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FINAL SUMMARY REPORT
for
SPACE SHUTTLE ORBIT MANEUVERING ENGINE REUSABLE THRUST CHAMBER PROGRAM

DATE: MAY 1975

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FOR

SPACE SHUTTLE ORBIT MANEUVERING ENGINE

REUSABLE THRUST CHAMBER PROGRAM

DATE: MAY 1975

PREPARED BY:

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TEXTRON Bell Aerospace

APPROVED BY:

Merl Lausten, Program Monitor
NASA-Lyndon B. Johnson Space Center
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FOREWORD

The purpose of this contract was to furnish information useful to the design selection and technology base of the Orbit Maneuvering Engine for the Space Shuttle efforts. Both the analytical and experimental efforts performed at the Bell Aerospace Company to accomplish this effort are summated in this report. The NASA-JSC Program Monitor for the NAS9-12803 program was Merl Lausten.

ABSTRACT

The objective of this contract was to evaluate potential reusable thrust chamber and injector concepts for the Space Shuttle Orbit Maneuvering Engine. The initial efforts included trade off studies and subsequent recommendations for the selection of propellants and thrust chamber cooling concepts. The Bell ADEPT (Advanced Design Engine Parametric Technology), computor program was developed and utilized for this effort.

Demonstration tasks were then performed to confirm data and provide information relative to the difficulties to be encountered in development. These tasks included component testing such as heat transfer tube testing with a silicone oil additive and combustion stability tests with a variety of damper configurations. Thrust chamber cooling concept testing included tests on a 6000 lb. thrust insulated columbium thrust chamber, and regeneratively cooled tests using a company furnished channel wall regeneratively cooled thrust chamber. The results of this testing produced excellent agreement with predicted values where a simulated altitude performance of 317 seconds specific impulse was obtained on the regeneratively cooled thrust chamber while producing compatible heat rejection to the thrust chamber walls. The final task of the program was to demonstrate that the originally developed 10 inch diameter combustion pattern could be compressed to operate in an 8 inch diameter thrust chamber. This task was completed with both performance and combustion stability demonstrated.

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I. INTRODUCTION

The large selection of propellants and hardware concepts which have been evaluated in recent years have resulted in a technology base where reasonable selectivity can be used for any application. The Space Shuttle OME was one such application where a number of designs could be made to function, and where a selection of cooling concepts as well as propellants was warranted prior to full development.

To assist in this selection, an analytical and experimental program was undertaken by the Bell Aerospace Company to present data on selection criteria and to show by demonstration, the capability of producing practical designs. These efforts were performed under contract NAS9-12803. This report summarizes the effort performed. The final detailed report is No. 8693-95001.

II. PROGRAM OBJECTIVES

The overall objective of the contract was the determination of the feasibility of potential reusable thrust chamber concepts for the OME. The methods of accomplishing the objective was to first examine propellant candidates, analytically combine these propellants with potential cooling schemes, and produce a data base of engine data which would assist potential vehicle contractors in an OME configuration selection. The data base verification was performed by the demonstration of a thrust chamber of a selected coolant scheme design.

The demonstration portion of the program was originally planned for a design selected from the propellant/coolant configuration trade studies. This objective was fulfilled with the operation of a full scale insulated columbium thrust chamber and later expanded to include alternate cooling demonstrations with a company furnished regeneratively cooled thrust chamber. Additionally, combustion stability of the injectors and a reduced size thrust chamber were experimentally verified as proof of concept demonstrations of the design and study results.

III. RELATIONSHIP TO OTHER NASA EFFORTS

Other related contracts were NAS9-12802 (Rocketdyne Division, Rockwell International), and NAS9-13133 (Aerojet Liquid Rocket Corporation). Both of these contracts examined the operation and performance which could be expected for the OME and OMS development programs. An additional related injector combustion stability program was also conducted at Rocketdyne under Contract NAS9-12524.

IV. METHOD OF APPROACH AND PRINCIPAL ASSUMPTIONS

The NAS9-12803 program was subdivided into fifteen tasks, each task was named and described with an individual objective. The various task descriptions are listed in Table IV-1.

For this report, these tasks have been generalized into the primary component of investigation as shown in Table IV-2. balance of this report addresses the approach assumptions and accomplishments achieved in evaluation of these primary components.

A. REUSABLE OME THRUST CHAMBER PARAMETRIC STUDIES

This effort included both Task I and II of the program and was performed to produce data for both OME and OMS configuration selection. The baseline engine was defined as 6000 lbs thrust, 125 psia chamber pressure with a nozzle exit diameter of 50 inches. Three types of regenerative cooled and one non-regenerative cooled thrust chambers were to be studied; channel wall, drilled aluminum and tubular regenerative cooled and insulated columbium non-regenerative cooled concepts were included. The engine assembly definitions included a stainless steel injector with appropriate thrust chamber attachment, radiation cooled nozzle extension, gas actuated series - parallel redundant engine propellant valves, and the engine gimbal mount. Baseline propellants were N_2O_4 and MMH with 50/50 an alternate fuel. The required parametric output data included specific impulse, engine assembly weight, envelope and feed pressures for the range of design variables:

Thrust	-	4000 t 10,000 lb_f
Chamber Pressure	-	100 to 200 psia
Nozzle Exit Diameter	-	40 to 60 inches
Mixture Ratio	-	Optimum ±20%
Nozzle % Bell	-	Approximately 72 to 100

A similar effort to Task I was performed in the Task II effort except the oxidizer was changed to LOX and MMH, N_2H_4, propane and RP-1 fuels were examined. The engine assembly definitions included an ignition system for the non-hypergolic propellants The preferred engine concept was to be established against the same technical and cost ratings as Task 1. A final recommendation was also made considering both Task I and Task II propellants and engine definitions.

B. INSULATED COLUMBIUM THRUST CHAMBER (TASKS III, IV, V, VI AND VIII)

This contract was originally organized to generate data leading to the selection of an OME design, then to prove the design viable by the demonstration of a selected thrust chamber. The

TABLE IV-1

Task Description - Reusable Thrust Chamber Program

Task No.	Title	Task Description
I	Reusable OME Thrust Chamber Evaluation	Definition and parametric analysis of pressure fed engine assemblies utilizing N_2O_4/MMH and N_2O_4/50-50 propellants.
II	Alternate OME Propellant Combinations	Definition and parametric analysis of selected pressure fed engine assemblies utilizing LOX/MMH, LOX/N_2H_4, LOX/Propane and LOX/RP-1 propellants.
III	Columbium Thrust Chamber Design	Detailed design of an experimental thrust chamber to demonstrate operation of an OME size Insulated Columbium Reusable Thrust Chamber.
IV	Columbium Thrust Chamber Fabrication	Fabrication of Task III thrust chamber.
V	Columbium Thrust Chamber Testing	Hot fire testing of the demonstrations columbium thrust chamber at simulated altitude. Tests were conducted with and without external insulation.
VI	Alternate Columbium Chamber Fabrication	Fabrication of a reduced length (L*) columbium thrust chamber.
VII	Heat Transfer Testing	Heated tube heat transfer testing of MMH with a silicone oil (30) additive.
VIII	Alternate Thrust Chamber and ρ Testing	Hot fire testing of demonstration columbium thrust chambers with lengths producing 26, 30 and 34L*. Film coolant rates (ρ) were varied and NTO/50-50 propellants also tested.
IX	Injector Characterization and Stability Testing	Hot fire injector performance heat rejection and stability tests were conducted at simulated regeneratively cooled thrust chamber operating conditions.
X	Regeneratively Cooled Thrust Chamber Demonstration	Hot fire testing to demonstrate operation of a company furnished regeneratively cooled thrust chamber.
XI	OME Model Injector	Delivery to NASA of an injector employed for chamber demonstration tests.
XII	Heated Propellant Injector Stability Testing	Hot fire stability testing to simulate regeneratively cooled propellant usage.
XIII	WSTF Test and Analysis Support	The Bell regeneratively cooled chamber was adapted to accept the WSTF furnished full nozzle extension and hot fire tests made at WSTF. Comparison of data between the Bell and WSTF tests was made.
XIV	Triplet Injector Dynamic Stability Testing	Combustion stability margins were examined on the 10" diameter injector by varying acoustic cavity areas and depths.
XV	OME 8 Inch Triplet Injector Optimization	An 8 inch triplet injector was designed, fabricated and tested to compare performance and stability to the 10" injector.

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TABLE IV-2

Major Program Investigation Components

Primary Component of Investigation	Contract Task Number
1. Reusable OME Thrust Chamber Parametric Studies	I, II
2. Insulated Columbium Thrust Chamber	III, IV, V, VI, VIII
3. Regeneratively Cooled Thrust Chamber	X, XIII
4. Heat Transfer Tube Tests	VII
5. Injector Evaluation and Stability Tests	IX, XI, XII, XIV
6. Reduced Diameter Combustor	XV

Bell program resulted in the selection of an Insulated Columbium Thrust Chamber for the initial demonstration. The design selected for the columbium chamber is shown in Figure IV-1. The design fabrication and test of the demonstration chamber was performed in Tasks III, IV, V, VI and VIII. Task III was related to design, Tasks IV and VI to fabrication of the two test chambers and Tasks V and VIII to demonstration.

1. COLUMBIUM THRUST CHAMBER DESIGN (TASKS III AND VI)

The columbium thrust chamber was designed in accordance with the Task I studies which set the chamber diameter at 10 inches, chamber and throat thicknesses at 0.105 and 0.150 respectively and the columbium combustion chamber length at 10.013 inches. The chamber length provided an L* of 30 inches including the combustion volume provided by the injector assembly. The engine was designed for N_2O_4/MMH propellants at nominal conditions of 6000 lbf and 125 psia chamber pressure (Figure IV-2). The divergent nozzle of the demonstration chamber was restricted to an area ratio of 15:1 by the altitude test facility at Bell. The development nozzle contour was established as a truncated section of the full nozzle contour rather than an optimized 15:1 shape. The contour selection was based on an agreement with the program monitor in order that the data generated would be comparable to the results generated under the second technology contract, NAS9-12802.

The predicted operating characteristics for the columbium thrust chamber from the Task I studies included the following:

Design nominal maximum temperature (Insulated)	2400°F
Off limits (-10% P_c, +12% O/F) maximum temperature (insulated)	2548°F
Nominal barrier flow (% of total flow)	7.657
$I_{sp_{\infty}}$ (corrected to e = 74.3)	310.5 sec.

The Task VI demonstration chamber was different only in the reduced length (L*) of the chamber. The two inch length change allowed testing of a 26L* chamber. A stainless steel chamber spool was also designed to extend the 26L* chamber to the original 30 L* length for backup testing. In addition the spool was also used to extend the original chamber to produce a 34L* chamber volume.

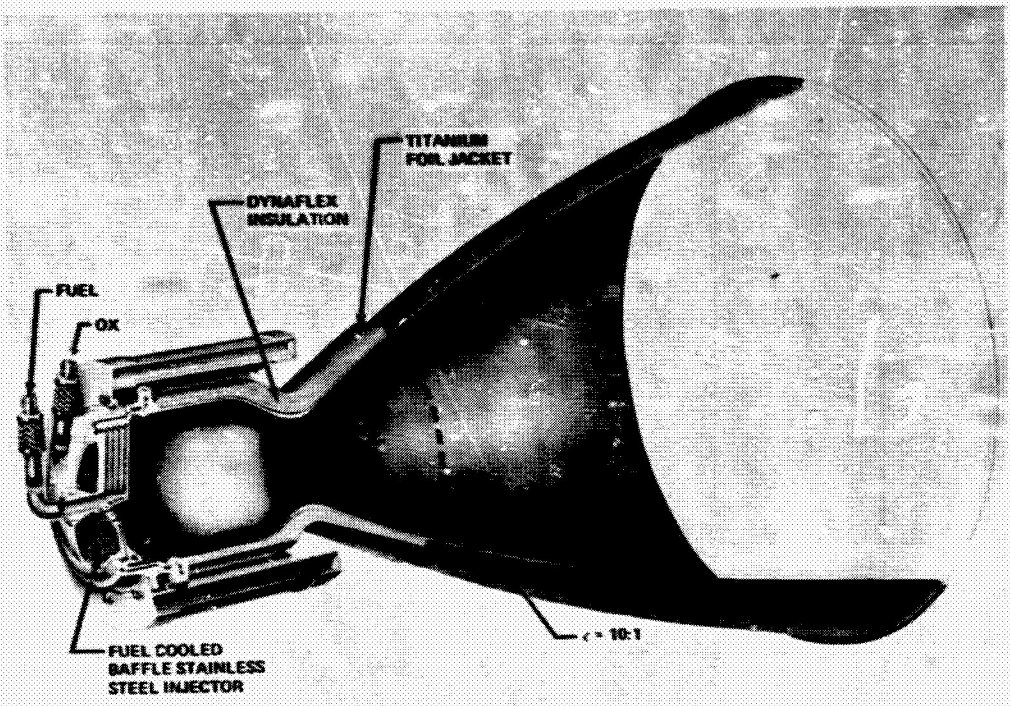

FIGURE IV-1. INSULATED COLUMBIUM THRUST CHAMBER ASSEMBLY

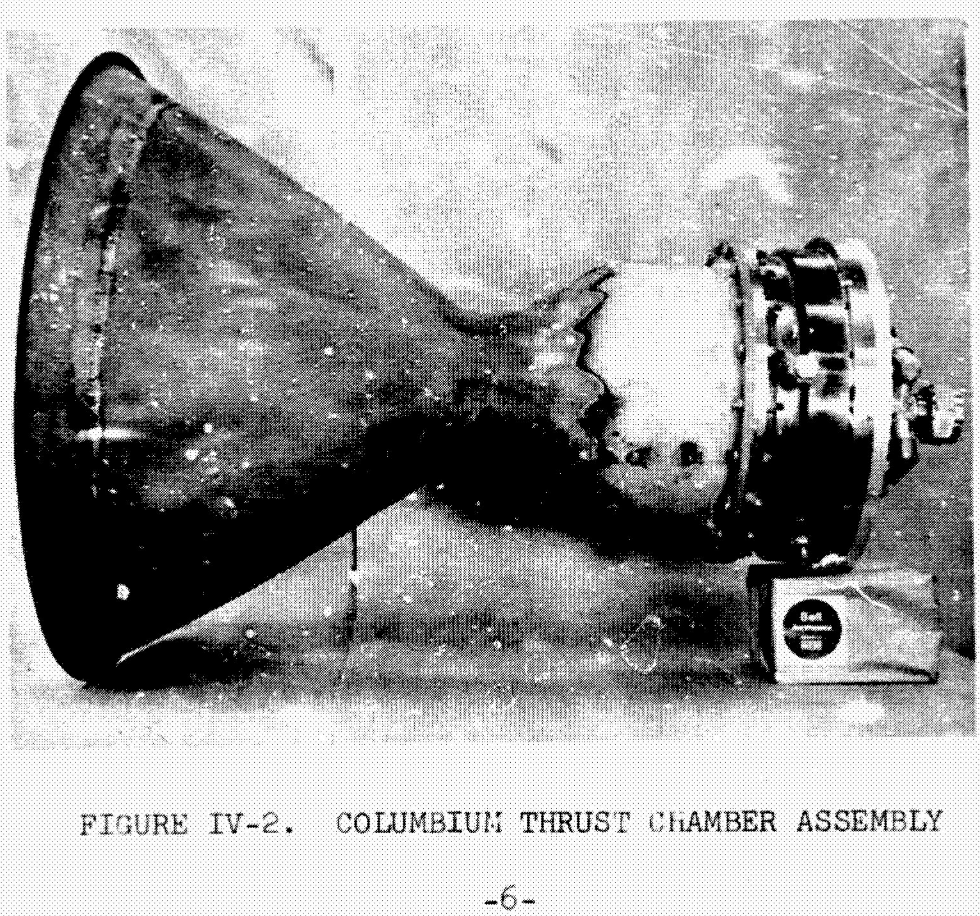

FIGURE IV-2. COLUMBIUM THRUST CHAMBER ASSEMBLY

2. DEMONSTRATION TESTING (TASK V AND VIII)

The initial columbium chamber testing (Task V) with the 30L* chamber was conducted in two phases. A Bell supplied five leg baffle, 304L stainless steel injector was used for the first test series. That injector was designed in accordance with the Task I and II OME definitions which included a 10 inch diameter chamber and injector. The injector incorporated triplet elements with an outer ring of unlike doublets.

The second test phase utilized a 6000 lbf, N_2O_4/MMH, 10 inch diameter injector, which was fabricated from 2219 aluminum and incorporated acoustic cavities for the suppression of high frequency combustion instability. Fuel vortex film cooling was maintained as the approach to the gas film temperature reduction. The triplet injector element was used exclusively. Aluminum was employed to expedite the injector fabrication, however, all injector internal hydraulic dimensions were flight configured for the longer life requirements of stainless steel.

The columbium chamber test program was formulated to provide engineering data as well as demonstrate chamber operation. To do this, the program was setup to initially test to temperature equilibrium without external insulation, and then demonstrate with insulation. All chamber tests were conducted in the Bell Test Center Altitude test facility. The test nozzle area ratio was limited to the facility (exhaust duct diameter available). The chamber operation was characterized over a ±10% range of chamber pressure and ±10% mixture ratio from the nominal values of P_c=125 psia and mixture ratio O/F=1.25. The initial tests were temperature monitored by two pyroscanners where these instruments were used to record temperature homogeneity and level. The final test demonstration was to monitor chamber temperature in insulated operation.

The Task VIII test effort examined L* and also compared MMH and A-50 as fuels. L* operation was examined from 26 to 34 L*.

C. REGENERATIVELY COOLED THRUST CHAMBER

The successful demonstration of the use of the triplet injector with hot fuel allowed the extension of testing with this injector to a regeneratively cooled thrust chamber. The original regeneratively cooled engine designed is shown in Figure IV-3. The regeneratively cooled channel wall test chamber furnished to the program and the task evaluation consisted of obtaining performance and heat rejection information. The test assembly (Figure IV-4) consisted of the S/N 2 aluminum injector

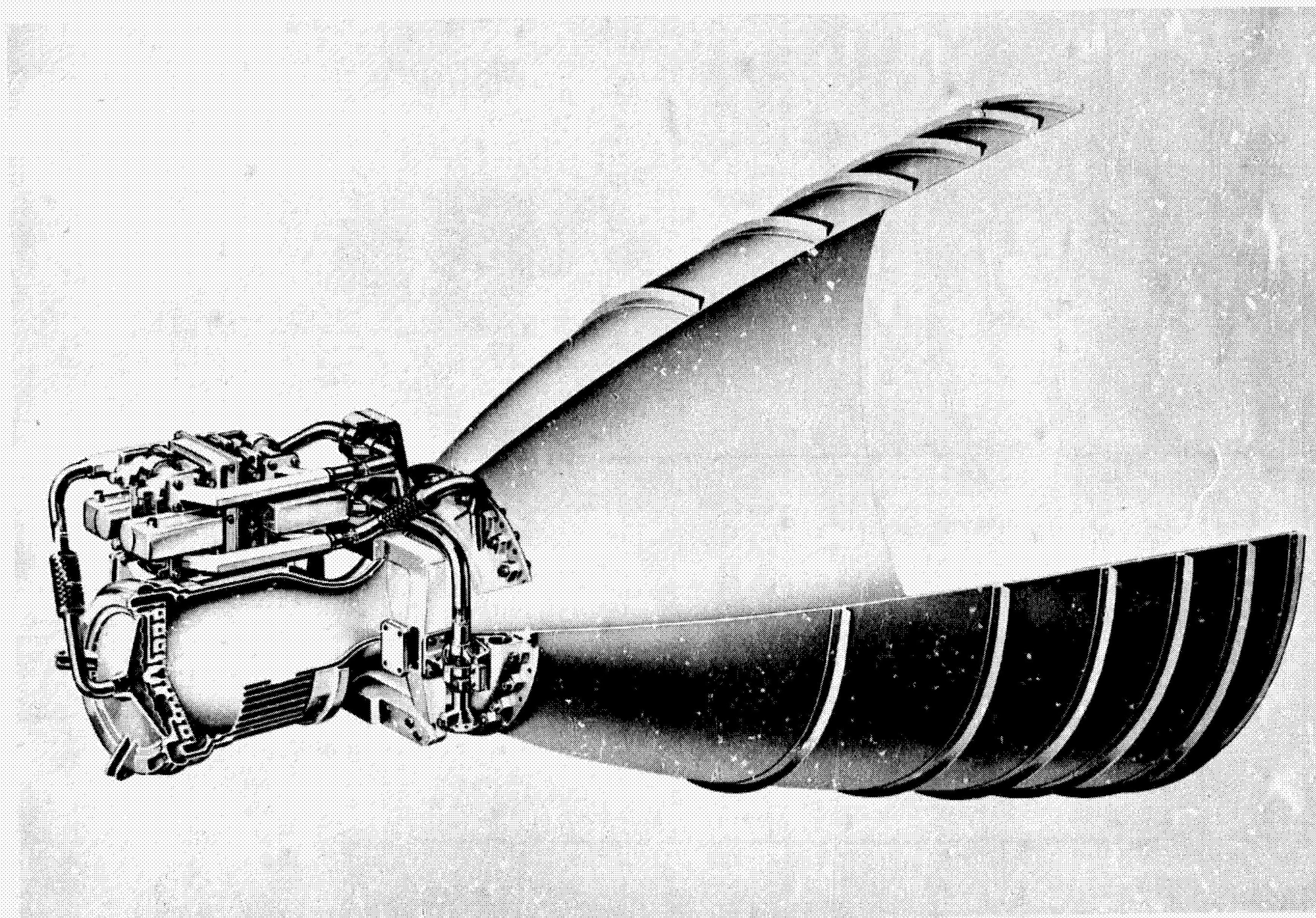

FIGURE IV-3. REGENERATIVE THRUST CHAMBER with its associated vortex cooling ring, the channel wall regenerative thrust chamber furnished by Bell, a columbium nozzle extension (to nozzle area ratio of 15) and a propellant valve adapted to the flow quantities for this engine. The nozzle area ratio was determined by the altitude test facility there testing of this assembly was conducted. This altitude test facility was previously rated at 3500# thrust and the throat section of the duct restricted the size of engine operation to the 15 to 1 nozzle area ratio used. The expense of modifying this facility to accept the full size OME nozzle was not considered to be necessary for the preliminary testing scheduled.

In addition to the initial testing of this engine at the Bell Test Center Facility (Task X), the test engine was modified to incorporate a full nozzle extension and tested at the NASA, White Sands Test Facility (WSTF), Figure IV-5. Data from both facilities were then compared and checked for both performance confirmation and facility differences.

D. HEAT TRANSFER TUBE TESTS

It has been known for years that propellant additives can be used to decrease the heat rejection to a combustion chamber wall. Bell experiments with amine fuels were conducted late in the 1950s and early 1960s where medium chamber pressure (500 psia) operation was demonstrated. The demonstration was to add silicone oil (Si) in small quantities to the UDMH or MMH and the resultant oxidized precipitants have been theorized to coat the combustor wall presenting a thermal resistant surface to heat flux. Consistant heat rejection reductions of from 30 to 40% have been measured. One system has recently been made operational.

Since a similar thermal reduction would also provide margins for a low pressure engine such as the Space Shuttle, interest into heat transfer specifics at lower chamber pressure were generated within the scope of the OME program. The evaluation in this case was to examine the possibility of cooling passage precipitants related to this lower chamber pressure operation.

The objective of the Task VII heated tube heat transfer test program was to determine, in an exploratory fashion, the effect of silicone additive to MMH and to 50-50. The program was conducted with and without silicone additive in both Hastelloy X and CRES 347 stainless steel, 1/8" O.D. tubes, with nominal velocities of 30 ft/sec., and 200 to 250 psia pressure. The test set up is shown in Figure IV-6. Generally, the tests were conducted by cycling up to and beyond the onset of nucleate boiling two or three times and then proceeding to the point of peak nucleate boiling with the associated tube burnout. Seven tubes were utilized, each of which were destroyed at the culmination of a test series.

