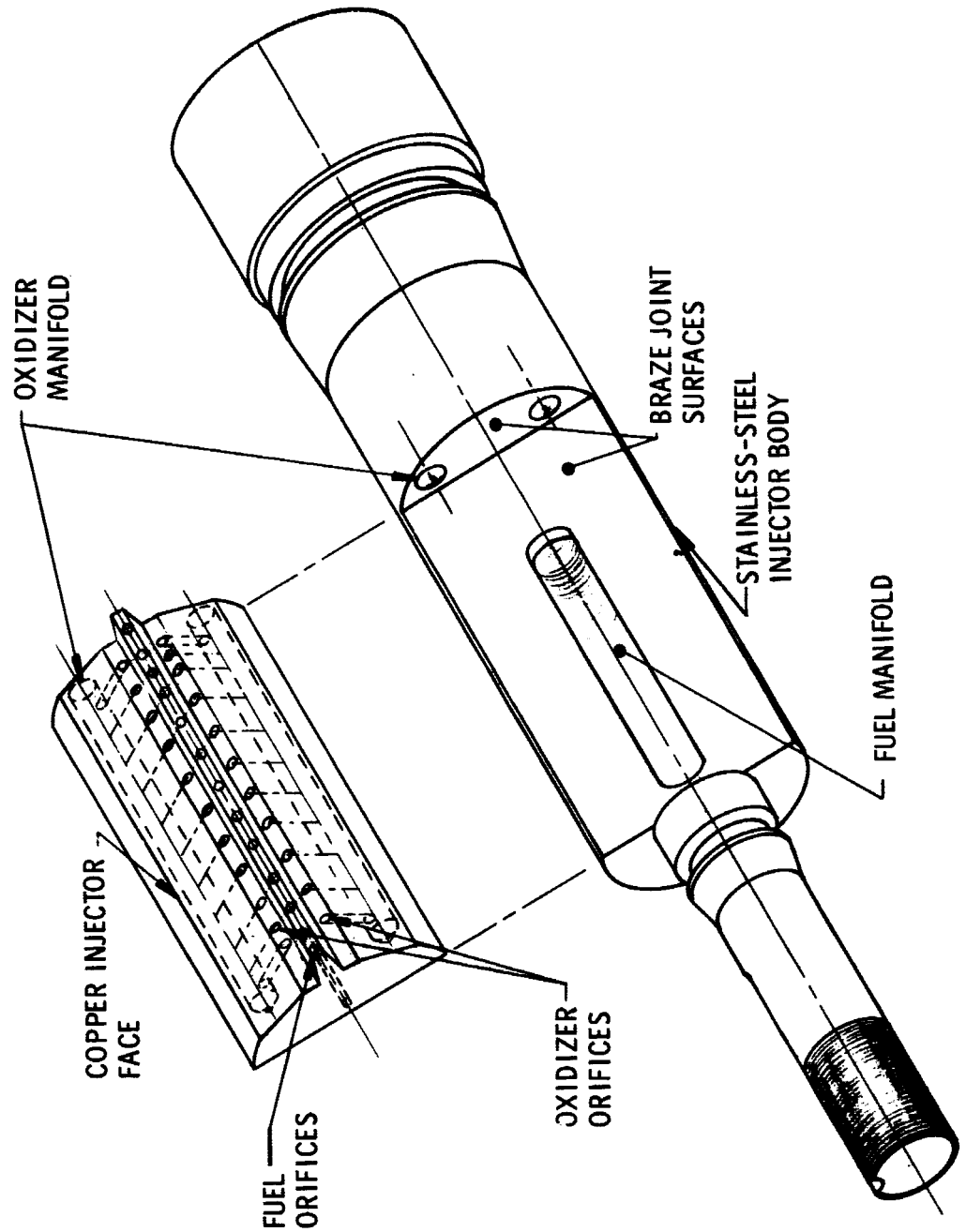

CONTRACT NAS 8-20672
THE STABILITY CHARACTERIZATION OF ADVANCED INJECTORS
TOTAL PROGRAM DOLLARS
CUMULATIVE DOLLARS (THOUSANDS)
400
300
200
100
BUDGET
ACTUAL
JUL AUG SEP OCT NOV DEC JAN FEB MAR APR MAY JUN JUL AUG
1968
1969