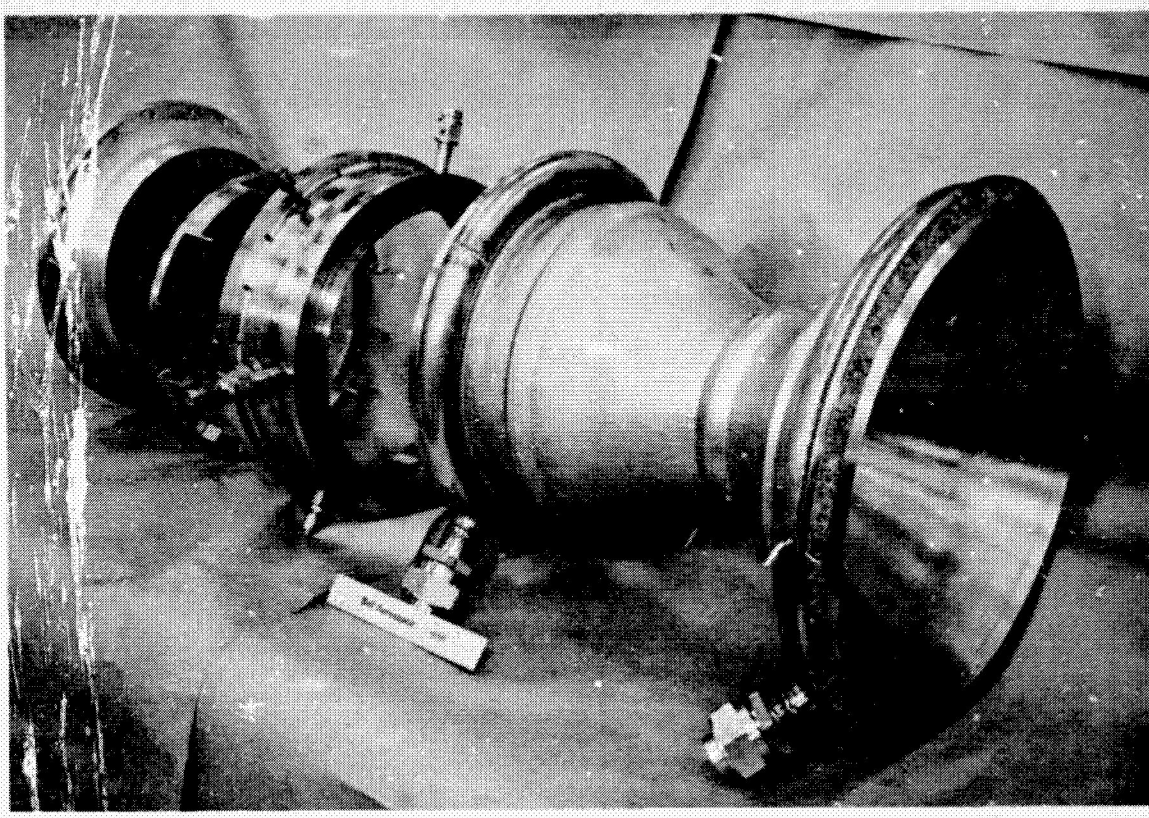

FIGURE IV-4. THRUST CHAMBER ASSEMBLY EXPLODED VIEW

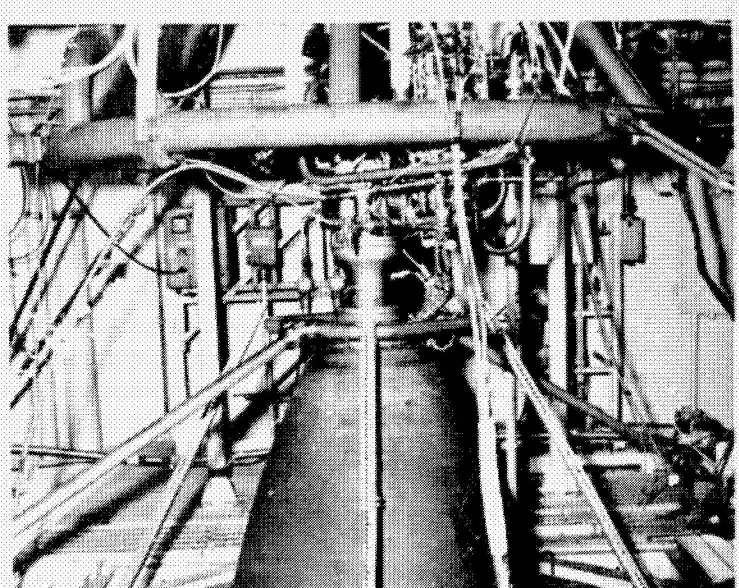

FIGURE IV-5. THRUST CHAMBER MOUNTED IN WSTF ALTITUDE TEST CELL AREA 401

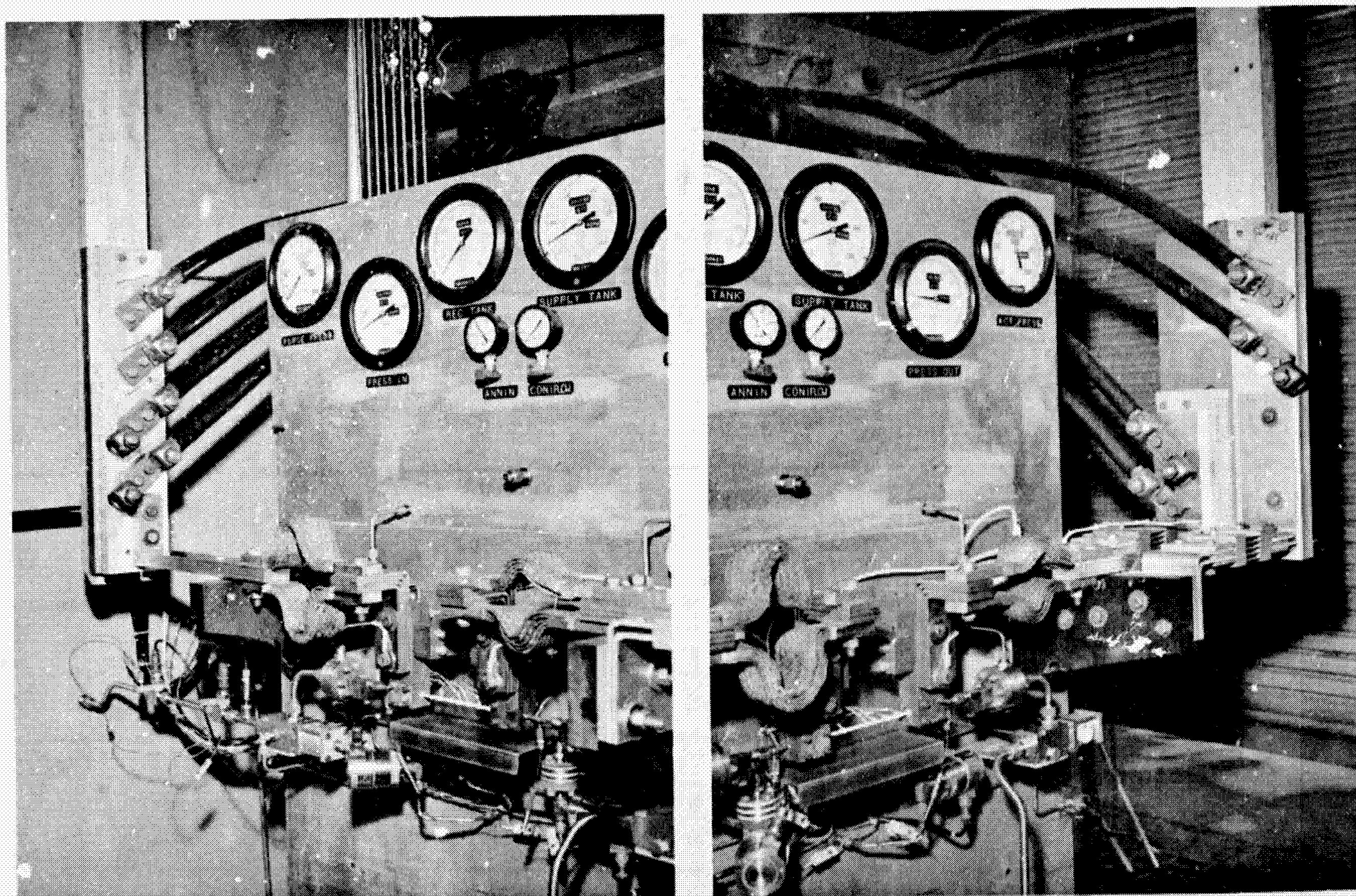

FIGURE IV-6. HEAT TRANSFER TEST PANEL

Test data examination entailed evaluation of thermal cycle differences on tube temperatures as an indication of some type of internal tube coating or deposit.

E. INJECTOR EVALUATION AND STABILITY TESTS (TASKS IX, XI, XII AND XIV)

Three injectors were used in support of the 10 inch diameter effort performed on this contract. The 2 fuel on 1 oxidizer triplet injection pattern was used on all three injectors, with the initial pattern based on previous experience and the latter patterns based on continued excellent results. Initially, substantial concern was displayed at the Bell selection of this pattern due to other negative contractor experience with conventional impinging pattern injectors. This negative experience was reported to be in the form of low performance, with substantially performance reduction as the fuel temperature was increased. No such performance loss was experienced with the Bell triplet injector, either at the system limits of operating conditions or with fuel temperatures at maximum anticipated values.

The initial regenerative operation evaluation for the injector simulated the fuel temperature increase expected on a regeneratively cooled thrust chamber. Task IX was performed to evaluate the injector with fuel supplied at temperatures to 250°F. Combustion stability tests were also performed in this task. The propellant requirement for these evaluation tests included a range of 200° to 250° for the fuel inlet temperature to the injector and an ambient simulated maximum temperature 100°F for the oxidizer.

The considerable success of the Task IX testing precipitated additional efforts with this type injector including the "sale" of one of the demonstration injectors to the program. Task XI "sold" the second flat face injector (Figure IV-7) allowing the company designed and developed unit to be modified as specified for other program tasks. Additional injector Tasks XII and XIV, added much more precise information on the combustion stability of the flat face injector equipped with an acoustic slot as the stabilizing device. Task XII was primarily concerned with examining the stability margin with reducing the slot open area where Task XIV examined the effect of the entrance of the slot on stabilization. Side efforts of these tasks was the development and use of bomb insertion mechanism and the development of a low cost bomb. The insertion device was used to reduce the thermal protection requirements on the bomb case and still be able to delay detonation until full combustor stabilization resulted. The low cost bomb used a molded cork case and plastic detonator to reduce both cost and shrapnel damage to the injector face.

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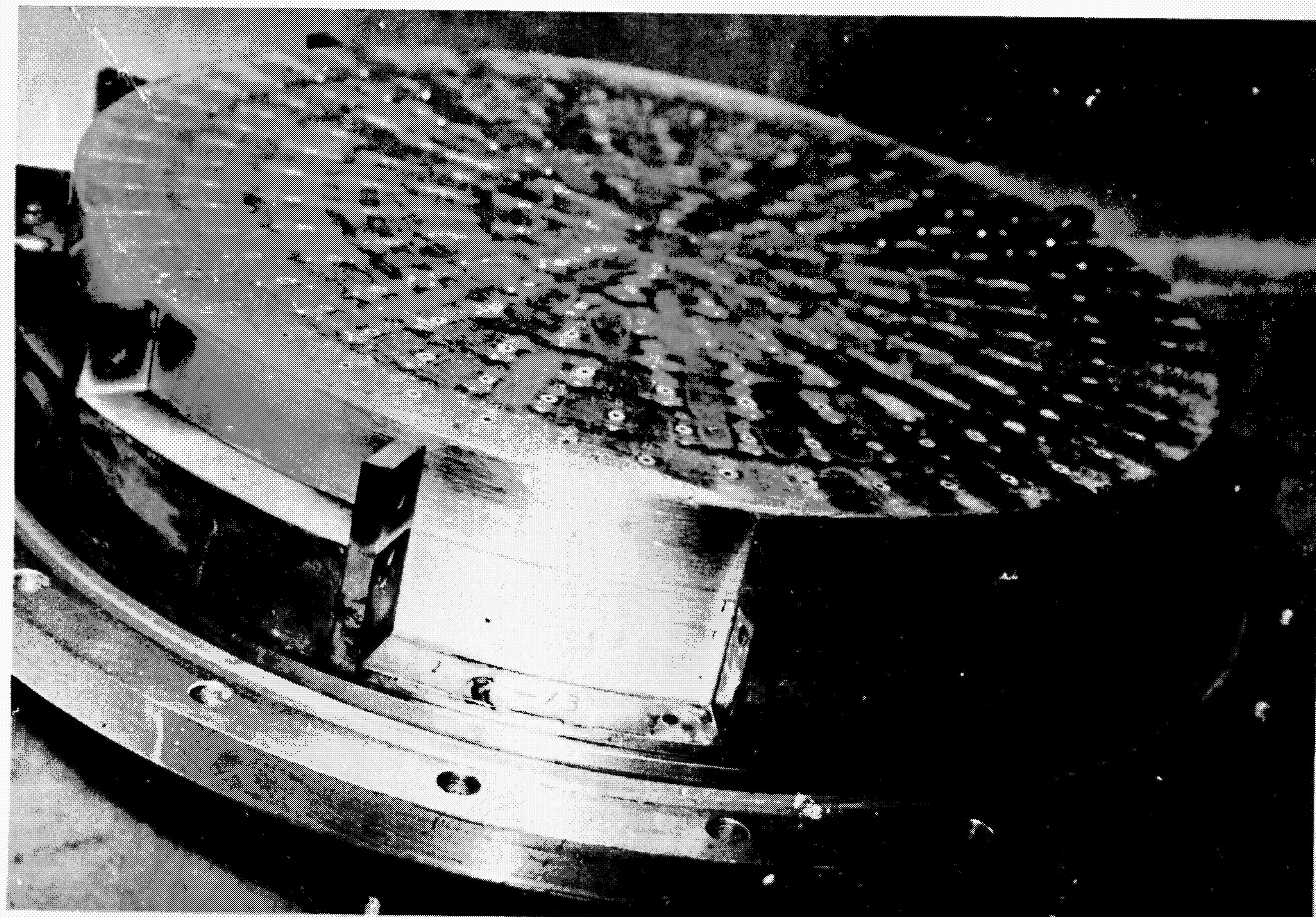

FIGURE IV-7. INJECTOR AND DAMPER RING (4 DEEP (1T) - 8 SHALLOW (3T) CAVITIES)

F. REDUCED DIAMETER COMBUSTOR (TASK XV)

During the course of studies leading to the OME engine definition, a 10 inch diameter combustion chamber was maintained as the least risk, most cost-effective approach to high performance. Other investigators elected smaller 8 inch diameters and required extended combustor lengths and/or pattern changes to achieve performance.

The 10 inch Bell design proved to be very satisfactory, and due to this success, an 8 inch injector triplet element injector design was undertaken. The objective of this task was to design, fabricate, and test an 8 inch diameter injector in an attempt to approximate the success of the larger 10 inch unit.

A small scale injector parameter trade-off study was conducted before the design release, primarily to examine the capability of compressing the backside propellant distribution manifolds without compromising the orifice entrance locations. In addition to the performance design trade-off studies conducted, an additional requirement of the injector design was to fit the interface of the 8 inch chamber interface design supplied by NASA-JSC. The injector evaluation testing included performance, heat flux, and stability data utilizing a combination of new and existing hardware. A new injector and acoustic cavity adapter/acoustic cavity ring were fabricated with the acoustic cavity made adjustable to allow stability configuration testing. To minimize the cost for test chamber hardware, the 10 inch diameter water cooled nozzle was used, and a new 8 inch diameter steel bomb chamber fabricated (Figure IV-8).

V. BASIC DATA GENERATED AND SIGNIFICANT RESULTS

The most significant results of this program was the demonstration of two substantially different thrust chamber designs, each of which could perform the OME mission. The initial thrust chamber tested the insulated columbium thrust chamber, represents a significant departure from convention and offers proof of feasibility of both the concept and practical operation. The more conventional channel wall regeneratively cooled chamber also was demonstrated to offer high performance as well as statistically and reasonable cooling margins. The concern in accomplishing adequate performance with the regeneratively cooled chamber came about when competing designs were found to be performance sensitive to fuel temperature. This sensitivity was not found using the Bell triplet injector design and led to the further testing for combustion stability at Bell and altitude performance tests at WSTF as proof of concept demonstrations. Subsequent testing showed the acoustic damper combustion stability devices to be very effective, performance tests demonstrated 317 seconds specific impulse with a properly shaped OME sized nozzle.

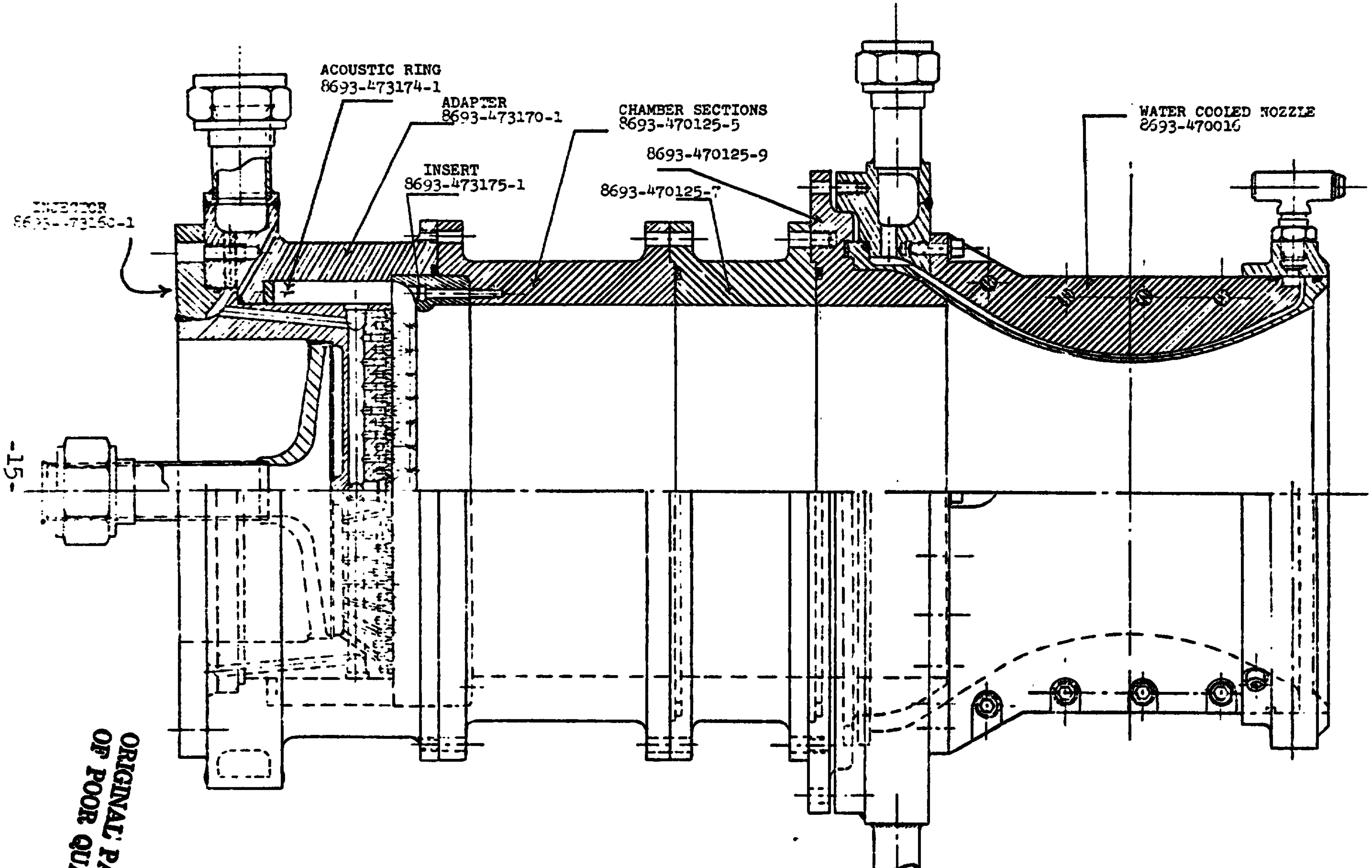

FIGURE IV-8

TEST HARDWARE ASSEMBLY
(8 INCH DIAMETER INJECTOR)

The final task of this program was the design, fabrication, and demonstration of an 8.2 inch diameter triplet element injector which was compatible with the size of competing thrust chambers. The testing conducted with the 8.2 inch injector confirmed the design and the sea level combustion efficiency was measured at a value which would produce 316 seconds I_{sp} on an OME engine. The limited stability testing conducted was also positive, producing a design with substantial stability margin and fulfilling all of the OME requirements. A summary of the demonstration tests performed within the program is as follows:

PROGRAM TEST SUMMARY (Number of Tests Conducted)				
Injector	SS 1 Stainless Steel With Baffle (10" Dia.)	Al 1 Aluminum Flat Face With Damper (10" Dia.)	Al 2 Aluminum Flat Face With Damper (10" Dia.)	Al 3 Aluminum Flat Face With Damper (8" Dia.)
Type of Testing				
Injector Testing Performance and Heat Rejection	18	42	18	11
Injector Stability (Bomb) Tests	5		60	
Columbium Chamber Demonstration and Evaluation (Altitude)				
Regenerative Chamber Demonstration and Evaluation (Altitude)			65*	
*Includes 47 tests at the NASA, WSTF				

A more detailed summary of the efforts and significant results accomplished in the primary component efforts is discussed in the following pages.

A. REUSABLE OME THRUST CHAMBER PARAMETRIC STUDIES

The full account of the parametric studies conducted in these tasks including assumptions, efforts and results has been detailed in Report No. 8693-953006 entitled "Space Shuttle Orbit Maneuvering Engine Reusable Thrust Chamber Program Parametric Engine Data Report". This report details the steps taken to develop the ADEPT computor program as well as to provide the results of the use of the program to produce data for the OME application. A sample computer program diagram is included in Figure V-1 to show the program configuration for the studies.

The programs included information for both Tasks I and II where Task I examined N_2O_4 oxidizer and Task II examined LOX as the oxidizer. The program was used to produce data for preliminary technical data for the various engine and propellant alternatives where separate ratings and recommendations were made to consider such aspects as cost.

1. PREDICTED ENGINE DATA

Nominal engine data provided by the ADEPT program is presented in Tables V-1 and V-2. These tables summarize the point designs made from the programs and indicate the available comparative values resulting from the computor programs. The term +Si, used in these tables, represents the use of silicone oil to lower the heat rejection, and subsequently the required wall cooling. An example of such benefits is the use of the silicone oil additive which improves the regen engine I_{sp} by 3 seconds at the expense of about 20 lbs higher engine weight. The Si additive nearly eliminates the requirement for chamber film coolant but does not effect nozzle extension heat rejection. The increased nozzle temperature thus requires a larger area ratio chamber/nozzle joint and heavier chamber.

One further result of this study was the increased predicted performance of the LOX system. The LOX/Amine engines of Table V-2 show an I_{sp} improvement of about 20 seconds relative to N_2O_4/Amine at the lower area ratios resulting from the reduced nominal P_c definition. The LOX/C_3H_8 and LOX/RP-1 I_{cb} engines' performance show little or no advantage relative to LOX/MMH. The weight of the LOX regen engines is 25 lbs higher than the N_2O_4 designs due to the ignition system and slightly larger chamber diameters for the lower nominal chamber pressure. The weight difference for the LOX I_{cb} engines is about 40 lbs above the values predicted for N_2O_4/Amine operation.