Figure 3



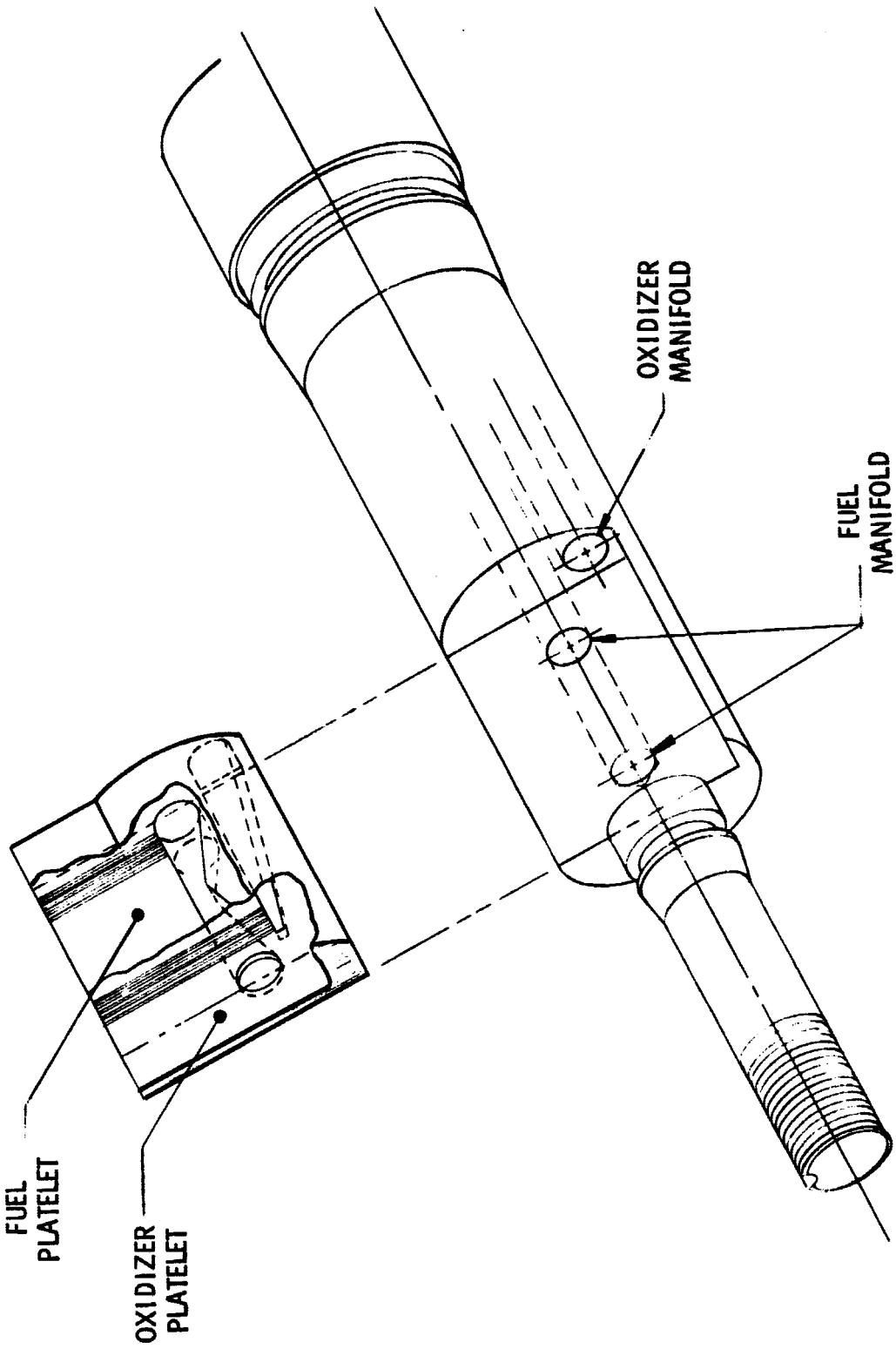
Analytical Model Status

Figure 4



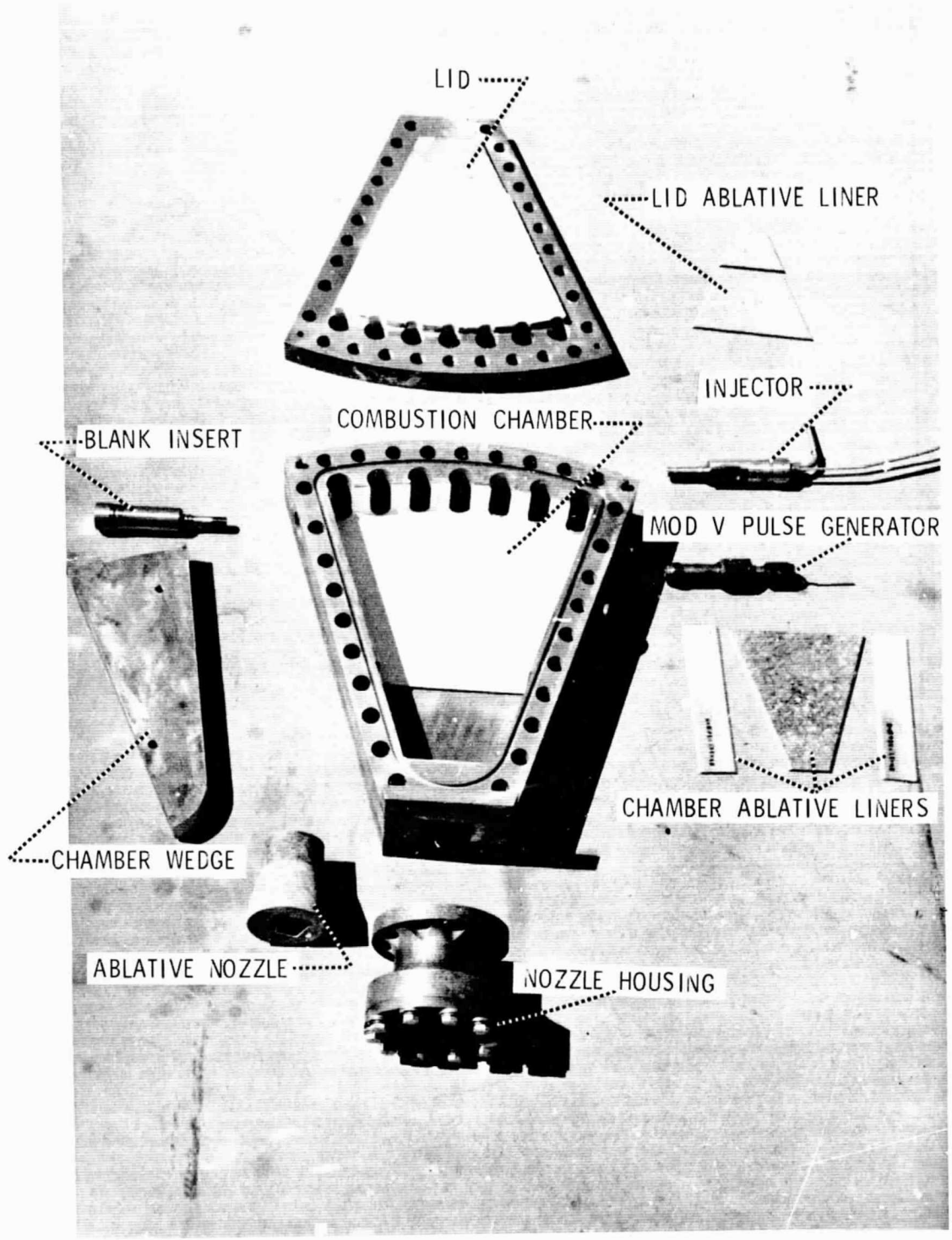
Triplet Injector for Excitation Chamber

Figure 5



HIPERTHIN Injector for Excitation Chamber

Figure 6



Transverse Excitation Chamber

Figure 7