2. TECHNICAL RATING

The technical rating factors for the Task I and II chamber cooling concepts and propellant combinations are shown in Table V-3. The listing of factors attempted to cover all operating and non-operating characteristics of the engines under the headings of

Complexity
Service and Maintenance
Fabricability
Start
Steady State Operation
Design Life

ENGINE COMPUTER PROGRAM – GENERAL – REGENERATIVE COOLING

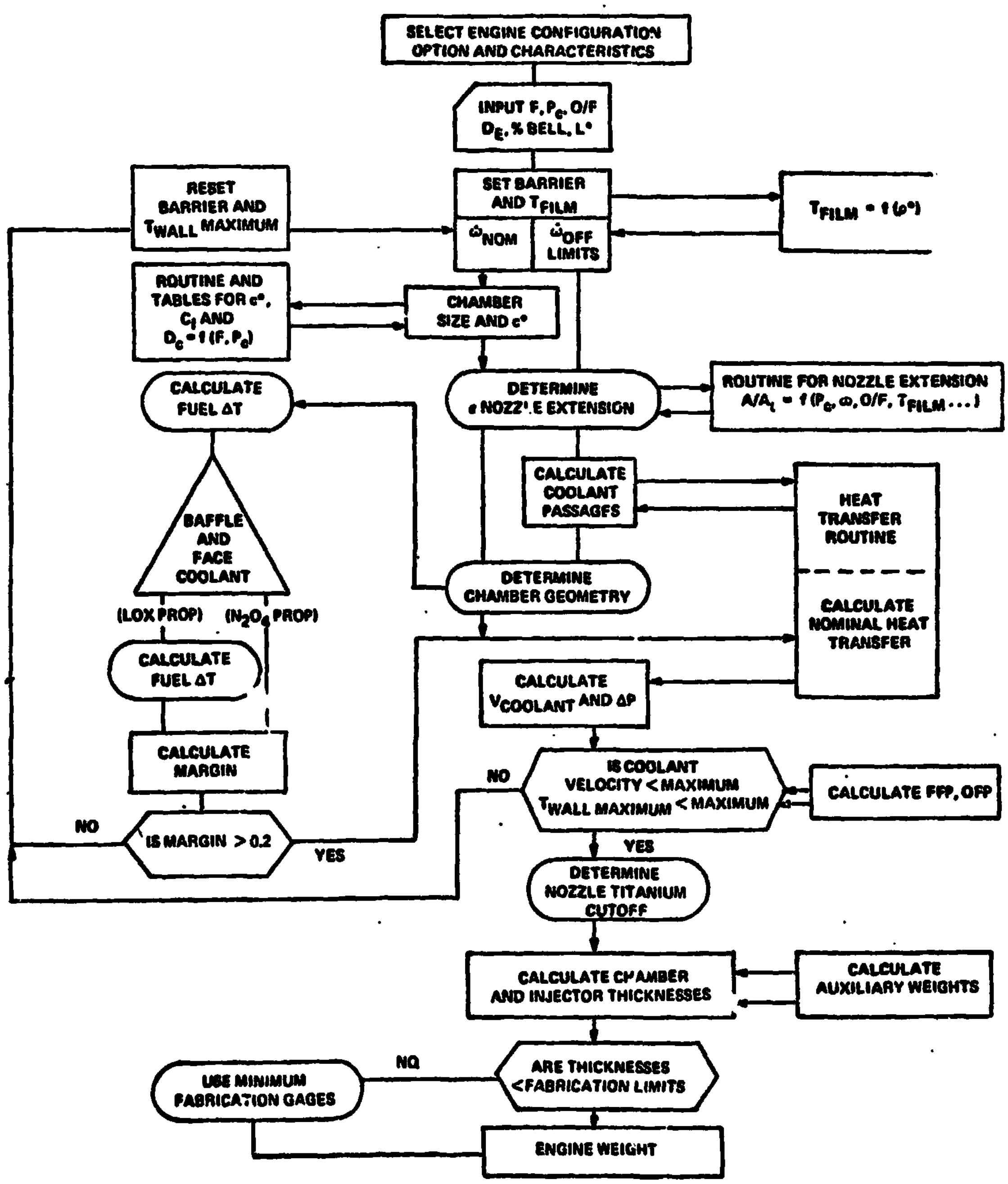

FIGURE V-1

TABLE V-1

TASK I
NOMINAL ENGINE DATA
P_C = 125 PSIA, DEX = 50, 80% BELL

PROPELLANTS	N_2O_4/MMH				N_2O_4/MMH + Si		N_2O_4/50-50		
CHAMBER COOLING CONCEPT	CHANNEL WALL REGEN	DRILLED AL REGEN	INSULATED COLUMBIUM	TUBE WALL REGEN	CHANNEL WALL REGEN	DRILLED AL REGEN	CHANNEL WALL REGEN	DRILLED AL REGEN	INSULATED COLUMBIUM
I_{sp_∞}, SEC	316.40	316.2	310.50	316.40	319.30	319.30	315.80	315.80	308.60
O/F	1.64 →						1.60 →		
AREA RATIO	74.33 →						74.65 →		
FFP, PSIA	205.30	228.90	183.70	203.20	203.8[illegible]	21[illegible].90	207.30	251.20	183.80
OFP, PSIA	190.60	190.70	184.60	190.60	190.60	190.60	190.60	190.60	184.60
WEIGHT, LB, TOTAL	1[illegible]4.80	183.70	191.40	208.68	216.30	201.70	194.20	184.50	191.30
LT, IN.	80.80	80.[illegible]	80.80			80.89	80.80		
D_{MAX}, IN.	66.73 →								
T_{MAX} OFF LIMITS, °F	-	-	2548	-	-	-	-	-	2513
BARRIER FLOW, %	3.59	3.73	7.65	3.32	0.01	0.01	4.69	4.68	8.65
EXTENSION A/AT	7.91	7.72		7.44	12.55	12.55	7.67	7.70	

Propellant Feed Temperature = 70°F

*Based on ±7° Gimbal Pitch and Yaw

TABLE V-2

TASK II
NOMINAL ENGINE DATA
P_C = 100 PSIA, DEX = 50, 80% BELL

PROPELLANT COOL CONCEPT	LOX/MMH			LOX/MMH + Si		LOX/50-50			LOX/C_3H_8	LOX/N_2H_4	LOX/RP-1
	CHANNEL WALL	DRILL AL	INSULATED COLUMBIUM	CHANNEL WALL	DRILL AL	CHANNEL WALL	DRILL AL	INSULATED COLUMBIUM	INSULATED COLUMBIUM	INSULATED COLUMBIUM	CHANNEL WALL
I_{sp_∞}, SEC.	333.30	333.20	329.60	336.0	336.0	335.0	335.7	327.20	315.9	331.2	332.3
O/F	1.0	1.0	1.10	1.0	1.0	1.0	1.0	1.00	2.30	0.70	2.40
AREA RATIO	58.96	58.96	54.54	58.96	58.96	59.40	59.40	59.34	60.15	59.25	61.23
FFP, PSIA	187.70	206.70	159.30	188.00	207.20	187.70	205.50	159.6	16[illegible]	159.90	181.00
OFP, PSIA	159.80	159.80	159.70	159.80	159.80	159.80	159.80	159.8	[illegible]	159.80	159.60
WEIGHT, TOTAL	220.70	209.60	234.20	248.90	233.70	225.20	214.90	227.20	2[illegible]9.00	228.20	243.50
LT, IN.	81.3 →									81.3	84.9
D_{MAX}*	66.5 →										
BARRIER FLOW, %	4.97	5.06	8.51	1.83	1.12	4.63	4.55	9.98	5.90	10.02	0.02
EXT. A/At	4.75	4.59	-	[illegible]84	13.73	5.66	5.66	-	-	-	8.19

FUEL FEED TEMPERATURE - 70°F
*±7% GIMBAL PITCH AND YAW

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Each sub-heading considered was given a numerical rating of 1 to 4 as defined by Table V-3. In addition certain sub-headings were weighed by repeating the numerical value. The weighing of 3 recognized the potential cost impact to the engine development program for problems resulting from the higher technical risk.

The "perfect" engine would have a total numerical score of 165 for the definitions of Table V-3. The accumulative results of these ratings was included in Table V-4. It may be noted that the ranking definitions reflected engine technology at the time of the rating and sought to include all areas of which presented potential development problems.

3. COST COMPARISONS

The engine's cost comparisons are presented in Table V-5. The I_{sp}, feed pressure and engine weight Δ costs were developed from the McDonnell-Douglas OMS trade studies on the basis of the nominal engine design data.

Other information used in the cost projection included feed pressure changes with propellants; baseline engine cost of $25,000,000 (insulated columbium) x 1.5 for NASA cost; "$Δ tech rating to reflect development costs; technical rating of 3 raised the development base of $825,000 to $2,475,000 in high risk elements.

The last element of [illegible] cost was the estimated difference to the OMS for the Task II propellants. The 10 million dollars is primarily associated with the development and qualification costs of the insulated LOX tank.

4. RECOMMENDATIONS

The recommendation for the OME reusable thrust chamber and propellant combination based on the Task I and II effort resulted from the technical and cost ratings. The four "best" approaches were:

Technical Rating			Cost Rating ($\times 10^{-6}$)		
135.5	Icb	N_2O_4/Amine	-3.31	Icb	LOX/MMH
118.8	DAR	N_2O_4/Amine	-1.13	DAR	N_2O_4/MMH + Si
110.5	Icb	LOX/MMH	0.00	Icb	N_2O_4/MMH
108.3	CWR	N_2O_4/Amine	+1.10	CWR	N_2O_4/MMH + Si

The apparent high technical rating of the Icb with N_2O_4/Amine propellant appeared to offset the small cost advantages of the LOX/MMH propellant combination with the Icb chamber and the DAR design with N_2O_4/MMH + Si. Therefore, the insulated columbium thrust chamber engine with N_2O_4/MMH propellants was reported as the best choice based on the Task I [illegible] studies and rates.

TABLE V-3

TECHNICAL RATING FACTORS TASK I AND TASK II

WEIGHT		4	3	2	1
	A. COMPLEXITY				
1	1. NUMBER OF MECHANICAL JOINTS	0	1	2	> 2
1	2. NUMBER OF METAL TO METAL JOINTS, CH	0 - 1	2 - 3	4-6	> 6
1	3. COATINGS, OXIDATION RESISTANCE	NONE	NOZZLE EXTENSION	CHAMBER AND NOZZLE EXT.	TCA AND COOLING PASSAGES
1	4. INSULATION REQUIREMENTS	NONE	VALVE	COMBUSTION CHAMBER	VALVES AND C.C.
1	5. START AUXILIARY CONTROLS	NONE	AUXILIARY VALVES	AUX. VALVES AND SEQUENCE	AUX. VALVES SEQ. AND SENSING
1	6. FEED SYSTEM MALFUNCTION DETECTION	CHAMBER PRESSURE	P_c AND TEMP.	P_c, TEMP $\frac{\dot{w}_o}{\dot{w}_f}$ →	PLUS OTHER
	B. SERVICING AND MAINTENANCE				
1	1. COMPATIBILITY PROPELLANT, FLUIDS	COMPAT DEMONSTRATED	POTENTIAL LONG TERM PROB.	REQUIRES PURGE	REQUIRES PURGE AND FLUSH
1	2. INSPECTIBILITY, SURFACE, VISUAL	EXCELLENT	GOOD	FAIR	NOT POSSIBLE
1	3. CHAMBER COMPLEXITY (ADD ITEMS A1-A4	16	12	8	4
1	4. START AUXILIARY CONTROLS, (A5)	NONE	AUX. VALVES	AUX. VALVES AND SEQ.	AUX. VALVES SEQ. AND SENSING
1	5. FEED SYSTEM MALFUNCTION DET. (A6)	CHAMBER PRESSURE	P_c AND TEMP	P_c TEMP $\frac{\dot{w}_o}{\dot{w}_f}$ →	PLUS OTHER
	C. FABRICABILITY (CHAMBER)				
1	1. JOINING TECHNIQUES	WELD	+ BRAZE OR E.D.	+ BRAZE AND E.D.	DIFFUSION + BOND
1	2. FABRICATION METHODS (CHAMBER)	MACHINING	PLUS EDM	PLUS SPINNING OR E.D	PLUS OTHER
1	3. INSPECTIBILITY, NDT	EXCELLENT	GOOD	FAIR	POOR
1	4. PROCESS CONTROLS	DEFINED	DEFINED FOR SMALLER	DEVELOPMENT REQUIRED	NEW PROCESS
1	5. TOOLING REQUIRED	NOMINAL	LOW COST RE-USABLE	LOW COST	HIGH COST
	D. START				
1	1. TYPE	HYPERGOLIC SPACE DEMO.	NON HYPER. SPACE DEMO	NON HYPER SL DEMO	NON HYPER NOT DEMOD
1	2. POTENTIAL RESTART RESTRICTIONS	NONE	< 5 MIN	5-15 MIN	> 15 MIN
1	3. AUXILIARY CONTROLS, A5	NONE	AUX VALVES	AUX. VALVES AND SEQ	AUX. VALVES SEQ AND SENSING
1	4. MALFUNCTION SENSING (TIME TO P_c)	REPRODUCIBLE START TRANS.	CLOSELY REPRODUCIBLE	FAIRLY REPRODUCIBLE	ALT START TRANS. NOT DEFINED
	E. STEADY STATE OPERATION				
3	1. CHAMBER COOLING EFFECT ON INJECTION	NONE	HEATED PROP. DEMOD	HEATED FUEL NOT DEMOD	HEATED FUEL AND OX NOT DEMOD
1	2. FEED SYSTEM MALFUNCTION DETECT, SENSITIVITY	NOT SENSITIVE	> 1 SEC.	< 1 SEC	≪ 1 SEC
1	3. FEED SYSTEM MALFUNCTON DETECT, COMPLEXITY (A6)	CHAMBER PRESSURE	P_c AND TEMP	P_c, TEMP $\frac{\dot{w}_o}{\dot{w}_f}$ →	PLUS OTHER
1	4. SENSITIVITY TO DESIGN POINT, I_{sp}	NOT SENSITIVE	< 1% LOSS	> 1 < 2% LOSS	> 2% LOSS
3	5. CONFIDENCE IN PREDICTED I_{sp} BASED ON COOLING REQ'D	± 0	± 1 %	± 2%	> ± 2%
1	6. DESIGN MARGIN, OFF LIMITS, CHAMBER	0.6	0.4	0.2	0
1	7. DESIGN MARGIN, OFF LIMITS, EXTENSION	> 0.6	> 0.4	> 0.2	> 0
3	8. DESIGN MARGIN, OFF LIMITS, INJECTOR	0.6	0.4	0.2	0
	F. DESIGN LIFE				
1	1. CHAMBER COATINGS	NOT APPLICABLE	DEMOD AT OPERATING COND	DEMOD AT TEMP	PARTIAL DEMOD
1	2. CHAMBER MATERIAL STRUCTURAL DESIGN CONFIDENCE	DESIGN TECH AND PROPERTIES AVAILABLE	EXTRAPOLATED MATERIAL PROPERTIES	POTENTIAL PROPERTY DEFINITION PROBLEM	CRITERIA AND PROPERTIES NOT DEFINABLE
1	3. GAS EMBRITTLEMENT, CHAMBER	LIFE DEMOD	NO INDICATION TESTING TO DATE	POTENTIAL PROBLEM	KNOWN PROBLEM
1	4. GAS EMBRITTLEMENT, NOZZLE EXTENSION	LIFE DEMOD	NO INDICATION, TESTING TO DATE	POTENTIAL PROBLEM	KNOWN PROBLEM
2	5. DEMONSTRATION REQUIRED	AVAIL DATA AND NOW FIRING	PLUS LIMITED FULL SCALE	EXTRAPOLATE FULL SCALE PLUS SUBSCALE	FULL SCALE TCA ONLY FULL LIFE FIRINGS

○ REPEATED FACTORS AND WEIGHT OF "START AUX CONTROLS" "FEED SYSTEM MALFUNCTION DETECTION" IS 3

E.D.M. - ELECTRODEPOSITED MACHINING

E.D. - ELECTRODEPOSITED

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TABLE V-4

ENGINE RATINGS

OXIDIZER / FUEL	N_2O_4 MMH				N_2O_4 MMH + Si		N_2O_4 50-50			LOX MMH			LOX MMH + Si		LOX RP1		LOX C_3H_8	LOX N_2H_4
CONCEPT	CWR	DAR	TWR	ICb	CWR	DAR	CWR	DAR	ICb	CWR	DAR	ICb	CWR	DAR	CWR	DAR	ICb	ICb
FACTOR																		
A 1	3	2	3	3	3	2	3	2	3	3	2	3	3	2	3	2	3	3
2	1	2	1	3	1	2	1	2	3	1	2	3	1	2	1	2	3	3
3	3	3	3	2	3	3	3	3	2	3	3	2	3	3	3	3	2	2
4	4	4	4	2	4	4	4	4	2	3	3	2	3	3	3	3	2	2
5	4	4	4	4	4	4	4	4	4	1	1	1	1	1	1	1	1	1
6	2	2	2	3	2	2	2	2	3	2	2	3	2	2	2	2	3	3
B 1	2.5	1	3	4	2.5	1	2.5	1	4	2.5	1	4	4	1	2.5	1	4	4
2	1	1	1	4	1	1	1	1	4	1	1	4	1	1	1	1	4	4
3	2.8	2.8	2.8	2.5	2.8	2.8	2.8	2.8	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5
4	4	4	4	4	4	4	4	4	4	1	1	1	1	1	1	1	1	1
5	2	2	2	3	2	2	2	2	3	2	2	3	2	2	2	2	3	3
C 1	3	4	2	4	3	4	3	4	4	3	4	4	3	4	3	4	4	4
2	2	4	2	2	2	4	2	4	2	2	4	2	2	4	2	4	2	2
3	3	3	2	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3
4	3	4	2	3	3	4	3	4	3	3	4	3	3	4	3	4	3	3
5	2	4	1	3	2	4	2	4	3	2	4	3	2	4	2	4	3	3
D 1 (x3)	12	12	12	12	12	12	12	12	12	3	3	3	3	3	3	3	3	3
2	2	2	2	4	2	2	2	2	4	2	2	4	2	2	2	2	4	4
3	4	4	4	4	4	4	4	4	4	1	1	1	1	1	1	1	1	1
4	3	3	3	4	3	3	3	3	4	1	1	1	1	1	1	1	1	1
E 1 (x3)	3	3	3	12	3	3	3	3	12	6	6	12	6	6	6	6	12	3
2	1	2	1	3	1	2	1	2	3	1	2	3	1	2	1	2	3	3
3	2	2	2	3	2	2	2	2	3	2	2	3	2	2	2	2	3	3
4	3	3	3	3	3	3	3	3	3	2	2	2	2	2	2	2	1	1
5 (x3)	9	9	9	9	9	9	9	9	9	6	6	6	6	6	6	6	6	3
6	2	2	2	4	2	2	2	2	4	2	2	4	2	2	2	2	4	4
7	3	3	3	4	3	3	3	3	4	3	3	4	3	3	3	3	4	4
8 (x3)	6	6	6	9	6	6	6	6	9	6	6	9	6	6	6	6	3	3
F 1	4	3	4	2	4	3	4	3	2	4	3	2	4	3	4	3	2	2
2	4	4	4	3	4	4	4	4	3	4	4	3	4	4	4	4	3	3
3	3	3	3	2	3	3	3	3	2	3	3	2	3	3	3	3	2	2
4	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2
5 (x3)	3	9	3	6	3	9	3	9	6	3	9	6	3	9	3	9	6	3
TOTALS	108.3	118.8	10-.8	135.5	108.3	118.8	108.3	118.8	135.5	86.0	96.5	110.5	87.5	96.5	86.0	96.5	103.5	89.5

TABLE V-5

COST COMPARISONS x 10^6

OX / FUEL	N_2O_4 MMH				N_2O_4 MMH + Si		N_2O_4 50-50			LOX MMH			LOX MMH + Si		LOX RP1	LOX C_3H_8	LOX N_2H_4
CONCEPT	ICb	DAR	CWR	TWR	CWR	DAR	CWR	DAR	ICb	CWR	DAR	ICb	CWR	DAR	CWR	ICb	ICb
\$, Δ I_{sp}	0	-8.16	-8.45	-8.45	-12.60	-12.6	-7.59	-7.59	+2.72	-27.25	-27.13	-22.82	-30.47	-30.47	-30.05	-6.65	-24.74
\$, Δ FEED PRESSURE	0	+7.23	+3.89	+3.60	+3.68	+5.95	+4.17	+10.36	+0.01	+ 5.04	+ 8.41	0	+ 5.10	+ 8.50	+ 3.97	0.392	+ 0.14
\$, Δ ENGINE WEIGHT	0	-0.18	+0.08	+0.40	+0.60	+0.25	0.067	0.186	0	0.70	0.44	1.03	1.38	1.01	1.25	1.142	0.88
\$, Δ TECH RATING (1)	0	+0.23	+1.39	+11.0	+1.39	+0.53	9.42	5.27	0	21.6	15.2	7.95	20.6	1[illegible].2	21.6	11.6	19.2
\$, Δ OMS (2)	0	0	0	0	0	0	0	0	0	10	10	10	10	10	10	10	10
SUM	0	- .88	-3.09	+ 6.56	+6.93	-6.26	+6.07	+7.87	+2.73	+10.09	+ 6.92	- 3.84	+ 6.61	+ 4.24	+ 6.77	+16.48	+ 5.48

CONVERSIONS	OX / FUEL	N_2O_4 AMINE	LOX AMINE	LOX C_3H_8	LOX RP1
\$/Δ I_{sp}		1.432	1.195	1.232	1.318
\$/Δ PSI		0.141	0.177	0.196	0.184
\$/Δ WEIGHT		0.024	0.024	0.024	0.024

McDONNELL DOUGLAS
OMS TRADE STUDIES
PROGRESS REPORT NO. 1
15 AUGUST 1972

(1) $1.5\left[\frac{135.5}{RATING} - 1\right]$ 85 FACTOR OF 1.5 ESTIMATED CONVERSION DAO PRICE TO TOTAL PROGRAM COST

(2) Δ OMS COST ASSUMED AT \$10 x 10^6 FOR LOX PROPELLANT COMBINATIONS

Subsequent effort under contract NAS 9-12803, showed that both the I_{cb} and CWR were capable of achieving their predicted performance with N_2O_4/MMH. The I_{cb} design also met its nominal temperature definition of 2400°F. The heat rejection to the MMH regen coolant was shown to be significantly less than predicted providing higher thermal margins and allowing the use of an uncoated Haynes 25 nozzle extension in place of the coated columbium extension. A flat face injector with acoustic cavities was shown to provide damping for bomb induced combustion disturbances. The successful bomb testing showed that the Task I and II definition of an injector with a 5 leg baffle could be changed to an injector with no baffle and incorporating the acoustic cavities. Finally, analyses of the feed system gas ingestion malfunction showed that engine damage could be prevented by chamber pressure sensing only. These changes allowed the values of Table V-4 to become: A6, B5, E3 = 4; E1=12 N_2O_4 = 9 LOX; E2 Regen = 3; E5 Regen N_2O_4 = 12; LOX Regen = 9; E6 Regen = 3; E8 Regen = 9; E5 CRW = 6.

The result of the above changes on the ratings for the 6 "best" are as follows:

Technical Rating			Cost Rating (X 10^{-6})		
142.5	I_{cb}	N_2O_4/Amine	-6.93	CWR	N_2O_4/MMH + Si
141.8	DAR	N_2O_4/Amine	-6.26	DAR	N_2O_4/MMH + Si
137.3	CWR	N_2O_4/Amine	-3.84	I_{cb}	LOX/MMH
117.5	Icb	LOX/MMH	-3.09	CWR	N_2O_4/MMH
113.5	DAR	LOX/Amine	-0.88	DAR	N_2O_4/MMH
109.0	CWR	LOX/Amine	0.00	Icb	N_2O_4/MMH

The technical ratings of all engines increase. The difference in technical rating between Icb and the regen chambers and between the DAR and CWR is reduced. The relative rating between N_2O_4 oxidizer and LOX oxidizer engines increases for the regeneratively cooled chambers. Therefore, with little technical rating difference between the various chambers the cost ratings would recommend either CWR and DAR with N_2O_4 and MMH with silicone oil additive. Prohibition of MMH + Si and LOX in order to have common propellants for the OMS and RCS makes CWR with N_2O_4/MMH the overall choice.

In conclusion, the large technology base established by the remainder of the program changes the recommended type of thrust chamber from I_{cb} to CWR and confirms the selection of N_2O_4/MMH as the OME propellant combination.

B. INSULATED COLUMBIUM THRUST CHAMBER

1. DESIGN

The columbium thrust chamber is a very simple design with the principal component being the columbium combustor shell. Past experience has allowed for operation of such combustor shells in either open (radiation cooled) or buried (insulated) installations. The thrust chamber designed for the OME application was a buried installation with all components fabricated from readily available materials and techniques. The simplicity of the columbium combustor shell is readily illustrated in Figure V-2.

The simplicity of the columbium thrust chamber shell readily promotes fabricability, however, it has been found that this simplicity also promotes reluctance of acceptability. The reluctance is promoted due to the thin shell which is indirectly considered sensitive to injector streaking. Secondly, oxidation without a coating can be a problem and therefore the coating tenacity and completeness is a requirement. Fortunately, neither of the above "problems" have been realistically encountered. However, until more actual experience is gained through the industry, the reluctance to accept this simple thruster concept will probably remain.

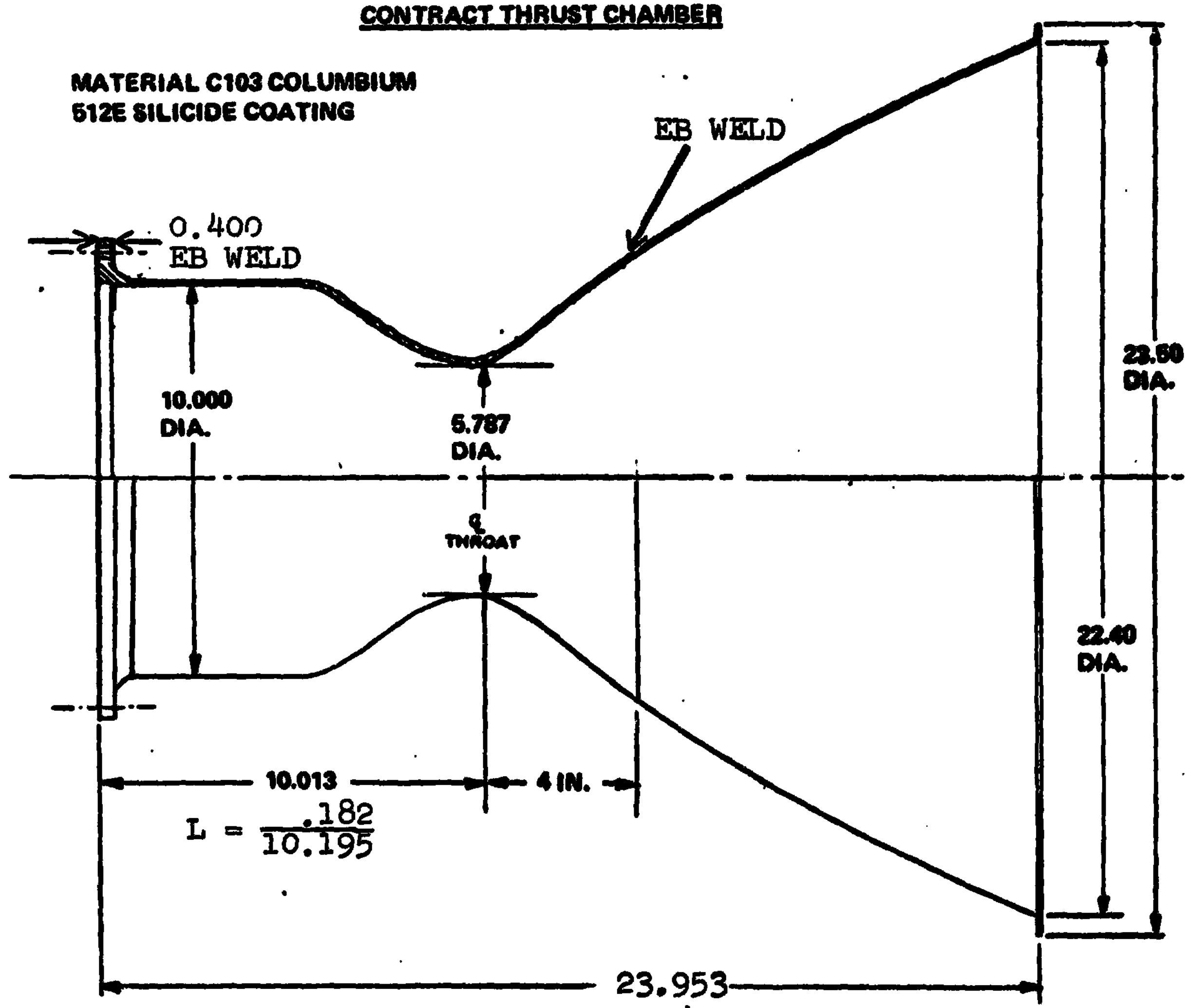
The wall cooling for this application was accomplished by vortex cooling (Figure IV-1). Here, the fuel is given a centrifugal motion by the wall injection scheme, and the outward component assists the adherence of the fuel cooling film as the chamber is traversed. The coating "problem" has also been treated extensively where currently the limit of 2400°F continuous operation on the chamber can be readily accomplished with the Sylcor 512E coating. The rather conservative combination of the C-103 columbium material with the Sylcor 512E coating was used to assure availability and also experience in operation. Other choices to operating levels of up to 3100°F could have been chosen, however, the operation assurance would have been drastically reduced due to shorter term properties. Thus, the material and coating for the demonstration chambers were selected on the basis of full assurance of availability and experience in usage.

2. FABRICATION

The chamber design reflected a study of alternate fabrication techniques to reduce cost and the fabrication schedule. The selected chamber definition included a weld-on injector mounting flange, a longitudinal weld seam in the barrel and convergent

FIGURE V-2

CONTRACT THRUST CHAMBER



THICKNESSES	CHAMBER BARREL	0.105 IN.
	THROAT STATION	0.150 IN.
	EXTENSION	0.080 IN.

divergent nozzle, and a weld-on nozzle extension to an area ratio of 15:1. The extension was fabricated with two longitudinal weld seams. All welds were established as full penetration electron beam type. Testing at Bell on other programs has shown that no significant columbium property reduction is encountered across narrow beam E.B. welds. The demonstrated characteristics of the columbium E.B. welds permitted the multi-section chamber design. The chamber columbium alloy selected, C-103, was based on structural analyses that confirmed its adequacy, the superior forming characteristics of that alloy relative to higher strength Cb alloys and the material availability. The multi-section approach and alloy selection minimized material, tooling and fabrication costs and insured a relatively short material procurement and chamber fabrication lead times.

The coating selected to prevent oxidation of the columbium shell and embrittlement from the combustion gas hydrogen species was HITEMPCO R-512E. The coating selection was based on its demonstrated compatibility with the propellants and all combinations of propellants, moisture and flushing fluids. The R-512E has demonstrated steady-state temperature life in combustion atmosphere (oxyacetylene torch) in excess of 212 hours at 2200°F and 80 hours at 2400°F. The coating has survived more than 10,000 thermal fatigue cycles to 2200°F. Previous experience at Bell and elsewhere has shown that the coating is not easily damaged during normal handling. The columbium coating diffusion zone has a Vickers hardness of 1000 which provides a high resistance to scratch damage. The diffusion zone also reduces base metal hardness increase by hydrazine embrittlement. Essentially no columbium elongation or tensile strength changes have been noted after 10,000 seconds of firing with RCS units.

3. DEMONSTRATION TESTING

While all of the columbium thrust tests were conducted at simulated altitude, three different injectors were used to evaluate the concept. The first injector tested was a stainless steel baffle injector, while the second and third injectors were both flat face aluminum units.

Test results with the stainless steel injector fell approximately 2% short of the predicted I_{sp} at the design operating conditions of 125 psia chamber pressure, N_2O_4/MMH propellants at a mixture ratio of 1.65 and a maximum insulated chamber temperature of 2400°F (Figure V-3). The injector operation in the columbium chamber was characterized by non-uniform throat temperatures associated with the baffle configuration. An attempt to improve the temperature uniformity was successful

DEMONSTRATION TEST DATA SUMMARY

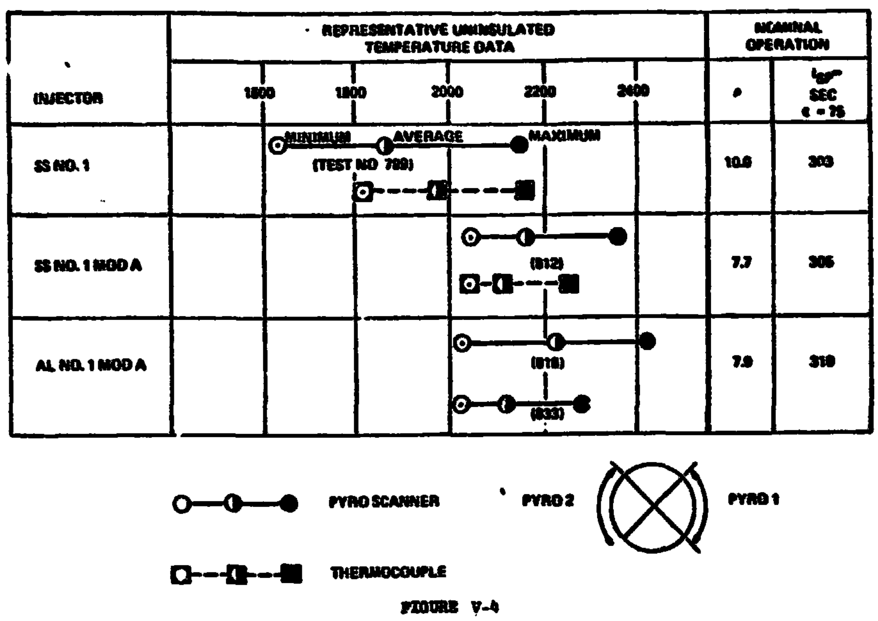


FIGURE V-3

DATA REGRESSION ANALYSES
ALUMINUM INJECTOR MOD A

c^* (INJECTOR P_c, CORE) $= 4919.7 + 0.23\,P_c + 812.7\,r_c - 230.5\,r_o^2 - 2317.9\,\rho$ (RUNS 816 - 839)

WHERE r_c = CORE O/F $= \dfrac{r_o}{1-\rho-r_o\rho}$

r_o = TOTAL O/F

ρ = % VORTEX FLOW

MAXIMUM RESIDUALS, + 21 FT/SEC

- 7 FT/SEC

C_{F_∞}, (ϵ = 75.1) $= 1.72606 + 0.03742\,r_o + 0.00033\,P_c$ (RUNS 816 - 839)

MAXIMUM RESIDUALS, $\left.\begin{array}{l} +0.0053 \\ -0.0059 \end{array}\right\} \approx 0.3\%$

T_{MAX} (PYROSCANNER) $= 3991.7 - 21445.8\,\rho$ (RUNS 829 - 839)

MAXIMUM RESIDUALS, + 67°F

- 47°F

FIGURE V-4

but was achieved at the expense of combustion efficiency. The net result of the change was little or no I_{sp} improvement within the 2400°F maximum temperature restraint. With the agreement of the program monitor, the Task V testing was transferred to the aluminum injector.

The second 6000 lb_f, N_2O_4/MMH, 10 inch diameter injector was fabricated from aluminum and incorporated acoustic cavities for the suppression of high frequency combustion instability. Fuel vortex film cooling was maintained as the approach to the gas film temperature reduction. Aluminum was employed to expedite the injector fabrication. Testing with the second injector with the 30 inch L* chamber demonstrated operation within 0.5 seconds of the I_{sp} goal at the maximum temperature of 2400°F at nominal operating conditions. Tests were also completed using an insulated chamber with this injector. The tests confirmed the off-limits operation temperature predictions. This testing resulted in the regression analysis shown in Figure V-4 and subsequently into the performance map shown in Figure V-5. The last of the testing with the aluminum #1 injector was conducted with an insulated thrust chamber. The temperatures recorded during this testing is shown in Figure V-6.

The predicted longitudinal temperature is also shown in Figure V-6.

4. TEST RESULTS - EFFECT OF L* AND A50 FUEL

Testing to define the effect of L* on performance was accomplished after the availability of the second aluminum injector (See Task XI). Along with the L* (combustor length) evaluation, the effect on heat rejection of reducing the amount of vortex barrier, and the comparison of performance between MMH and A50 fuels were also tested with this injector. The method of obtaining these variables included the film coolant reduction by reducing the flow through the separate vortex manifold. The L* (combustor) variation was accomplished by using a shorter (alternate) columbium chamber for the 26 L* testing and a stainless chamber insert for the 34 L* testing. The A50 fuel tests were made so as to allow a direct comparison with MMH using both 34 and 30 L* chambers. The data accumulated during this testing is assembled in Table V-6.

5. THE EFFECT OF VARYING L* USING MMH FUEL

In order to establish the best estimate of engine performance as a function of the design and operating parameters, R_o, ρ, L* and P_c, of the three L* configurations (26, 30 and 34), the performance data accumulated from 11 the tests, 841 through 869, on the columbium thrust chamber were subjected to statistical evaluation. Short duration firings were not considered and standard regression analysis techniques were employed.

COLUMBIUM THRUST CHAMBER PERFORMANCE - TEMPERATURE MAP ALUMINUM INJECTOR TESTS 829 TO 839

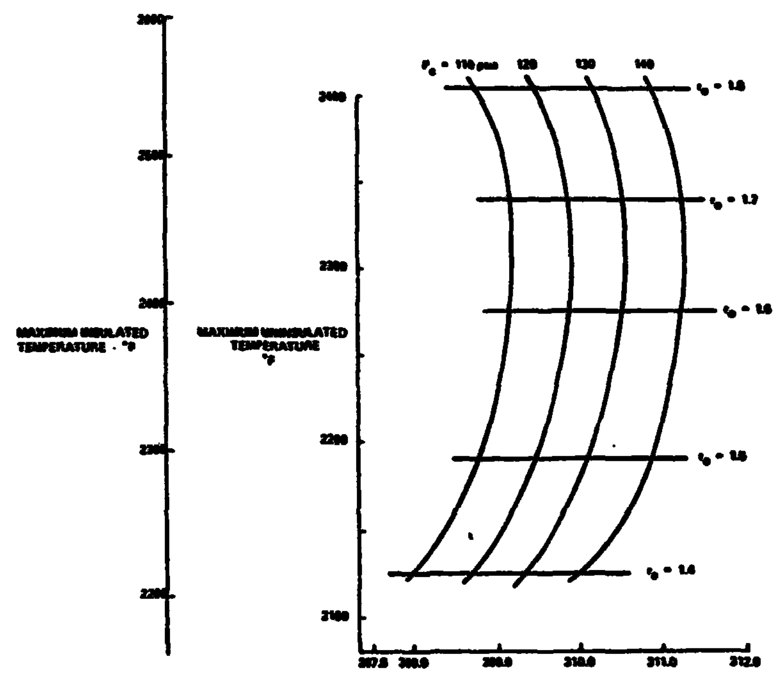


FIGURE V-5

LONGITUDINAL TEMPERATURE DISTRIBUTION INSULATED TEST No. 828 60 SEC. DATA

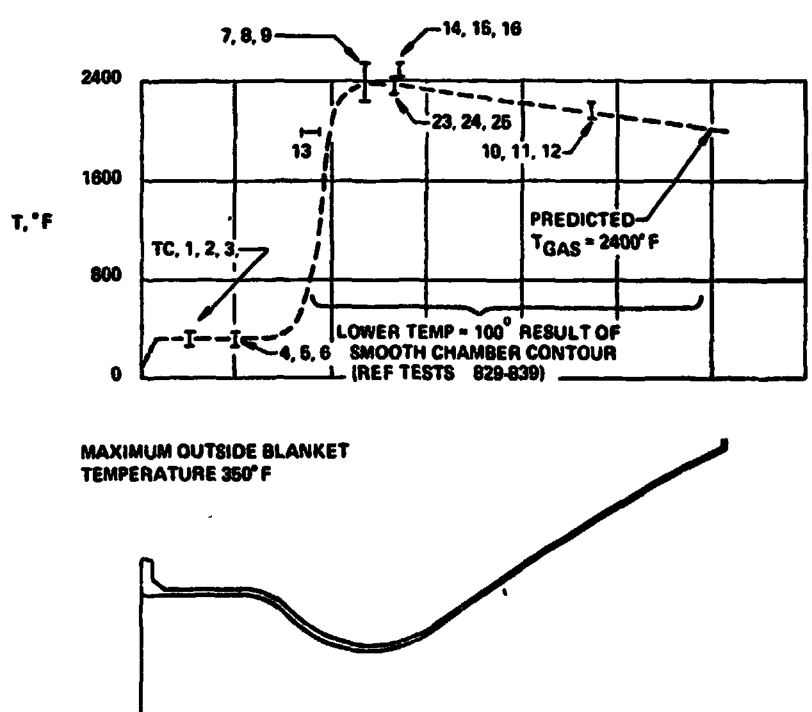


FIGURE V-6

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TABLE V-6

TEST DATA SUMMARY - ALUMINUM NO. 2 INJECTOR

TEST NO.	DURATION (SEC)	$P_{C_{corr}}$ (PSIA)	O/F	ρ %	η_{C*} %	I_{sp_∞} ϵ=75:1	$I_{sp_{\infty N}}$ (4) ϵ=75:1	(1) C* (FT/SEC)	C_F	(2) $T_{MAX.}$ (°F)	L* (IN.)	PROPELLANTS
841	2.0	122.9	1.628	7.99	95.2	307.1	-	5437	1.819	NA	30	N_2O_4/MMH
842	8.1	120.8	1.560	8.08	96.6	311.3	311.2	5509	1.820	2453	30	N_2O_4/MMH
843	19.8	122.0	1.657	7.93	95.8	310.3	310.0	5471	1.827	2410	30	N_2O_4/MMH
844	9.9	121.6	1.451	8.60	98.1	310.1	310.6	5513	1.811	2546	30	N_2O_4/MMH
845	13.0	121.4	1.460	8.58	96.9	310.2	310.7	5503	1.815	2642	30	N_2O_4/MMH
846	20.0	121.3	1.800	7.50	95.2	309.3	308.5	5428	1.835	2271	30	N_2O_4/MMH
847	2.0	125.5	1.658	7.94	95.4	306.5	-	5452	1.810	NA	26	N_2O_4/MMH
848	20.1	121.8	1.619	8.03	95.4	308.2	308.0	5447	1.822	1943	26	N_2O_4/MMH
849	20.0	121.4	1.445	8.61	96.1	307.2	307.7	5455	1.814	NA	26	N_2O_4/MMH
850	20.0	123.7	1.591	7.01	96.3	309.7	308.4	5499 (3)	1.813 (3)	2223	26	N_2O_4/MMH
851	20.0	123.9	1.408	7.54	97.2	308.8	308.1	5504 (3)	1.806 (3)	2175	26	N_2O_4/MMH
852	20.0	121.7	1.648	6.80	95.[illegible]	310.0	308.4	5457 (3)	1.829 (3)	2334	26	N_2O_4/MMH
853	20.0	121.4	1.483	7.26	96.1	309.6	308.6	5465 (3)	1.824 (3)	2243	26	N_2O_4/MMH
854	20.1	124.0	1.642	6.81	96.3	310.0	308.5	5498 (3)	1.816 (3)	2346	26	N_2O_4/MMH
855	5.0	122.3	1.672	8.96	95.6	309.7	-	5461	1.826	1798	34	N_2O_4/MMH
856	19.8	120.5	1.661	8.99	96.1	311.5	312.4	5488	1.828	2617	34	N_2O_4/MMH
857	20.1	122.6	1.659	7.96	96.5	312.8	312.5	5515	1.827	2606	34	N_2O_4/MMH
858	20.1	122.4	1.655	7.98	96.6	313.0	312.8	5517	1.827	2630	34	N_2O_4/MMH
859	4.9	124.4	1.629	8.08	95.8	310.0	-	5485	1.820	1620	30	N_2O_4/A-50
860	20.0	121.8	1.635	8.02	95.9	310.5	310.6	5499	1.821	2268	30	N_2O_4/A-50
861	MALFUNCTION										30	N_2O_4/A-50
862	20.1	124.1	1.573	7.09	96.7	313.1	311.4	5540	1.820	2418	30	N_2O_4/A-50
863	19.8	122.7	1.628	6.82	96.4	312.5	310.3	5524	1.822	2466	30	N_2O_4/A-50
864	19.8	121.9	1.634	9.06	95.4	309.3	311.3	5463	1.823	2255	34	N_2O_4/A-50
865	MALFUNCTION										34	N_2O_4/A-50
866	20.0	123.3	1.589	8.17	96.0	312.9	313.2	5504	1.821	2451	34	N_2O_4/A-50
867	20.0	124.5	1.384	8.87	97.1	313.9	315.7	5543	1.823	2378	34	N_2O_4/A-50
868	19.7	122.4	1.617	6.96	96.8	313.5	312.1	5526	1.827	2530	34	N_2O_4/MMH
869	20.1	122.0	1.686	6.67	96.7	313.8	312.1	5526	1.828	2543	34	N_2O_4/MMH

NOTES:
(1) P_c injector x 0.981 correction
(2) Pyroscanner maximum temperature
(3) Temperature for throat area based on pyroscanners
(4) Normalized to 8% ρ film cooling

All performance data were normalized to a ρ of 8% to provide a common reference for performance comparison of the three L* configurations. A plot of $I_{sp_{\infty n}}$ with mixture ratio for the three L* configurations is shown in Figure V-7.. A high degree of consistency was achieved with maximum deviations between best fit and experimental results of less than ±0.2%.

A comparison of maximum temperatures in the columbium chamber at 30 L* is also included as Figure V-8.

6. THE EFFECT ON PERFORMANCE OF VARYING ρ

A series of tests were made varying ρ (film coolant percent of total propellants) at a constant 26 L*. These results are given graphically in Figure V-9. In the test region, the results appear to produce a two second performance decrease, as a 1% increase in film coolant is injected.

7. EFFECT OF ρ ON NOZZLE TEMPERATURES

The data obtained with 34, 30 and 26 L* chambers relating film cooling (ρ) to the maximum chamber temperature is shown in Figures V-10 and V-11. The data plotted in both curves is as recorded although the shape of the curves shown in Figure V-10 present some difficulty in interpretation and understanding. The 26 L* data appears straight forward and believable. However, the 30 L* and 34 L* data appear to be reversed in the expected effect of increasing the fuel flow.

The most reasonable explanation for this reversal would appear to be that the "unknown" effect of mixing between the fuel film and the primary combustion was changing as the film coolant quantity changed and the resulting maximum temperature, at the throat was as shown. Subsequent test data using the same instrumentation produced predicted results lending credence to the accuracy of the recorded information.

The testing with A50 fuel resulted in the data shown in Figures V-12 and V-13. Operation with this fuel was performed without incidence, attesting to the interchangeability of the two propellants.

SPECIFIC IMPULSE VS MIXTURE RATIO
ALUMINUM INJECTOR NO. 2-26L*, 30L*, 34L*
CHAMBER RUNS 841-858, 868, 869

I_{SP} CORRECTED TO VACUUM OPERATION
AND AREA RATIO 75:1 AND NORMALIZED
TO 8% ρ FILM COOLING
N_2O_4/MMH PROPELLANTS

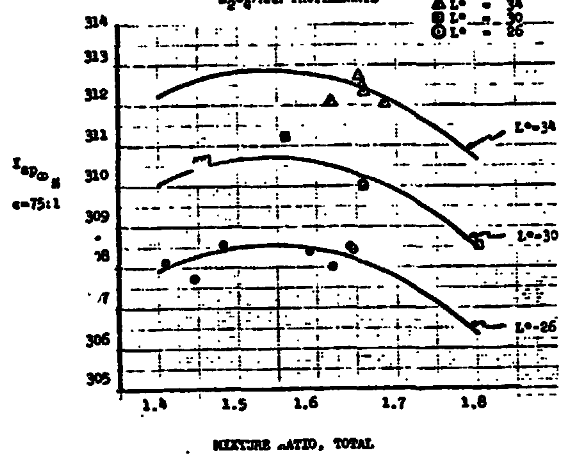


FIGURE V-7

MAXIMUM PYROSCANNER TEMPERATURE
VS
MIXTURE RATIO
ALUMINUM INJECTOR NO. 2-30L* CHAMBER
RUNS 841-846

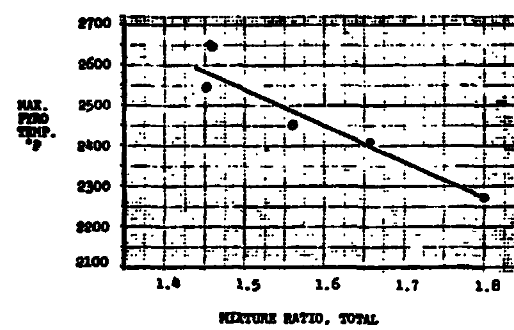


FIGURE V-8

MAXIMUM PYROSCANNER TEMPERATURE
VS
PERCENT VORTEX FLOW
ALUMINUM INJECTOR NO. 2 - 30L*, 34L*
RUNS 841-846, 855-858, 868, 869
N_2O_4/MMH PROPELLANTS

SPECIFIC IMPULSE VS MIXTURE RATIO
ALUMINUM INJECTOR NO. 2 - 26L* CHAMBER
RUNS 847-854

I_{SP} CORRECTED TO VACUUM OPERATION
AND AREA RATIO 75:1

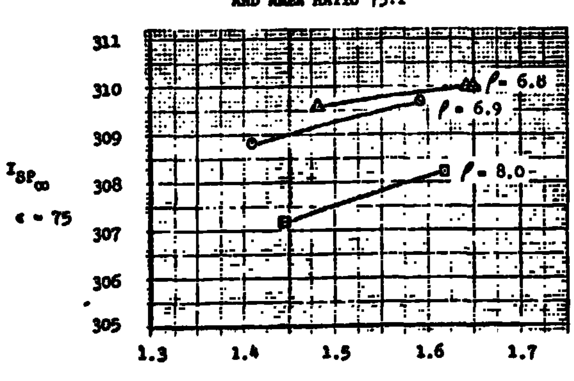


FIGURE V-9

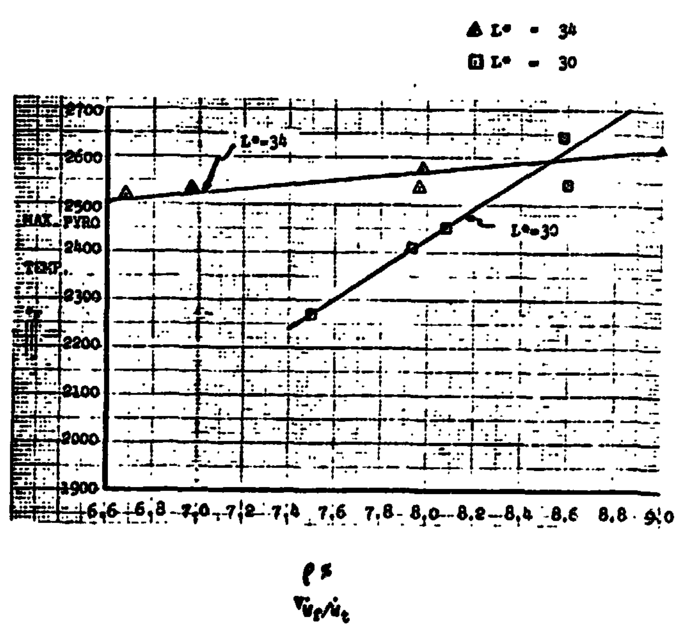


FIGURE V-10

MAXIMUM PYROSCANNER TEMPERATURE
VS
PERCENT VORTEX FLOW
ALUMINUM INJECTOR NO. 2 - 26L* CHAMBER
RUNS 847-854

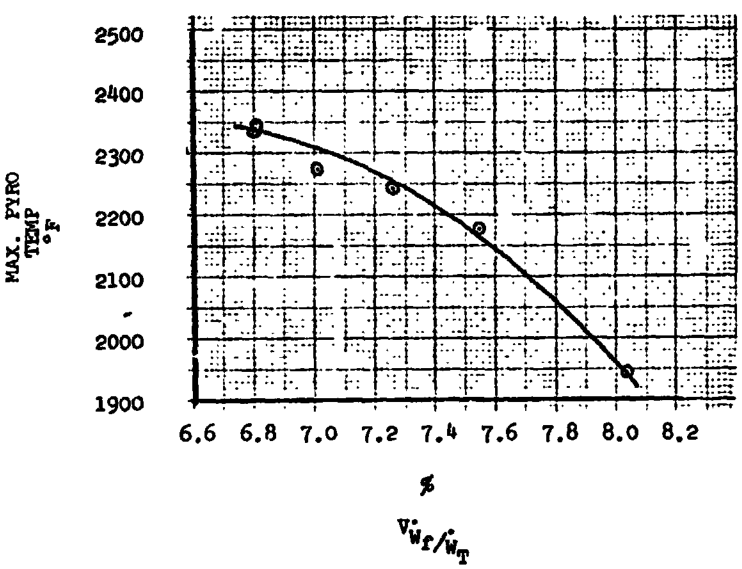


FIGURE V-11

SPECIFIC IMPULSE VS MIXTURE RATIO
ALUMINUM INJECTOR NO. 2 - 30L*, 34L* CHAMBER
RUNS 859 - 867
I_{SP} CORRECTED TO VACUUM OPERATION AND AREA RATIO 75:1 AND NORMALIZED TO 8% ρ FILM COOLING

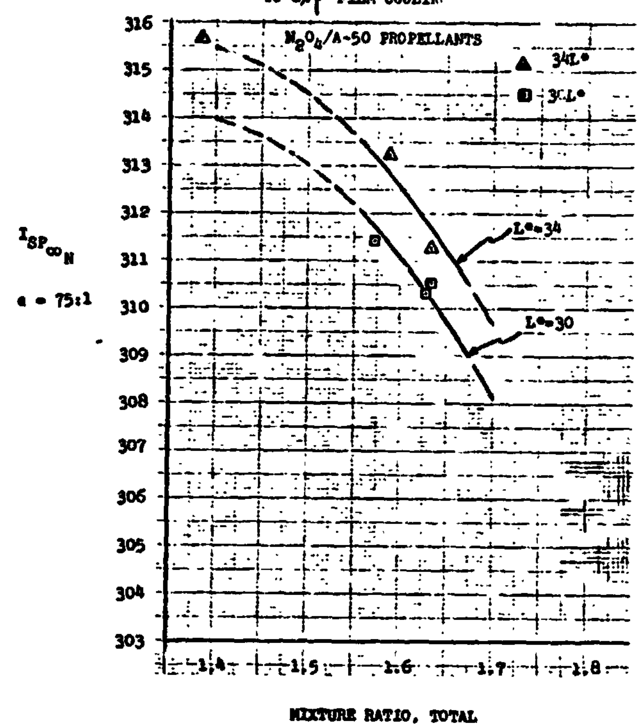


FIGURE V-12

MAXIMUM PYROSCANNER TEMPERATURE
VS
PERCENT VORTEX FLOW
ALUMINUM INJECTOR NO. 2 - 30L*, 34L* CHAMBER
RUNS 859 - 867
N_2O_4/A-50 PROPELLANTS

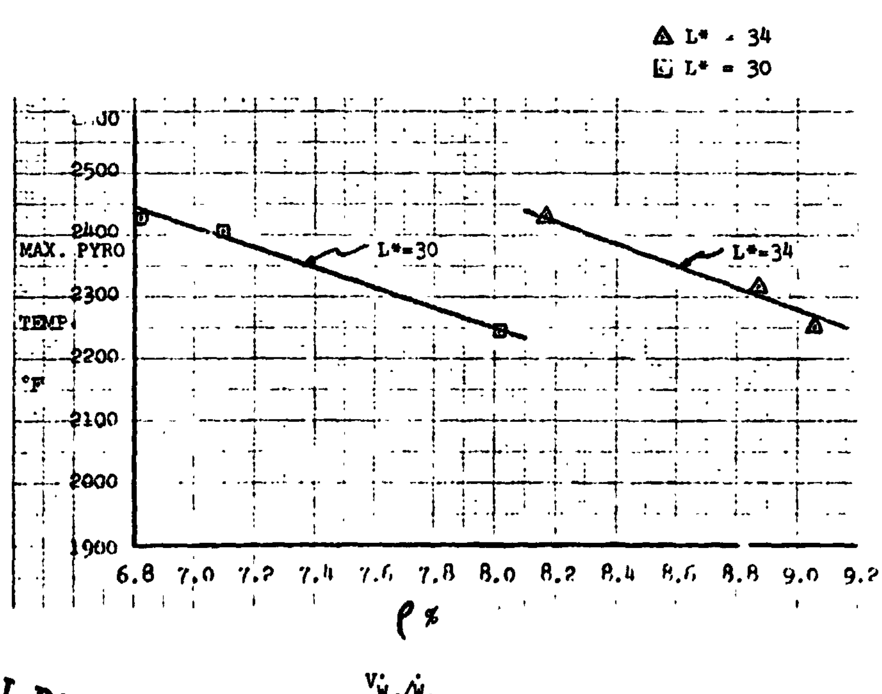


FIGURE V-13

The conclusion drawn from this testing was that the two fuels, A50 and MMH, are nearly interchangeable. Some adjustment in the amount of film coolant would be required to adjust wall temperatures, but the basic hardware would appear, on this cursory evaluation, to be readily usable for either propellant.

8. CONCLUSIONS FROM TESTING

The extensive testing, conducted the columbium thrust chamber during the Task V and VIII effort proved that the operation of the columbium chamber was not only a viable concept but that operation could be predicted and adjusted to meet the design, operation and margin requirements of the engine. The fabrication of this engine was simple in comparison to most other contemporary concepts of reusable rocket thrust chambers. The simple use of a spun metal shell, coated and insulated far simplifies comparable multi-channel or multi-layer concepts of regenerative or ablative cooled units. However, with other non-perfect systems, a slight penalty would be paid with usage and this penalty would be in an increased wall coolant, and consequent performance, reduction to pay for the simplicity and low cost usage of the concept. With the achievement of 310 seconds in the current design, the performance penalty would be expected to be approximately 1 1/2% as compared to a regeneratively cooled thrust chamber. It is possible that this penalty could be reduced with further definition of coating capability or cycle reduction or metal improvement. However, at the present time it would appear that the current insulated columbium engine design would be limited to the materials and coatings currently being used.

C. REGENERATIVELY COOLED THRUST CHAMBER

Two series of tests were conducted with the regeneratively cooled thrust chamber to demonstrate operation as well as to define the level of performance that could be expected for such an engine. One series of simulated altitude tests were conducted at the Bell Test Center Altitude facility and the other at the NASA-WSTF. The initial series of tests were conducted at the Bell facility where the engine operation as well as performance was defined. Unfortunately, this facility was limited by the center duct size such that a nozzle area ratio of only 15 could be used. The full size engine was tested at WSTF with a nozzle furnished by WSTF and attached to the Bell thrust chamber by means of a special adapter. The conduct of these two series of tests allowed a rather unique comparison of facilities to be made, as well as to obtain a more universally acceptable performance.

1. BELL TEST CENTER TESTING

Eighteen test firings, with an accumulated run time of 199 seconds, were conducted on the regeneratively cooled chamber S/N 2 flat face injector combination. Testing was conducted in the Bell 1BN test complex where a nominal altitude of 110,000 ft. is maintained during operation. A test nozzle with an area ratio of 15 was used on all tests, this nozzle size being limited by the test facility. A two phase flow system was also used to accelerate fuel coolant film times during this program. This procedure was required as the injection manifolds were sized for a larger flow (i.e. greater than 4#/second film or vortex cooling), while the regeneratively cooled chamber requires less than 2#/second fuel flow. As a consequence, the tests were initiated at the higher flow (4%) and then switched to the lower flow (2%). One side benefit of such a procedure was to obtain several data points on each test.

2. PERFORMANCE

The performance data is presented in the plot of specific impulse and versus mixture ratio for all test durations of 10 seconds or greater. The specific impulse versus mixture ratio is shown in Figure V-14 for fuel vortex percentages of 4.1% and 1.9% at high, nominal and low chamber pressure.

The specific impulse and characteristic velocity versus test duration for the 30 second test (1BN-887) is shown in Figure V-15. Stabilization of specific impulse was achieved after approximately 5 seconds duration. Characteristic velocity shows a slight decrease with duration due to apparent throat area

ONE PERFORMANCE

$I_{SP\infty}$ VS O/F

$\epsilon = 75:1$

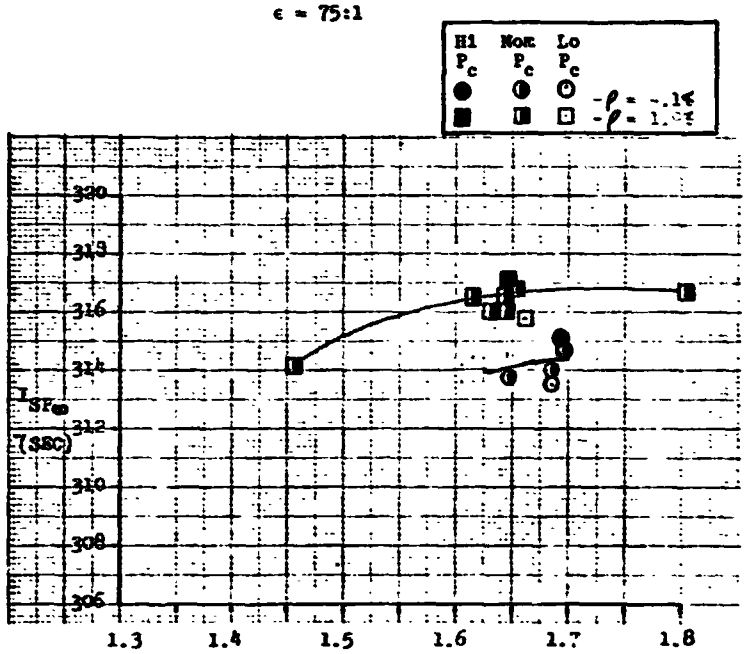


FIGURE V-14

ONE PERFORMANCE VS TIME

RUN NUMBER 1BN-887

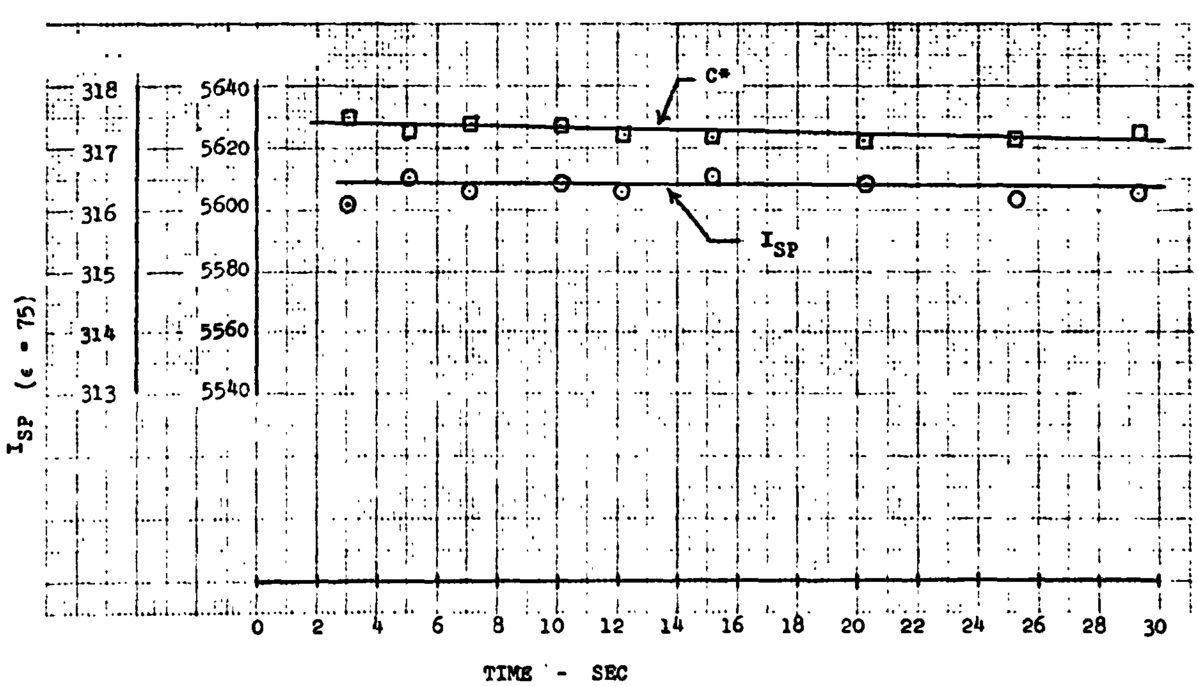


FIGURE V-15

change which was not applied to these data.

3. HEAT REJECTION

The fuel jacket temperature rise effect versus mixture ratio is shown in Figure V-16 for the two vortex flow conditions and at the three chamber pressure levels. These data are presented at the 10 second point. The fuel jacket temperature rise versus test duration for the 30 second test is given in Figure V-17. The total chamber heat load vs chamber pressure is shown in Figure V-18.

4. SHELL AND NOZZLE TEMPERATURE

The external shell of the regeneratively cooled chamber was instrumented with a total of fifteen (15) chromel/alumel thermocouples, with an additional 9 thermocouples located on the columbium nozzle extension.

The temperature profile produced by 13 of these thermocouples (chamber flange) on the chamber skin and the 9 nozzle extension temperatures for the 30 second duration test is shown in Figure V-19 (at the end of run data points).

The maximum ch[illegible] skin temperature versus mixture ratio is shown in Figur[illegible] for all tests of 10 seconds duration or greater. The da[illegible]resented are for all 10 second points. The maximum temperature measured is located at the injector end of the chamber (thermocouple T-1 or T-2). The increase of maximum chamber temperature between the 10 second data point and the thermally stabilized end of run data point is less than 10°F.

5. START TRANSIENTS

Starting of the regeneratively cooled engine was predicated on several setup requirements. These being an oxidizer lead requirement and also that the fuel film coolant and main fuel flows enter as simultaneously as possible to eliminate possible flashback and overheating. Unfortunately, in this separately fed hardware, to achieve this type of propellant timing required orificing and volume changes because of different flow adjustments to the film manifold. While this criteria (fill time entry) was accomplished, the feed system used was much more involved than normal, leaving very little useful data to project to a final flight engine transient analysis.

FIGURE V-16

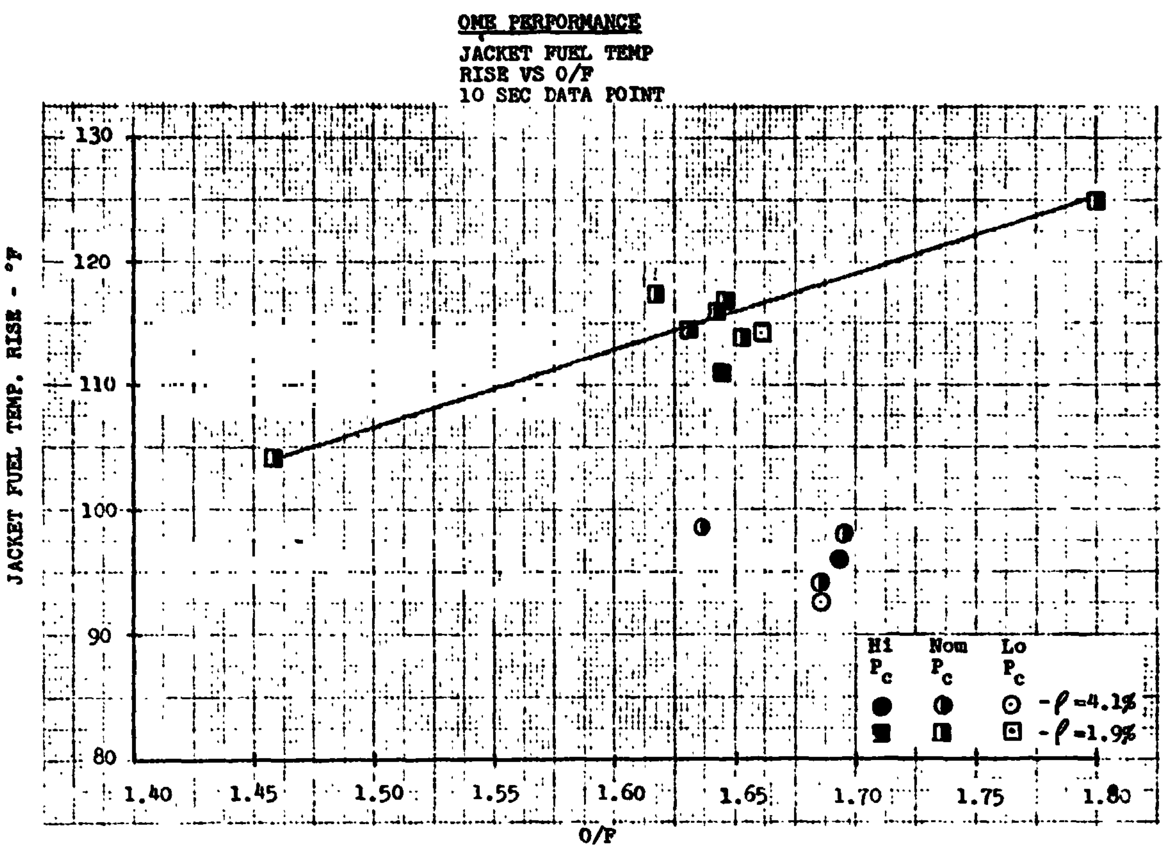


FIGURE V-17

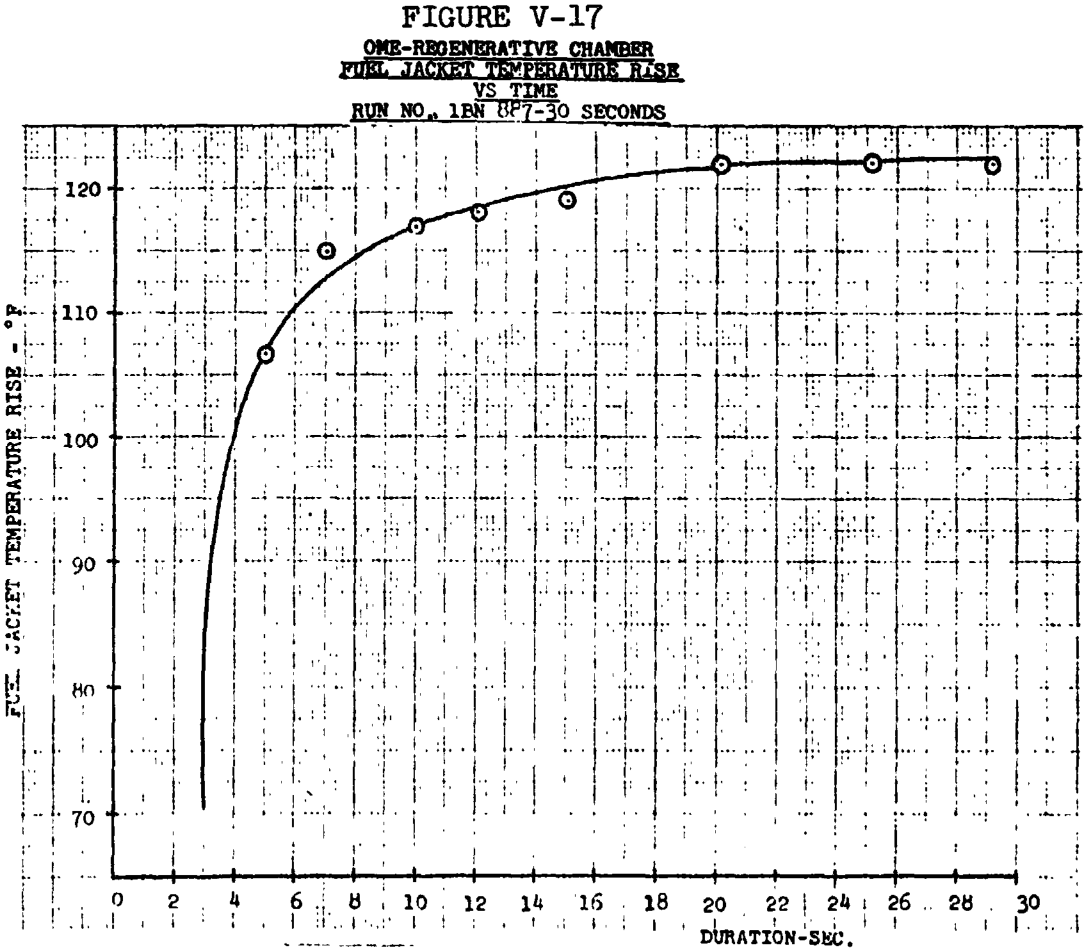


FIGURE V-18

TOTAL HEAT LOAD VS CHAMBER PRESSURE

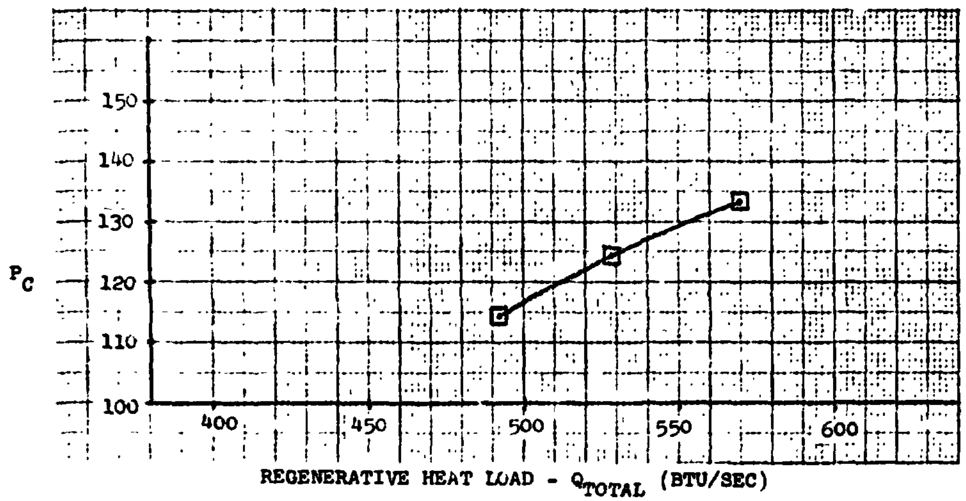
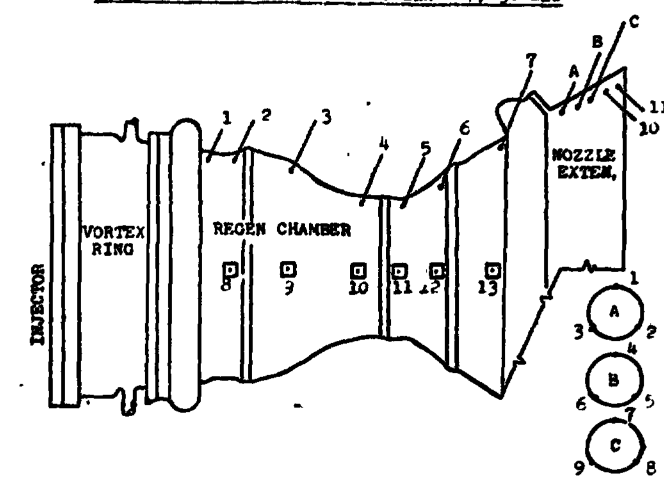
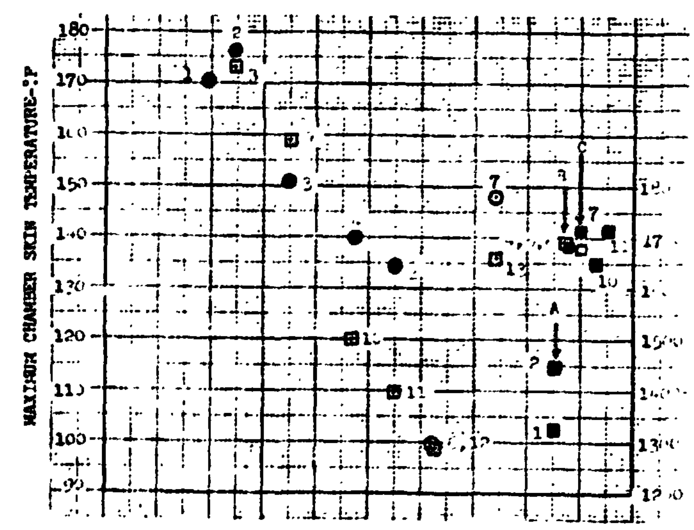


FIGURE V-19

OME - REGEN CHAMBER

TEMPERATURE PROFILE, RUN NO. 1BN-887, 30 SEC





OME PERFORMANCE

MAXIMUM CHAMBER SKIN TEMPERATURE (AT INJECTOR END) VS O/F

10 SECOND DATA POINTS

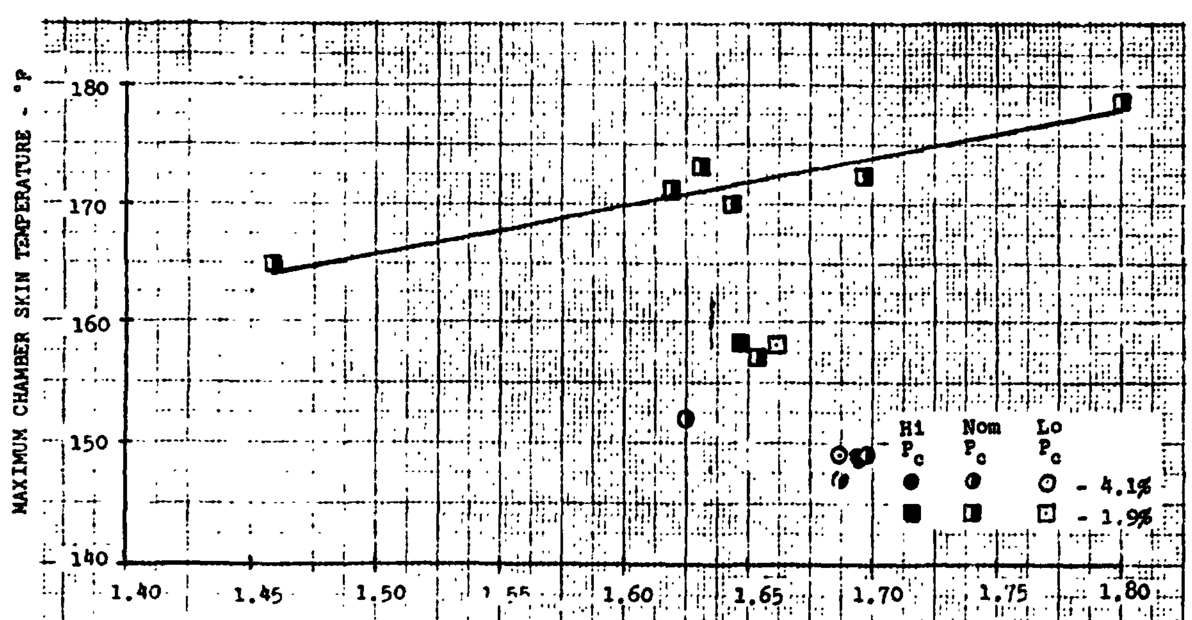


FIGURE V-20

O/F

6. WSTF TEST AND ANALYSIS SUPPORT

The test objective for the full scale OME tests at WSTF was to define performance for this thrust chamber assembly at a simulated altitude of approximately 100,000 feet with N_2O_4/MMH propellants. Sub-objectives addressed the performance variation with excursions of mixture ratio, chamber pressure, combustion length, helium saturated propellants and heated propellants.

A total of 47 altitude test firings were conducted at WSTF using the 76.7 area ratio nozzle extension. Data are supplied for these tests as well as comparison with data from testing at BAC with a 15:1 area ratio nozzle extension and projected to a typical OME vehicle nozzle envelope at an area ratio of 72.7.

These tests with N_2O_4/MMH propellants confirmed the BAC performance predictions which were determined by the JANNAF procedures. The WSTF data indicated a slightly higher performance (0.2%) than predicted from the BAC data (Isp of 317.5 vs 317.0 seconds). No noticeable performance differences were observed with increased chamber length (30 L* to 34 L*) or with helium saturated propellants versus unsaturated propellants. A chamber pressure variation resulted in a performance change of about 0.03% seconds I_{sp}/psi P_c. A detectable increase in performance of about 0.3% was also noted when propellant temperatures were increased.

No combustion instability was noted during the test series and the projection of maximum nozzle extension temperature was lower than the original study value but almost identical to the temperature predicted from the 15:1 area ratio nozzle tests at BAC.

7. TEST RESULTS

Through previous experience, it has been shown that raw test data, normalized by "best fit" procedures using statistical regression techniques, results in a much superior representation of a single variable of investigation. This technique was applied to the WSTF data, as well as the Bell Test Center data, and the results used to present the single variable influence as well as to make direct comparison of test facilities. In general, the test data presented have been corrected to the following conditions

$$P_c = 125 \text{ psia}$$
$$\rho = 1.9\%$$
Propellant Temperatures = Ambient ($75 \pm 10°F$)

Data is presented comparing the BAC test cell 1BN data with WSTF data. Data from 1BN were obtained with a 15/1 area ratio nozzle. Those at WSTF were for an $\epsilon = 76.7/1$. This data is compared by extrapolating each set of data to a common nozzle configuration by using the standard form of procedures. The typical shuttle geometry selected for presentation has a nozzle area ratio of $\epsilon = 72.7/1$ and a length of 59.1 inches. The test nozzles were "corrected" to this condition to present comparison data.

8. WSTF TEST DATA NORMALIZED

The WSTF test data were normalized and plotted with the effects of the test variables considered for each presentation. On the various graphs, the nominal curve is based on the 45 tests normalized to 125 psia chamber pressure and ambient propellant temperatures. The 1.5 second and 5.0 second tests are not included because of their short run durations.

The $I_{sp_{\infty}\ N\ (\epsilon = 76.7)}$ for this test data was normalized to 125 psia chamber pressure and nominal propellant temperature in accordance with the following equation:

$$I_{sp_{\infty}\ N\ (\epsilon = 76.7)} = 180 + 0.093\ P_{cc} + 0.83\ (\text{Hot})$$

$$+\ 137.267\ R_{o/f} - 37.18\ R_{o/f}^{\ 2}$$

where: P_{cc} = total chamber pressure at entrance to nozzle

$R_{o/f}$ = overall propellant mixture ratio

Adjustments to the values for chamber pressure and propellant temperature can be made as follows:

For P_{cc} use Δ from 125

For hot propellants use factor of 1.0

Figure V-21 resulted from the compilation of all WSTF test data and results in an excellent presentation of the specific impulse variation expected as the test mixture ratio is varied. A comparison of normalized unsaturated and saturated data indicated that the effects of helium in the propellants is insignificant and that these results are presented in Figure V-22.

The effects of L* variation appeared to be insignificant over the range of test conditions and indicated the insensitivity of the triplet injector design to L*'s between 30 and 34 inches (Figure V-23).

The effects of propellant temperature were significant as shown in Figure V-24 where heated propellant tests are compared with ambient propellant results for the 34 L* chamber at a chamber pressure of 125 psia. At a mixture ratio (O/F) of 1.65 the gain in performance was 0.83 seconds of impulse or 0.27% when oxidizer temperature is raised from approximately 75°F to 94°F and fuel is raised from 75°F to 104°F.

Chamber pressure effects were also significant as shown in Figure V-25 where 125 psia data is compared with 135 psia and 115 psia data. Both 30 L* and 34 L* data were used since there was no significant performance difference between them. The test data points for each chamber pressure range ($115\pm^{2.2}_{1.3}$, $125\pm^{1.5}_{2.3}$, $135\pm^{1.1}_{2.1}$) were normalized to nominal values of 115, 125 and 135 psia in the plot and the curves for 115 psia, 125 psia and 135 psia are based on test data in those ranges. At a mixture ratio 1.65, the specific impulse for the three chamber pressures varied as follows:

	P_c Psia	I_{sp_∞} (ϵ =76.7) lb_f - sec-lbm	ΔI_{sp} lb_f-sec	%
	135	317.8	+0.9	+0.28
Nominal	125	316.9	-	-
	115	316.0	-0.9	-0.28

This indicated that a 10 psi increase or decrease in chamber pressure from the nominal would result in a 0.28% increase or decrease in impulse performance respectively.

9. WSTF AND BAC DATA COMPARISON

One of the original objectives of the WSTF test program was to obtain direct data comparison between facilities, in this case the BAC altitude test cell 1BN and the WSTF facility. Tests were to be conducted at WSTF with the ϵ =15 nozzle so that the direct data comparison could be made. Unfortunately these tests were not conducted, and the comparison was made on the basis of a JANNAF extrapolation from the 15 to 76.7 nozzle.

FIGURE V-21

SPECIFIC IMPULSE VS MIXTURE RATIO
6K OME REGEN CHAMBER - 30L*, 34L*
ALUMINUM INJECTOR NO. 2
WSTF TEST DATA
I_{sp} Normalized to 125 Pc and Nominal Propellant Temperature

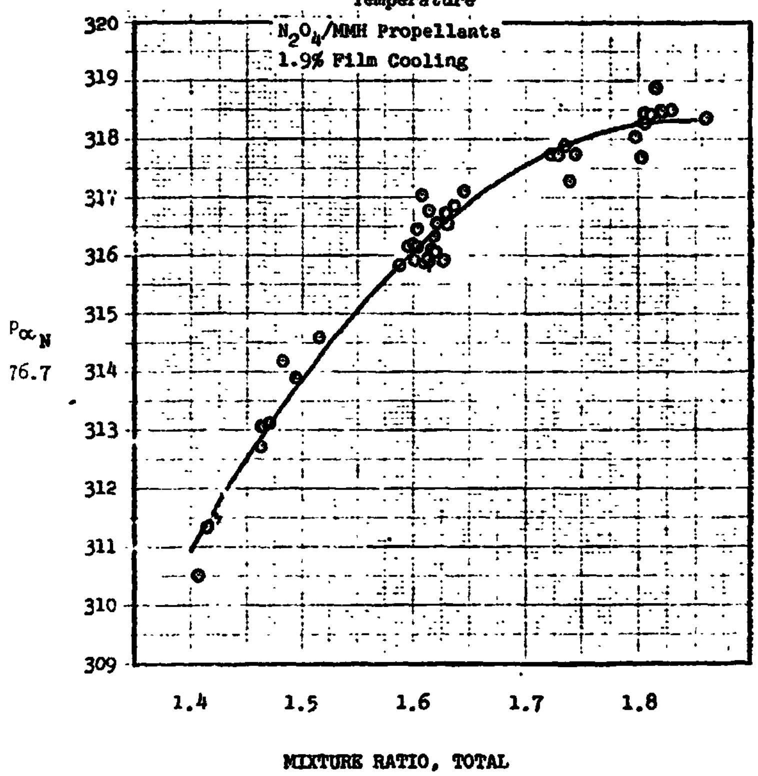


FIGURE V-22

SPECIFIC IMPULSE VS MIXTURE RATIO
6K OME REGEN CHAMBER - 30 L*
ALUMINUM INJECTOR NO. 2
WSTF TEST DATA

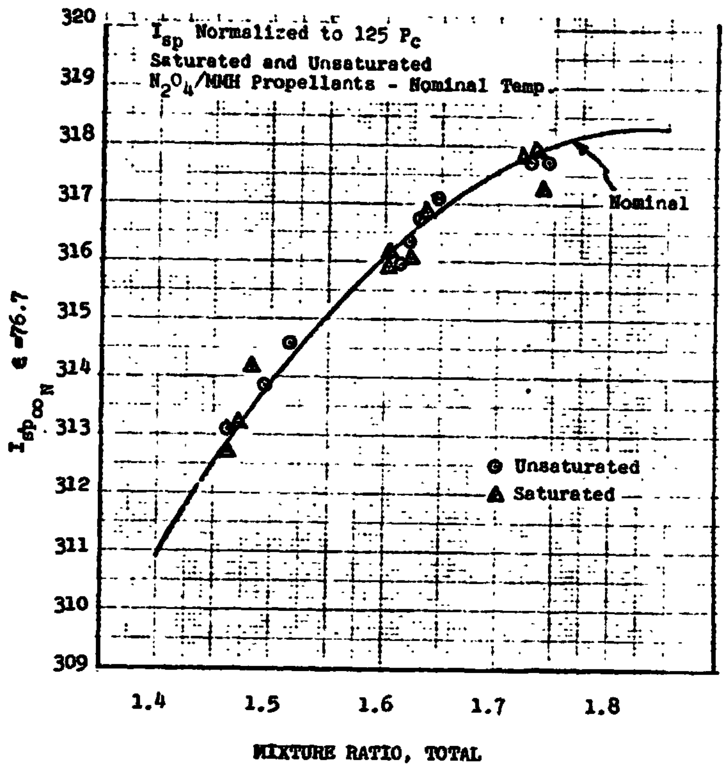


FIGURE V-23

SPECIFIC IMPULSE VS MIXTURE RATIO
6K OME REGEN CHAMBER - 30L*, 34L*
ALUMINUM INJECTOR NO. 2
WSTF TEST DATA
I_{sp} Normalized To 125 P_c
Unsaturated N_2O_4/MMH Propellants-
Nominal Temp.

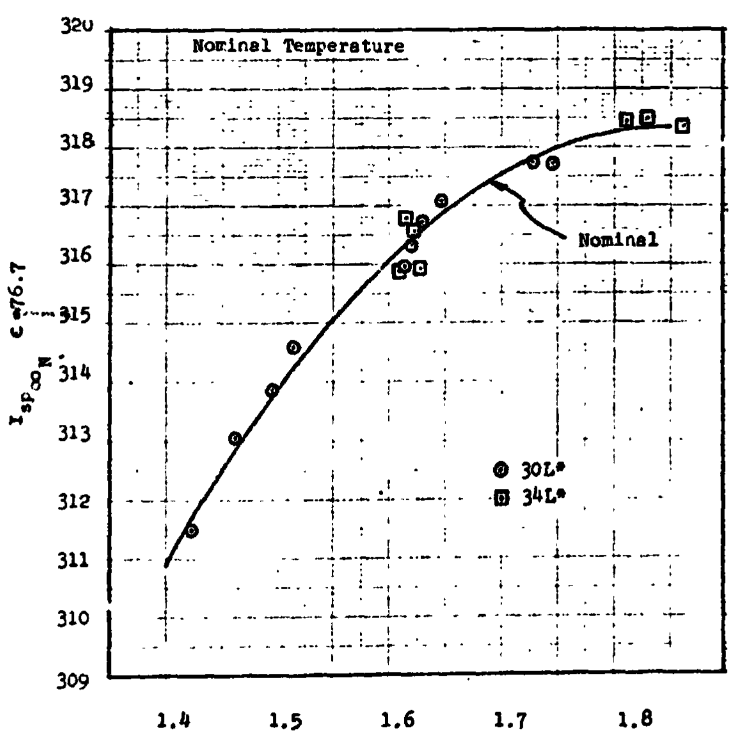


FIGURE V-24

SPECIFIC IMPULSE VS MIXTURE RATIO
6K OME REGEN CHAMBER - 34 L*
ALUMINUM INJECTOR NO. 2
WSTF TEST DATA

I_{sp} Normalized To 125 Pc
Unsaturated N_2O_4/MMH Propellants -

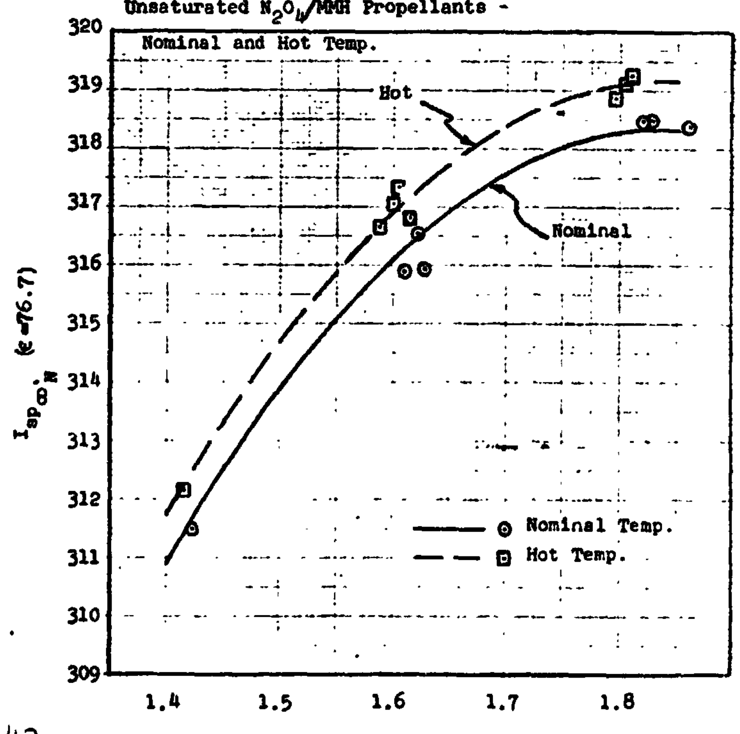


FIGURE V-25

SPECIFIC IMPULSE VS MIXTURE RATIO
6K OME REGEN CHAMBER - 30L*, 34L*
ALUMINUM INJECTOR NO. 2
WSTF TEST DATA
Unsaturated N_2O_4/MMH Propellants -
Low, Nom. and High Chamber Pressure

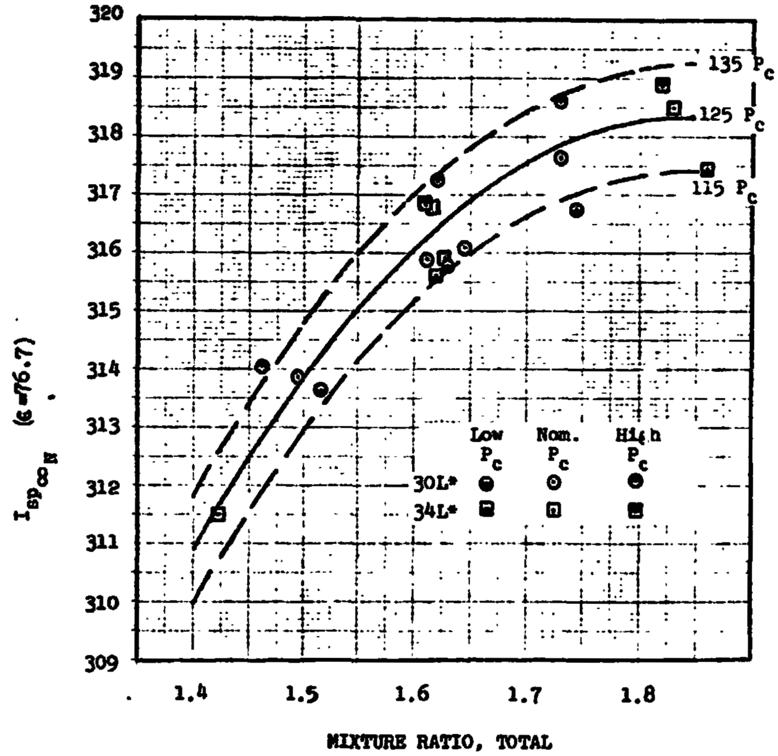


FIGURE V-26

SPECIFIC IMPULSE VS MIXTURE RATIO
6K OME REGEN CHAMBER
ALUMINUM INJECTOR NO. 2
COMBINED WSTF/BAC DATA
I_{sp} Corrected to Vacuum Operation and Area Ratio 72.7:1, Normalized to 125 P_c, 30 L*, 1.9% Film Cooling, Unsaturated and Nominal Propellant Temperature - N_2O_4/MMH Propellants

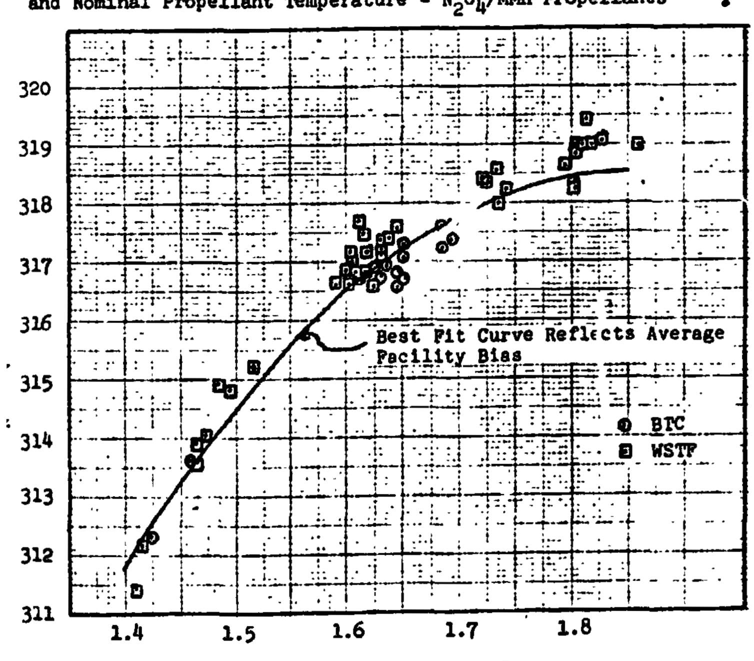
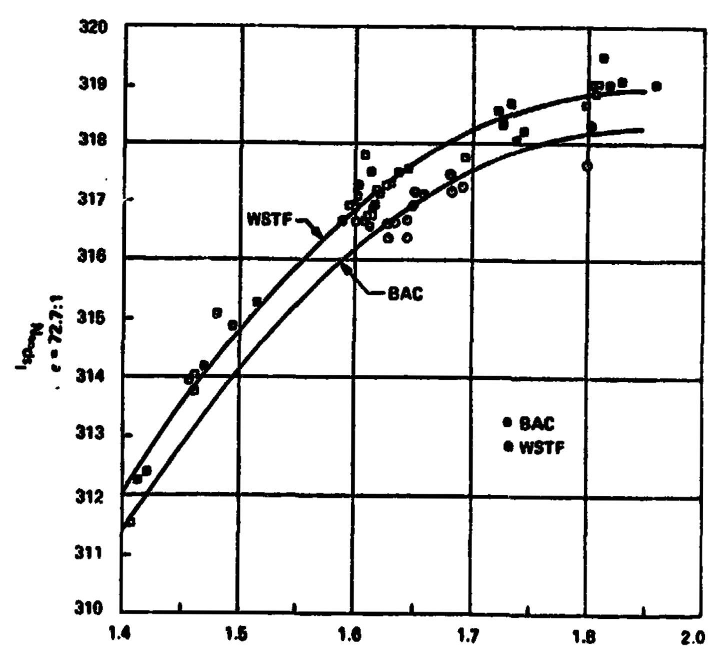


FIGURE V-27

SPECIFIC IMPULSE VERSUS MIXTURE RATIO
6K OME REGEN CHAMBER
ALUMINUM INJECTOR NO. 2
COMBINED WSTF/BAC DATA
I_{sp} CORRECTED TO VACUUM OPERATION AND AREA RATIO 72.7:1, NORMALIZED TO 125 P_c, 30L*, 1.9% FILM COOLING, UNSATURATED AND NOMINAL PROPELLANT TEMPERATURE - N_2O_4/MMH PROPELLANTS



Comparisons of the WSTF data and BAC data were made and performance values analyzed by means of a multiregression correlation analysis. The influence coefficients were determined for chamber pressure, mixture ratio, propellant saturation, propellant temperature, chamber L* and vortex flow. Performance was normalized for a chamber pressure of 125 psia, chamber L* of 30, 1.9% vortex flow, unsaturated and nominal propellant temperature. The specific impulse versus mixture ratio corrected to vacuum operation and nozzle area ratio of 72.7:1 is shown in Figure V-26.

The normalization was accomplished by using the derived correlation equation of the test data:

$$I_{sp_{oo}\ (\epsilon = 72.7)} = 189.5 + 0.073\ L^* - 117.2\ \rho + 0.094\ P_{cc} + 0.086\ \mathrm{Sat} + 0.85\ \mathrm{Hot} + 127.9\ R_o - 34.8\ R_o^{\ 2} - 0.5_{(1BN)}$$

where:

R_o	=	overall propellant mixture ratio
ρ	=	vortex flow percentage
P_{cc}	=	total chamber pressure at entrance to nozzle
L*	=	chamber size
Sat	=	1.0 for saturated propellants
Hot	=	1.0 for hot propellants
1BN	=	facility bias

The WSTF data indicates somewhat higher performance over the mixture ratio range tested. At nominal conditions, the specific impulse measured at WSTF is about 0.3% higher than BAC values (Reference Figure V-27). Based on the data on both facilities, a specific impulse of at least 317 seconds is indicated at the nominal operating conditions of P_c=125 psia, $R_{o/f}$=1.65 and ρ=1.9% in an 30 L* chamber, whereas WSTF data indicates a specific impulse of 317.5 seconds. These results are based on a nozzle ϵ=72.7 which would be typical for Space Shuttle envelope.

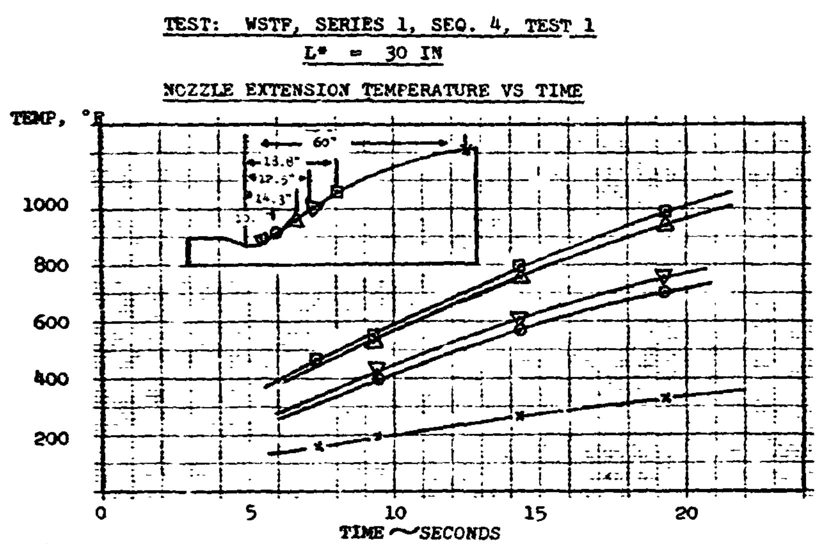
10. THERMAL DATA

Thermal data taken during these tests included regen chamber fuel inlet and outlet temperatures, regen chamber back wall temperatures and radiation cooled nozzle extension temperatures. Representative data plots for nozzle extension temperature (Figure V-28), regenerative coolant temperature (Figure V-28), and regenerative chamber heat loads (Figure V-29) have been included in this summary. Additional detailed data is included in the final program report.

11. TEST CONCLUSIONS

The testing conducted at both WSTF and the Bell test facility have adequately demonstrated the performance stability and operating capability of the regeneratively cooled thrust chamber. Test results have proved to be in reasonable agreement with predicted values and in some cases exceeded expectations. Of equal importance to the data obtained, the facility comparison has shown an excellent agreement, even with a relatively large variation in increased conditions (i.e. nozzle area ratio of 15:1 to 76:1), and a resulting confirmation that analytical methods of data presentation are correct.

FIGURE V-28



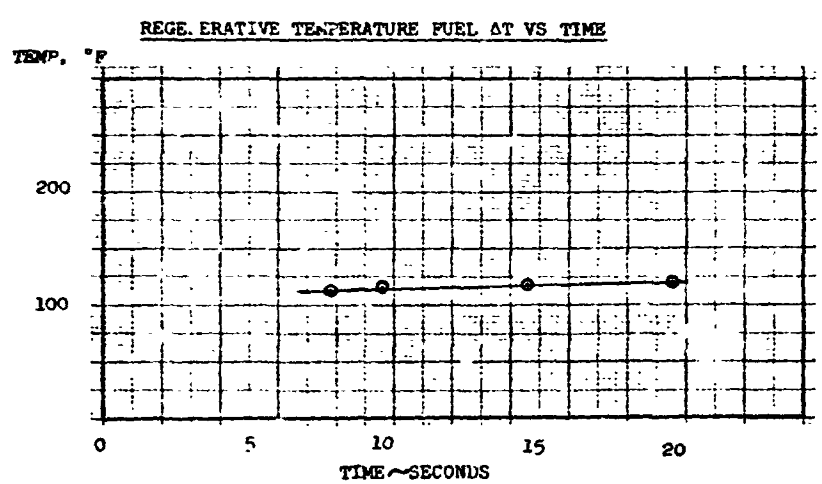
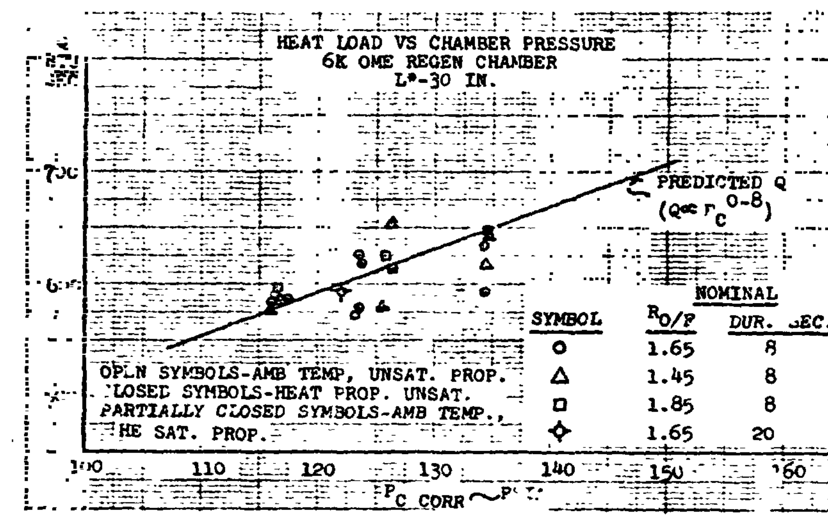
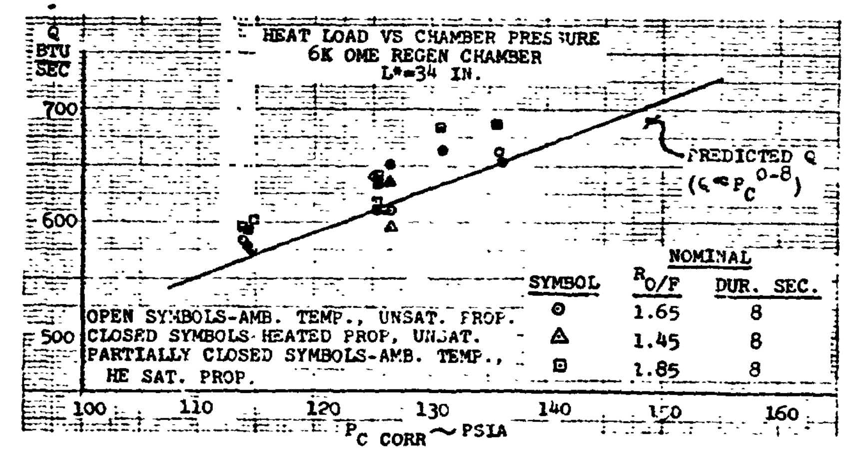


FIGURE V-29





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D. HEAT TRANSFER TUBE TESTS

Generally, the results of this test program showed that the MMH with additive behaved similarly to the clean MMH. A peak nucleate boiling point of between 5.0 and 5.3 Btu/in^2 sec., with the onset of nucleate boiling occurring at forced convection region was between .007 and .009 Btu/in^2 sec. °F. The long duration run seemed only to have the effect of raising the wall temperature somewhat but the peak nucleate boiling heat flux was unaltered. This probably means that an insulating film was deposited on the inside wall that reduced the effective film coefficient very slightly.

Typical Q/A vs inside wall temperature functionalities are shown in Figures V-30, and V-31 for MMH no cycling and MMH with cycling. The additive used with MMH was 1% hexamethyldisilazane, and 1% methylcyanoethylpolysiloxane was added to 50-50.

The 50-50 had a slightly higher peak nucleate boiling, and the additive seemed to raise it a small amount, 5.64 and 6.17 Btu/in^2 sec., respectively, the onset of nucleate boiling was at 2.5 and 2.8 Btu/in^2 sec., and there was no difference in the forced convection film coefficient. The wall temperature in the nucleate boiling regime as a result was about 35°F higher with additive.

Although this test series was very limited, certain tentative conclusions resulted from the test data.

The onset of nucleate boiling with MMH occurred below the reported saturation temperature when the tube was clean or had very little time at temperature. Subsequently, a change appeared where it was hypothesized that a film builds up on the inside of the tube that either insulates the tube wall or prevents decomposition because of passivation or that the fuel does not contact the Hastelloy X or CRES 347.

The cycling or time at temperature had no effect on the peak nucleate boiling heat flux, however, the tube wall temperature was substantially higher, $\approx$ 100°F.

The general effect of the silicone additive to the fuel was not extensive and could well be within the use criteria for any design. Although more detailed information should be generated for point designs, the silicone additive, as a chamber heat depressant remains as a viable propellant additive for the OME application.

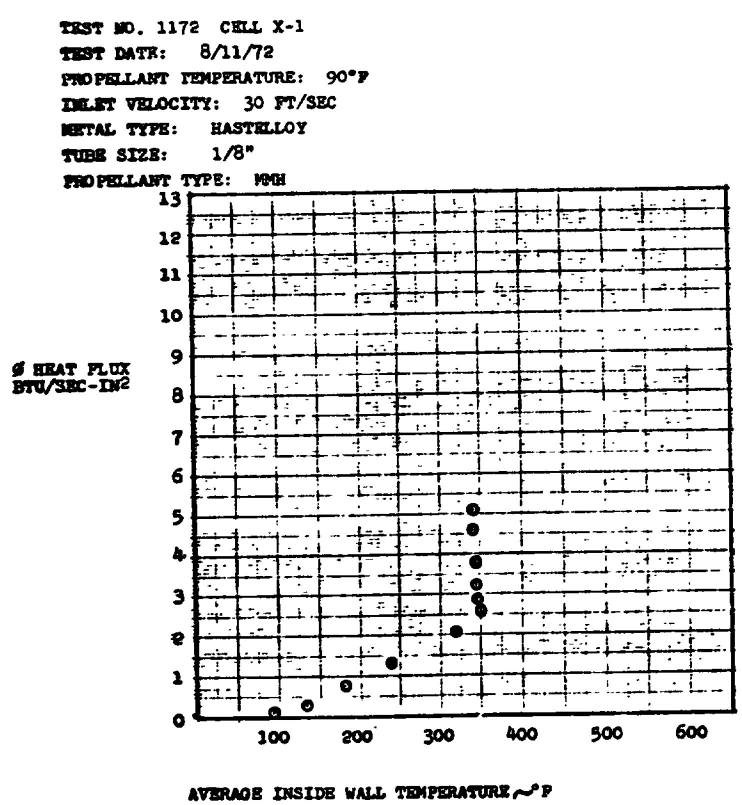


FIGURE 30

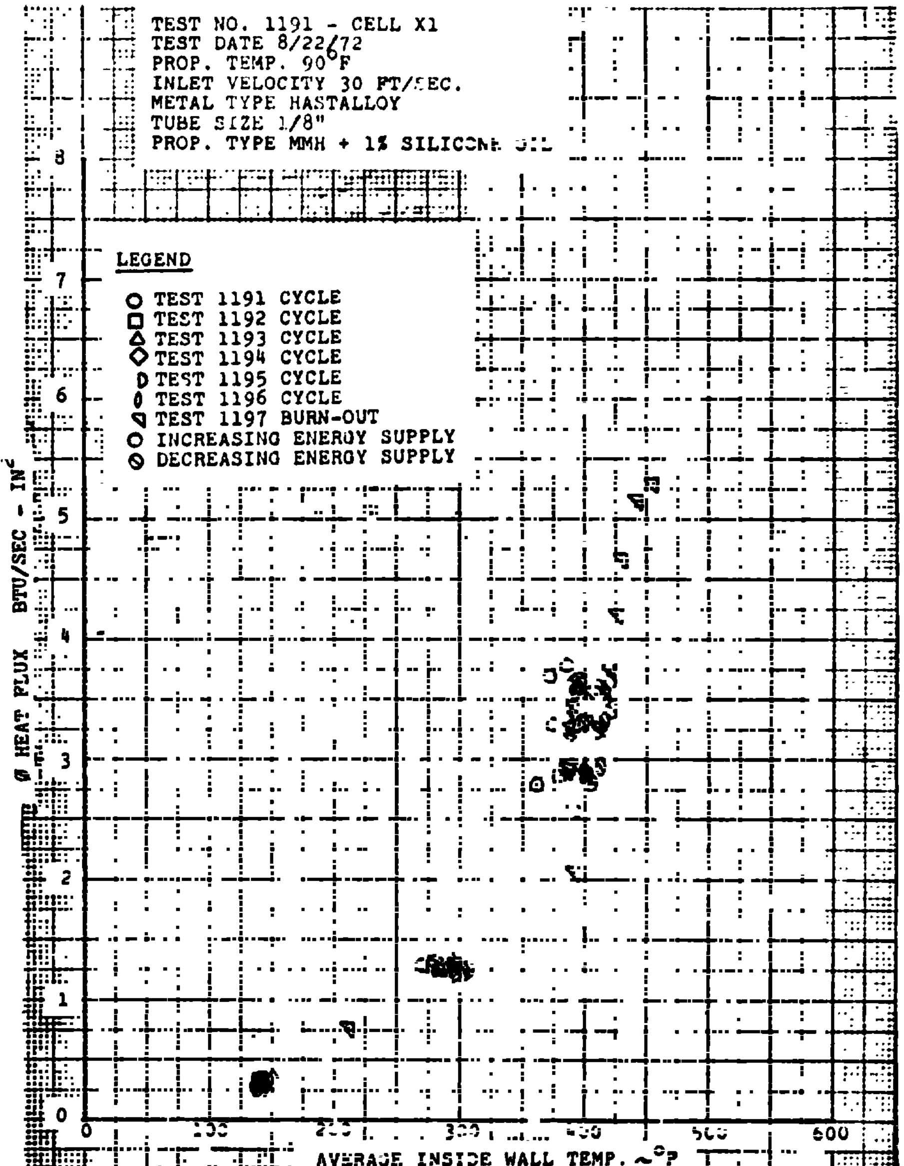


FIGURE 31

E. INJECTOR EVALUATION AND STABILITY TESTS

Four tasks were used to divide efforts related to evaluation and characterization of injector parameters on this contract. The initial test results were related to use on an insulated columbium thrust chamber and the acceptable performance, heat rejection characteristics and stability related to that type of chamber. Time and circumstances subsequently shifted both a concern and effort from the insulated columbium chamber concept to the regeneratively cooled chamber. This change resulted in the requirement for the propellant supply temperature increase and also a reduction in the film coolant flow. During the injector tests the temperature variation was accomplished with preheated propellants. The initial change in film coolant was accomplished with a reduced supply to the film vortex cooling manifold to incorporate a face injected fuel film coolant and the vortex cooling eliminated. The type of testing to be accomplished the various tasks is summated in Figure V-32.

1. HEATED FUEL INJECTOR TESTS

Injector S/N 1A was initially tested with heated fuel to see if the performance of this injector decreased as rapidly as that reported on other related NASA contracts. Data was obtained with both heated fuel and ambient oxidizer and also with heated fuel and heated oxidizer. The test results with heated fuel are shown in Figure V-33. The conclusion of the tests were to show a negligible effect of heated fuel on performance. There was however, an effect on performance as the oxidizer temperature exceeded 100°F with the heated fuel. A performance loss was approximately 2/3% in the region of 105°F oxidizer. The effect of vortex flow variation on heat rejection, obtained during these tests, as shown in Figure V-34.

2. COMBUSTION STABILITY SCREENING TESTS

A series of preliminary tests were made with the stability test hardware using the stainless steel baffled injector S/N 1 to insure the operation of the combustion stability systems installed in the test cell. These tests were completed with a demonstration of five tests of damping from 3.2 to 5.8 M.S. Tests covered a P_c range from 106.6 to 131.7 psia and O/F values from 1.40 to 1.80.

Following the stainless steel injector combustion stability checkout series, the first group of stability tests were made with the 10 inch flat face injector stabilized with acoustic dampers. The test configuration is shown in Figure V-35. The test results are noted as follows:

FIGURE V-32

INJECTOR TEST SUMMARY

TASK	INJECTOR DESIGNATION	TYPE OF INJECTOR	TYPE OF FILM COOLING	TYPE OF TESTS	TYPE OF RESULTS
IX	S/N 1A	Al Flat	Vortex (Regen)	Performance vs propellant temp	O/F @ heated fuel Heat rejected vs vortex flow rate
IX	S/N #1	SS Baffle	Vortex (Insul)	Stability	Stability vs O/F @ P_c
IX	S/N 1A	Al Flat	Vortex (Regen)	Stability	Stability flat face 10 inch diameter
XI	S/N 2	Al Flat	Vortex (Regen)	Injector Charact.	Performance and heat rejection
XII	S/N 2	Al Flat	Vortex (Regen)	Stability	Effect of damper configuration on stability
XIV	S/N 2A	Al Flat	Film (Regen)	Stability	Effect of damper entrance on stability

FIGURE V-33. C* VERSUS TOTAL O/F HEATED FUEL, AMBIENT OXIDIZER % ρ (NOMINAL P_C, O/F) = 2.9

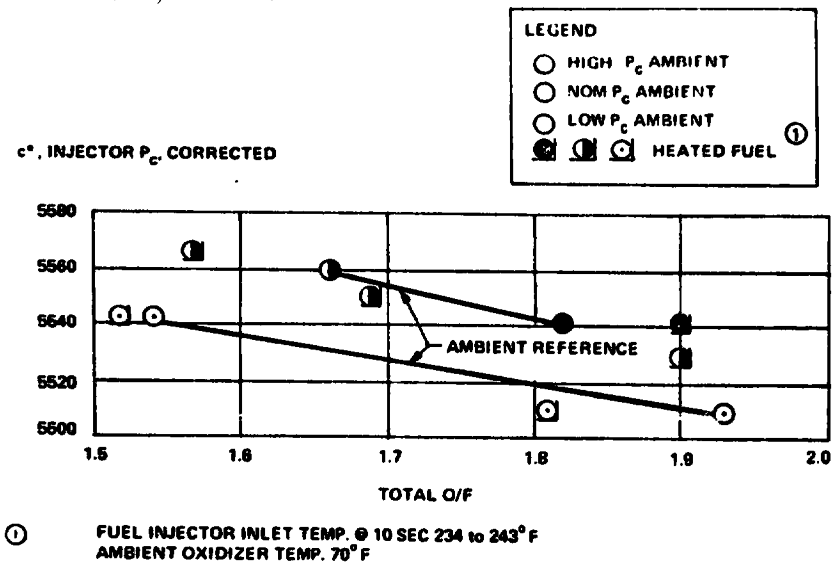
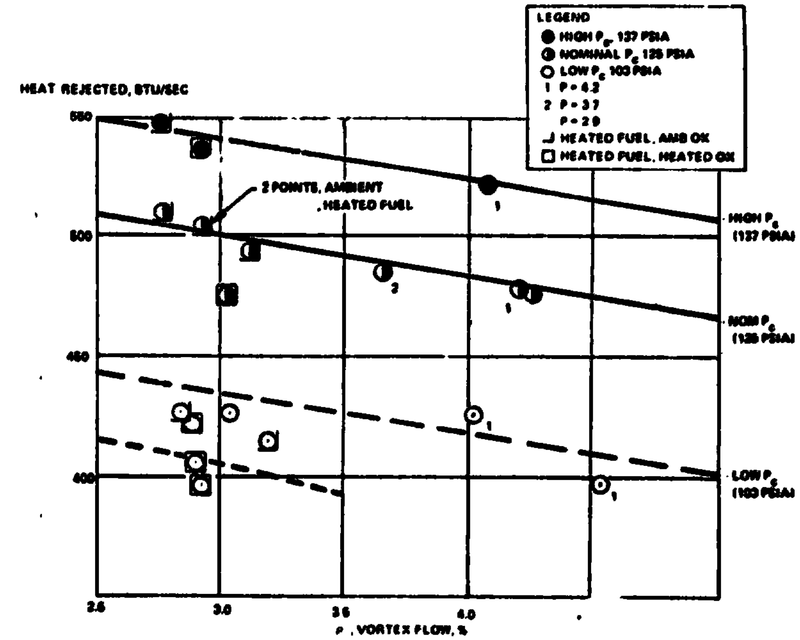
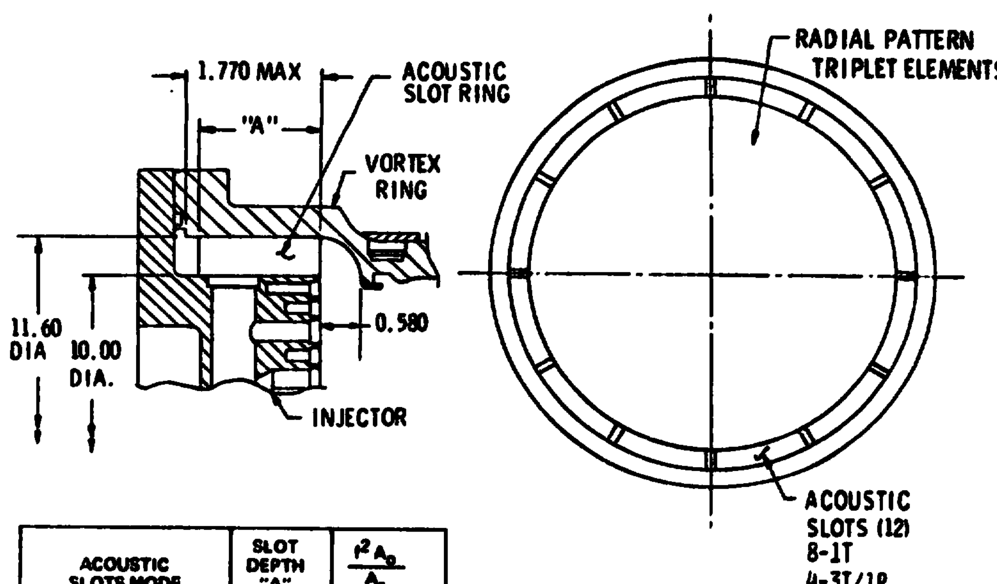


FIGURE V-34. 10 INCH ALUMINUM NO. 1 MOD A HEAT REJECTED VERSUS VORTEX FLOW 10 SEC. DATA





ACOUSTIC SLOTS MODE	SLOT DEPTH "A"	$\frac{\ell^2 A_o}{A_c}$
1ST TANGENTIAL	1.650	1.007
3RD TANGENTIAL OR 1ST RADIAL	0.769	2.34

FIGURE V-35. ALUMINUM INJECTOR ASSEMBLY - DESIGN DATA

Number Of Tests	Number Of Bombs Detonated	Fuel Inlet Temp.	Oxidizer Inlet Temp.	Damp Time
11	16	Ambient	Ambient	2.0 to 3.0 M.S.
2	2	225	70	2.0 M.S.

The positive results of these screening tests is obvious and served as the basis of defining "limit" tests to for further damper evaluation.

3. OME MODEL INJECTOR

This task involved the acceptance of and the delivery to the program of the S/N 2 aluminum injector. The design was originally conducted as a company effort, however, to allow the program the latitude of modification the injector was "sold" to the program.

4. HEATED PROPELLANT INJECTOR STABILITY TESTING

The manner in which these tests were conducted was somewhat unusual. Prior to the test series, temperatures in the cavities were measured. These temperatures were in the order of 3000°F and showed a substantial time was required for the temperature to come to equilibrium (7 to 10 seconds). Consequently, a bomb sequence was incorporated detonating charges at 0.5, 2.0 and 7 seconds representing approximately 1000, 2000 and 3000°F in the acoustic cavity. By bombing over this temperature range, a broad spectrum of conditions relating to the speed of sounc in the damper cavity were covered and produced a large amount of useful data with a modest number of tests.

In retrospect, no effect of temperature in the cavity was really defined. There were only two occurrences of instability noted during this series and these occurrences were with the 1T cavities reduced to 3T depths. Both tests were stable until the bomb was detonated, and neither bomb test recovered after detonation of the first bomb. The test results are summarized in Table V-7. The data is also presented on a mixture ratio/chamber pressure box chart in Figure V-36.

The time temperature recording for the acoustic damper cavity vs time is shown in Figure V-37. The high temperature recorded was at the damper surface; the reduced temperatures were recorded at the center of the deep cavity and bottom of the shallow cavity (Figure V-38). The basic entrance to the cavity is shown in Figure V-39 along with the test chamber installation.

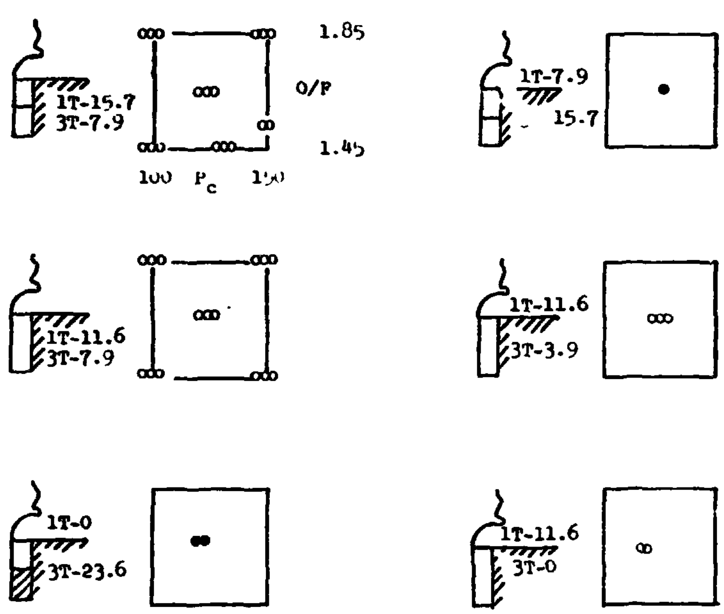
TABLE V-7

TEST SUMMARY

1T	3T	RESULT	NO. TESTS	NO. BOMBS	REMARKS
15.6	7.8	S	4	1	CAVITY TEMP > 3000°F THERMAL BOMB - OLD STYLE
15.6	7.8	S	6	17	NEW BOMBS DAMP < 5 MS
11.6	7.8	S	5	15	DAMP < 5 MS
0	23.4	U	2	3	BOTH TESTS UNSTABLE
7.8	15.6	U	1	3	INITIAL BOMB UNSTABLE TO SHUTDOWN
11.6	3.9	S	1	3	DAMP < 5 MS
11.6	0	S	1	3	DAMP < 5 MS

FIGURE V-36

TASK XII - STABILITY TEST RESULTS
(FUEL VORTEX COOLING)



o STABLE
o MARGINAL
● UNSTABLE

1T - 1ST TANGENTIAL OPEN AREA %
3T - 3RD TANGENTIAL/1ST RADIAL OPEN AREA %

FIGURE V-37

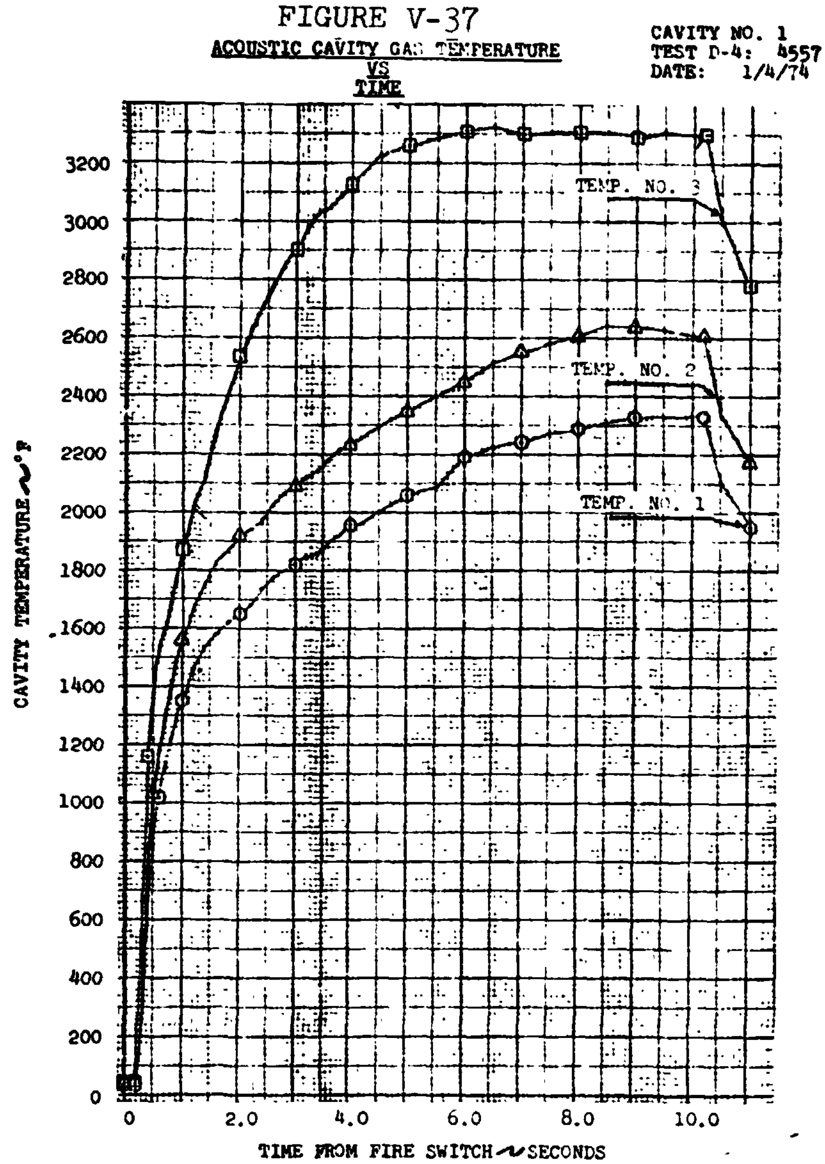


FIGURE V-38

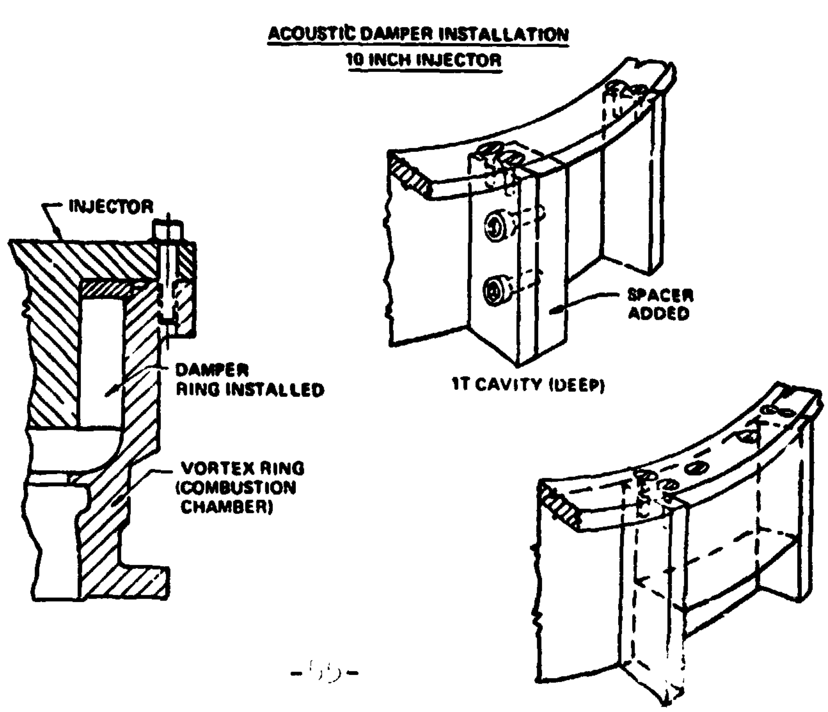
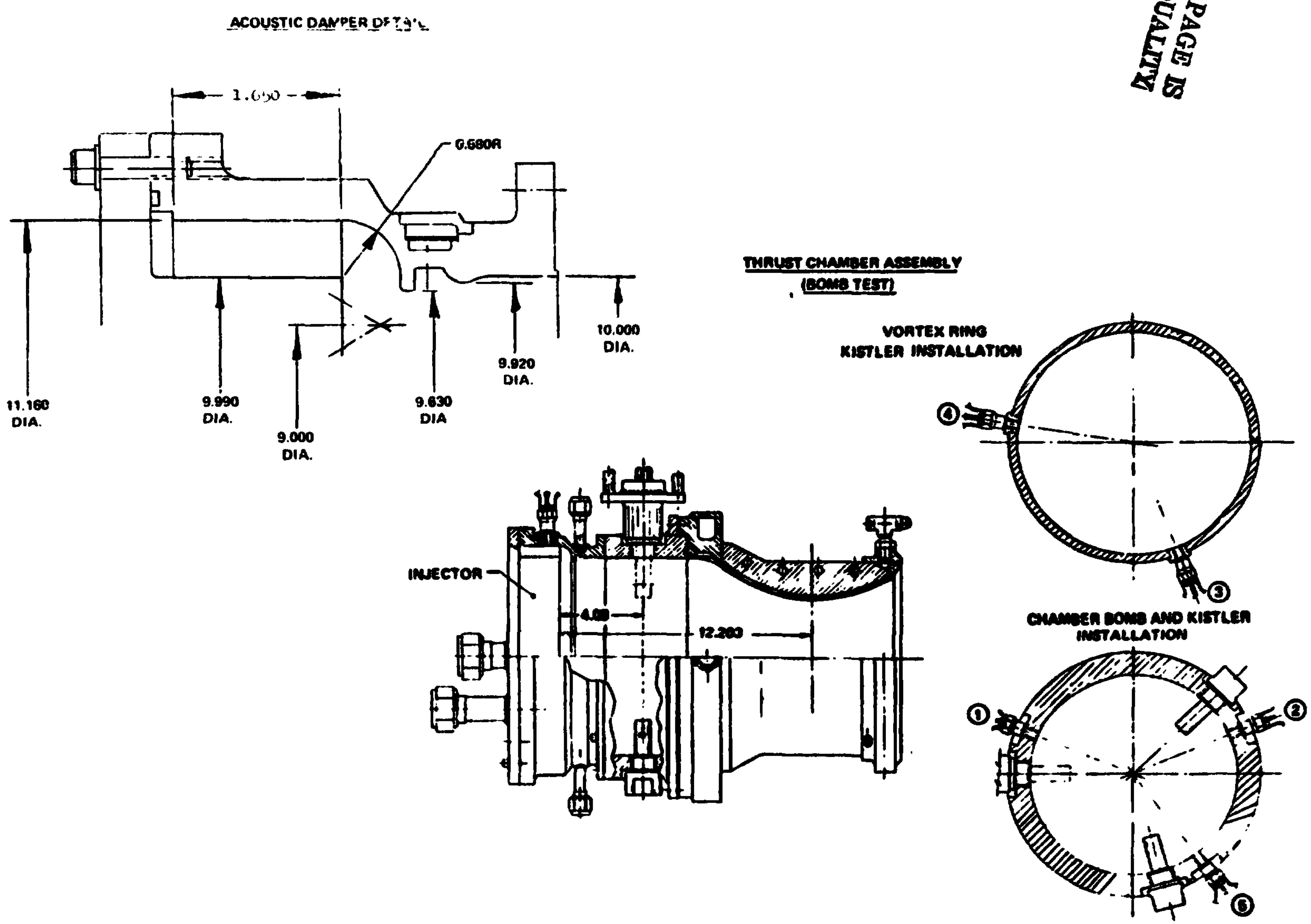


FIGURE V-39


ACOUSTIC DAMPER
0.680R
11.160 DIA.
9.990 DIA.
9.000 DIA.
9.630 DIA
9.920 DIA.
10.000 DIA.
THRUST CHAMBER ASSEMBLY
(BOMB TEST)
INJECTOR
4.08
12.260
VORTEX RING
KISTLER INSTALLATION
4
3
CHAMBER BOMB AND KISTLER
INSTALLATION
1
2
5

The bomb insertion device, to insert the bomb for the 7 second detonation is shown in Figure V-40.

5. TRIPLET INJECTOR DYNAMIC STABILITY VERIFICATION

The product of Task XIV of the program was the evaluation of different acoustic cavity entrance configurations relative to combustion stability. One basic change in the injector was incorporated before these tests. That being the incorporation of fuel film coolant at the injector face and the elimination of the vortex cooling ring. The entrance (and acoustic cavity) configurations tested during this task are shown in Figure V-41. The test results are shown in Figure V-42.

One further product of this task was the development of a low cost moldable cork insulation bomb arrangement. The Insul Cork installation (Figure V-43) was chosen after evaluation of metal detonator, teflon insulation and cork insulation comparison tests. This installation was chosen as being the best combination of least-cost low-damage detonation devices.

One final conclusion was made as a result of these tests, that being the effect of film vs vortex cooling as being undefinable. Temperature measurements in the acoustic cavities showed little, if any, change from the measured cavity temperatures obtained with the vortex cooling arrangement.

F. REDUCED DIAMETER COMBUSTOR

The design rationale for the 8 inch diameter injector included the retention, whereever possible, of the features of the highly successful 10 inch diameter injector. Various tradeoffs were completed to establish triplet element arrangement and to retain the manifolding of the 10 inch unit. The resulting design features are compared to the 10 inch injector in Table V-8. The primary purpose of the evaluation program was to examine performance in the smaller diameter chambers. The performance was considered critical in view of external predictions that combustion with impinging injectors in confined areas would not produce complete combustion. Testing showed the high performance of the original design to be maintained. In the interest of economy, tests were combined on the 8 inch injector evaluation program to produce stability, performance and heat rejection all in one group of tests.

1. PERFORMANCE

The initial evaluation tests were conducted in a 30 L* chamber of approximately 16 inches in length. The test c* results from a chamber pressure measured in the acoustic cavity and corrected to the entrance of the nozzle. The performance assessment has been done using the combustion efficiency (c*).

FIGURE V-40

BOMB INSERTER ASSEMBLY

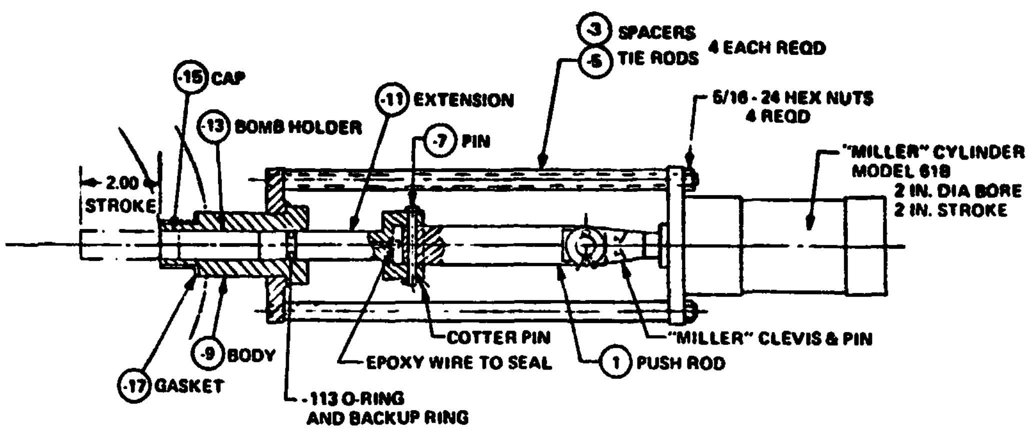


FIGURE V-41

ACOUSTIC DAMPER CONFIGURATIONS

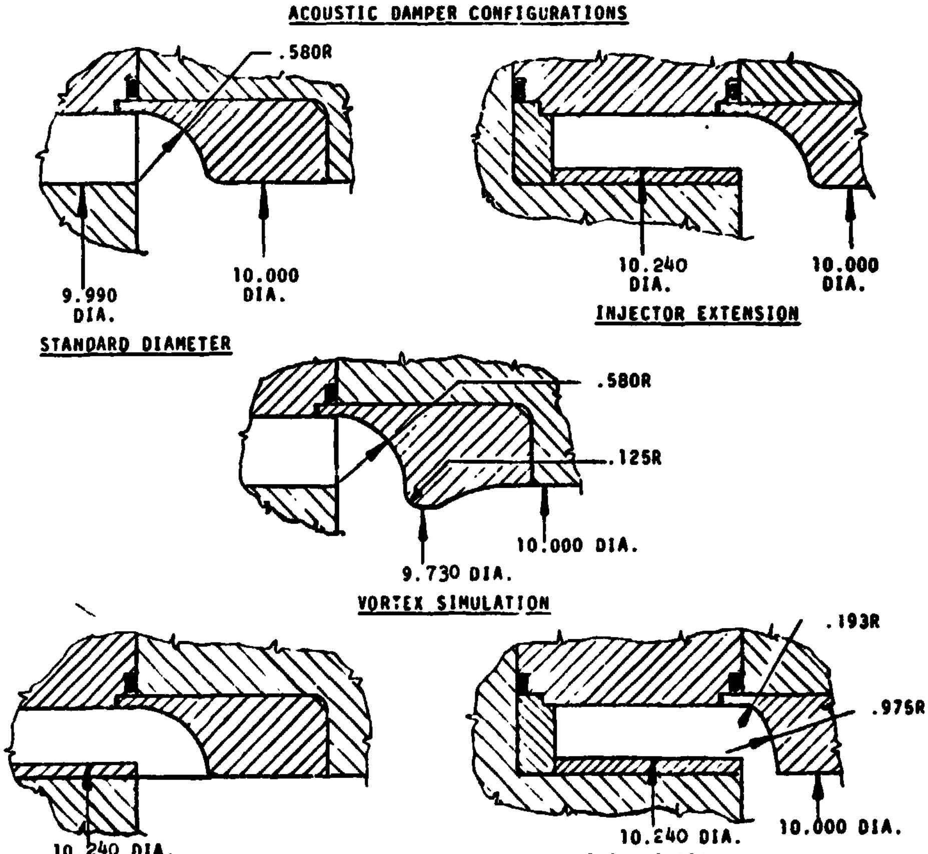
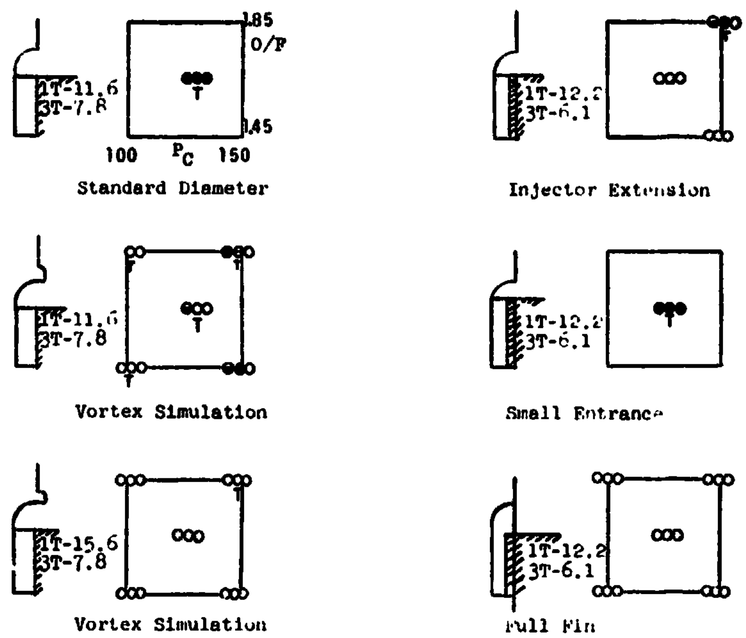


FIGURE V-42

TASK XIV - TEST RESULTS



O	STABLE	1T	-	1ST TANGENTIAL OPEN AREA %
◐	MARGINAL	3T	-	3RD TANGENTIAL/1ST RADIAL OPEN AREA %
●	UNSTABLE	O/T	-	THERMAL DETONATION

FIGURE V-43

INSUL. CORK BOMB
(6.9 GRAIN)

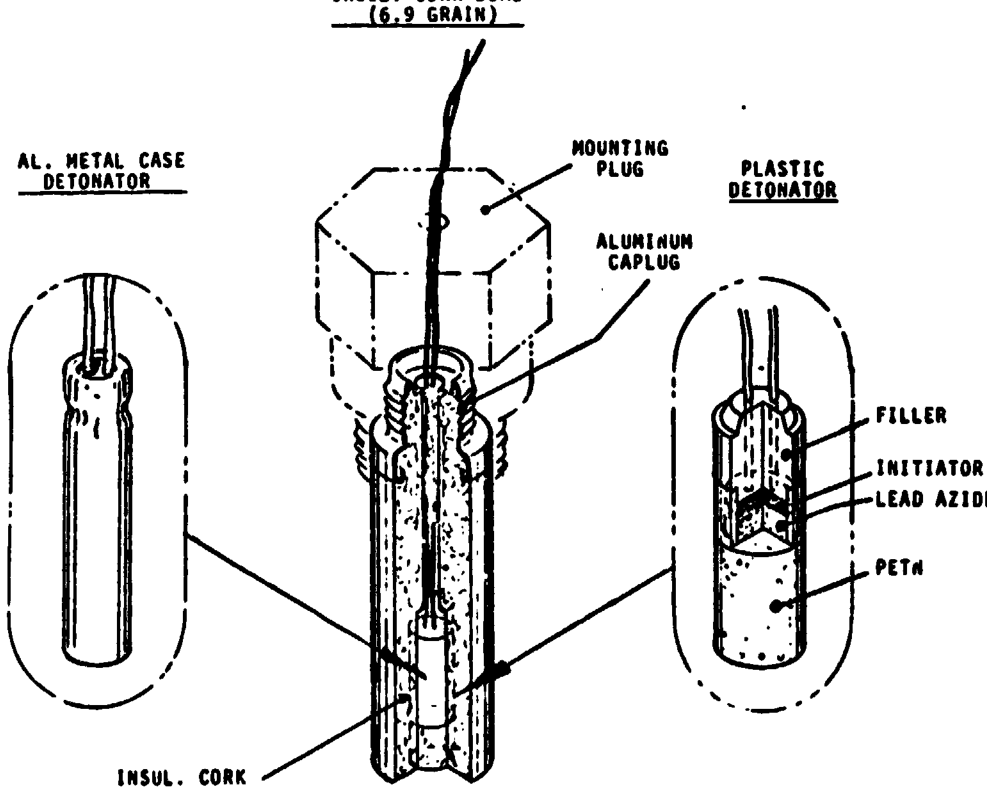


TABLE V-8

INJECTOR COMPARISON

	10" INJECTOR	8" INJECTOR
Total Number of Elements	196	196
Number of Element Rows	7	7
Impingement From Face	.397" (32°)	.397" (28°)
Nominal Face Ring Width	.250"	.220"
Pressure Drop Across Orifice (psi)	45 (Both Sides)	45 (Both Sides)
Primary Oxidizer Orifice	.0492" (148 Holes)	.0492" (148 Holes)
Primary Fuel Orifice	.0295" (296 Holes)	.0295" (296 Holes)
Fuel Film Coolant	.0197" (48 Holes)	.0197" (48 Holes)
Minimum Oxidizer Orifice L/D	3.96	3.96
Minimum Fuel Orifice L/D	5.1	5.1

Although the test sample is limited, the test results were reasonably consistant and indicated that the performance goal was actually exceeded. The original performance goal for this injector was unofficially considered to be approximately 97% with no real "guess" as to the effect of the short chamber. The performance noted was above 98% and only a few (15) feet per second decrease for the reduced length. These results more than justified the design care exercised in the injector. Comparable performance by selected runs are indicated in Table V-9.

TABLE V-9

Injector Performance

Run No.	Chamber Diameter	Injector To Throat Length	P_c (Psia)	O/F (Mixture Ratio)	c*
4610	8.2	15.9	126.6	1.67	5636
4614	8.2	12.8	127.5	1.69	5614
4604	10	12.8	127	1.66	5589

The interesting data from these tests was the lack of performance decrease noted when the shorter chamber was used. Literature information using this chamber had indicated significant performance changes in going to the longer combustor. The conclusion, which must be preliminary in view of the small data base, was that the combustion process had adequate volume in the twelve inch length and the longer chamber was not really required.

The further examination of the 10 inch injector c* should also be made. During the 4609-4619 test series, the injector tested was with fuel film cooling, but also some face damage was observed resulting from bomb shrapnel. The lower than expected performance may have been attributed to the film cooling but it is considered more likely to be a result of performance calibration de-emphasis for the bomb tests and/or orifice distortion due to the shrapnel effects. As a consequence, it is felt that the 5589 ft/sec is pessimistic for the design, and that the previously recorded value for this injector with vortex cooling is more accurate. In that case the 10 inch injector would have a c* near the 5650 ft/sec recorded at WSTF, and would have approximately the same performance recorded for the 8 inch injector.

2. STABILITY TESTS

The acoustic damper configuration tested on the 8 inch diameter injector was incorporated as a direct result of the 10 inch diameter testing conducted in Task XIV. No attempt was made to optimize this damper as neither available time or funding was available for that activity.

The flush fin damper configuration was selected for simplicity considerations. In the 10 inch injector testing, little if any erosion of the flush fins took place. For hardware stability and simplicity, the uncooled arrangement (Figure V-44 was selected).

A diagramatic depiction of the tests conducted with this injector are shown in Figure V-45.

The tests conducted in both the long and short chamber were at conditions found more sensitive at the 10 inch diameter. Since the 10 inch injector bomb sensitivity was found primarily at lean mixture ratios, most of the investigation was at those (oxidizer rich) operating conditions. No sensitivity was found on these bomb tests.

The tests were also examined for low frequency instability and for "football" type starts. The "football" type start (with approximately 500 hz frequency), was noted on test 4617 D4 where a chamber pressure of less than 100 psia occurred. These oscillations are typical of limiting combination of operating conditions such as lean (mixture ratio) operation and low chamber pressure. Both conditions produce a low fuel pressure drop which has previously been limited to about 23 psi. This test produced a fuel pressure drop of approximately 20 psi and the start oscillations resulted. The oscillations damped on this test and were not observed on other tests which maintained a higher fuel pressure drop.

3. NOZZLE HEAT REJECTION

The water cooled nozzle data for both the 10 inch and 8 inch diameter injectors were examined in order to ascertain the effects on the local heating rates. In order to compare the results, the measured total heating rate was used to predict the heat flux at the throat station and this value was then corrected to the nominal chamber pressure and mixture ratio.

The ratio of the heat flux at the throat to the total heating rate was found using theoretical gas side heat transfer coefficients. The ratio was found to be 0.006308 ($1/in^2$) for the 10 inch injector and 0.008044 ($1/in^2$) for the 8 inch injector. In order to correct for chamber pressure and mixture

FIGURE V-44

ACOUSTIC CAVITY TEST CONFIGURATION

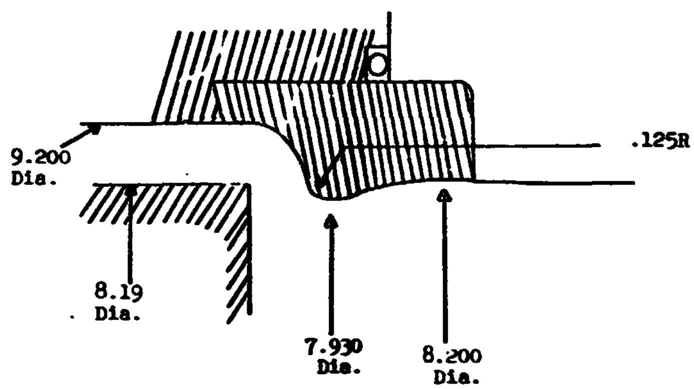
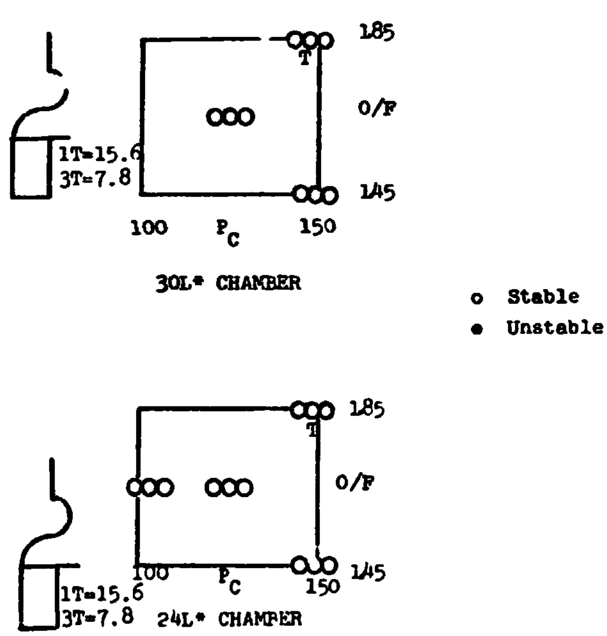


FIGURE V-45

TASK XV

STABILITY TEST RESULTS



ratio (which effects the percent barrier flow), a previously developed regression fit of test data was used. The equation is:

$$Q \propto P_c^{.8} / \rho^{.186}$$

This equation was used to correct all the test data to the nominal chamber pressure of 125 psia and 2 percent barrier at a mixture ratio of 1.65. The following table presents the results of the testing.

Injector Diameter	Barrier	Chamber Length	Throat System Heat Flux-Average	Number Of Tests Averaged
10 in.	Vortex	12 in.	3.095 Btu/in sec	7
10	Axial	12	3.253	18
8	Axial	12	3.306	4
8	Axial	16	3.325	3

It is seen that for the axial injection of the barrier the heat flux was about 5 percent higher than the vortex barrier for the 10 inch diameter injector. The 8 inch injector showed slightly higher heat flux; about 2 percent more than for the 10 inch injector.

VI. CONCLUSIONS

The design analysis and subsequent demonstration testing conducted on this contract were extensive and encompassed many elements of rocket engine technology. The successful demonstration of both computor techniques and operating designs resulted in the unique accomplishment of demonstrating, by test, two distinctly different cooling schemes which could be applied to the the OME mission. The successful demonstration of both cooling schemes was attributed to the practical inputs of the analytical studies, as well as the well developed empirical criteria for the triplet element injector and vortex cooling arrangements. On the basis of results from this program, the following conclusions are made.

A complete computor program was assembled to predict engine data for trade off studies on this program. This computor program and input data was confirmed in testing of the subsequent engine designs although, in some cases, substantial simplification of analytical techniques were made. The logical simplifications incorporated into the computor program were made so that the extensive data required for the tradeoff studies could be accumulated within the limited resources of the contract. This data base served as the design selection criteria, subsequently substantiated by test of the selected engines.

The uniformity of the vortex method of fuel film coolant was considered to be a major contributor in the successful demonstration of the Insulated Columbium Thrust Chamber. The successful demonstration of this simple metal shell combustor represents a significant contribution in the technology of refractory metal combustion chambers. The program test results provided data for an engine which could be readily constructed to provide a performance of some 310 seconds I_{sp} in an OME application.

The triplet injector concept also provided the basis for a regeneratively cooled chamber operating at a level of 317 seconds I_{sp} and a reduced diameter combustor. Limit stability testing of the 10 inch injector with heated propellants indicated a substantial stability margin and a capability of operation under unusual conditions with acoustic slot combustion stability devices. The altitude performance demonstration for the regeneratively cooled engine provided the information to confirm the performance and operating requirements. The triplet injector was also demonstrated by an 8 inch diameter injector to show a high level of performance achievement and stability. The testing performed confirmed the selection of injection parameters used and also indicated a very much lower sensitivity to chamber length than originally anticipated. The results of these tests were sufficiently encouraging to recommend further testing.

VII. RECOMMENDATIONS

The complete success of the first item tests conducted with eight inch diameter injector provide the basis for recommending a more detailed test program for design validation. The following items would provide the basis for this validation.

Perform altitude performance tests at WSTF with the injector available, and in a NASA provided regeneratively cooled thrust chamber. This data confirmation would provide "certainty" of performance as well as transient operation data.

Conduct "limit" combustion stability testing with the injector and a variety of acoustic cavity combinations. This information would provide a basis of final acoustic cavity sizing, and an empirical comparison with the larger diameter previously tested combustor.

Fabricate a stainless steel injector of the 8.2 inch design. The construction of such an injector would complete the identification of the 8.2 inch injector for performance, stability and fabrication.