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WEAPON SYSTEM 107A-2  
PRODUCT ENGINEERING PROGRAM  
Contract AF 04(694)-212/SA3

Report 212/SA3-2. 2-M-1

16 July 1963



**AEROJET-GENERAL CORPORATION**  
SACRAMENTO, CALIFORNIA

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WEAPON SYSTEM 107A-2

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PRODUCT ENGINEERING PROGRAM

Contract AF 04(694)-212/SA 3

1 June through 31 June 1963

Prepared by

AEROJET-GENERAL CORPORATION  
Liquid Rocket Plant  
Sacramento 9, California

Prepared for

BALLISTIC SYSTEMS DIVISION  
AIR FORCE SYSTEMS COMMAND  
Norton Air Force Base, California

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**AEROJET-GENERAL CORPORATION**  
A SUBSIDIARY OF THE GENERAL TIRE & RUBBER COMPANY

FOREWORD

This report is the first in a series of monthly reports prepared in accordance with AFBM Exhibit 58-1 and submitted in partial fulfillment of Contract AF 04(694)-212, Supplemental Agreement No. 3.

Direction for contract performance is provided by C. L. D'Ooge, Program Manager, Research and Advanced Technology Division, Liquid Rocket Plant.

The contract for the continuation of the Product Engineering Program received by Aerojet-General in mid-June, is made up of four projects:

<u>PROJECT TITLE</u>	<u>PROJECT ENGINEER</u>
1. Coated Metallic Thrust Chambers	S.E. Adair, Jr.
2. Expandable Nozzle	D.M. Green
3. Combustion Instability Scaling Concepts	F.H. Reardon
4. Ablative Thrust Chamber Feasibility	H.H. Mueggenburg and T.A. Huges

Three of the projects are a continuation of the work conducted previously under Contract AF 04(647)-652/SA 33, and Project C is new work. The three other projects that were conducted under the project engineering contract are currently being funded under separate contracts. These are:

Metalized Thixotropic Propellants  
VāPak Pressurization  
Combustion Instability Periscope Development

The primary effort expended in June has been on detailed program planning and budget preparation. Several discussions with personnel of the Liquid Propulsion Laboratory, Edwards, California, have been held in the process of developing these plans.

TABLE OF CONTENTS

	<u>PAGE NO.</u>
I. COATED METALLIC THRUST CHAMBERS	I-1
A. INTRODUCTION	I-1
B. PROGRESS DURING REPORT PERIOD	I-1
C. NEXT REPORT PERIOD	I-2
II. EXPANDABLE NOZZLES	II-1
A. PROGRESS DURING REPORT PERIOD	II-1
B. NEXT REPORT PERIOD	II-2
III. COMBUSTION INSTABILITY SCALING CONCEPTS	III-1
A. INTRODUCTION	III-1
B. PROGRESS DURING REPORT PERIOD	III-2
C. NEXT REPORT PERIOD	III-3
IV. ABLATIVE THRUST CHAMBERS	IV-1
A. INTRODUCTION	IV-1
B. PROGRESS DURING REPORT PERIOD	IV-1
C. NEXT REPORT PERIOD	IV-3
V. PROGRAM REPORTING	V-1

TABLE LIST

	<u>TABLE NO.</u>
Coated Metallic Thrust Chambers Milepost Chart	I-1
Test Schedule for Coated Metallic Thrust Chamber Tests	I-2
Expandable Nozzle Milepost Chart	II-1
Combustion Instability Scaling Concepts Milepost Chart	III-1
Ablative Thrust Chambers Feasibility Milepost Chart	IV-1
Program Reporting Milepost Chart	V-1

FIGURE LIST

	<u>FIGURE NO.</u>
Uncooled Combustion Chamber, Cylindrical Throat Configuration	IV-1

I. COATED METALLIC THRUST CHAMBERS

A. INTRODUCTION

1. Purpose

The objective of this project is to develop a reliable thermal barrier capable of surface temperature operation above 3300°F for use on regeneratively-cooled thrust chambers employing Aerozine 50/N<sub>2</sub>O<sub>4</sub> propellant. Thermal barrier film-coating is a technique for improving liquid rocket engine performance by reducing film-cooling flow rates.

2. Approaches

The development of the thermal barriers will be accomplished in the following ways:

a. Full-scale testing of coated YLR91-AJ-5 engines (Titan II second stage) utilizing fuel film-cooling flow rates of 5% or less.

b. Laboratory thermal shock, thermal conductivity, erosion-corrosion, and metallographic testing.

B. PROGRESS DURING REPORT PERIOD

The YLR91-AJ-5 thrust chamber which was fabricated and coated on the previous product engineering contract\* has been shipped to Test Area C to undergo three tests at the planned 17% fuel film-cooling flow rate for duration of 5, 10, and 15 sec. A design study is underway to determine the best means for modifying an un baffled injector to give 5% fuel film cooling for all subsequent tests.

\*AF 04(647)-652, SA33

I, B, Progress During Report Period (cont.)

Table I-1 is the program milepost chart, and Table I-2 graphically presents the test schedule.

C. NEXT REPORT PERIOD

The thrust chamber coated on the previous contract will be test fired at the normal film cooling flow rate of 17%. A 5% cooling injector design will be made and hardware modifications started. The laboratory equipment will be prepared and modified as necessary for specimen testing and full-scale coating. A special fixture for photographic coverage of the coating between full-scale tests will be made.



**TABLE I-2**  
**TEST SCHEDULE FOR COATED METALLIC THRUST CHAMBER TESTS**

LINE NO	CURRENT CHANGE	Coated Metallic Thrust Chamber	PRESENT SCHEDULE			P R I O R I T Y	CY 63												CY 64					LINE NO		
			DAY	MO	YR		J	J	A	S	O	N	D	J	F	M	A	M								
1		Phase I - Analysis and Data Evaluation																								1
2		A. Heat Transfer																								2
3		1. Initiate	1	6	63	▲																				3
4		2. Complete	31	5	64																			▲		4
5		B. Thermal Stress																								5
6		1. Initiate	1	6	63	▲																				6
7		2. Complete	31	5	64																			▲		7
8		C. Erosion and Corrosion																								8
9		1. Initiate	1	6	63	▲																				9
10		2. Complete	31	5	64																			▲		10
11		D. Data Evaluation																								11
12		1. Initiate	1	6	63	▲																				12
13		2. Select coating for chamber 2	1	8	63			▲																		13
14		3. Select coating for chamber 3	9	10	63					▲																14
15		4. Select coating for chamber 4	23	12	63									▲												15
16		5. Select coating for chamber 5	17	3	64																		▲			16
1		6. Complete	30	5	64																			▲		1
2		E. Specification for Best Thermal Barrier																								2
3		1. Initiate	17	3	64																		▲			3
4		2. Complete	30	5	64																			▲		4
1		Phase II - Laboratory Investigations																								1
2		A. Initiate Laboratory Investigation	1	6	63	▲																				2
3		1. Thermal Shock Tests	1	6	63	▲																				3
4		2. Thermal Conductivity Tests	1	6	63	▲																				4
5		3. Erosion-Corrosion Tests	1	6	63	▲																				5
6		4. Coating Application Technique Dev.	1	6	63	▲																				6
7		5. Metallography	1	6	63	▲																				7
8		B. Complete Laboratory Investigation	31	5	64																			▲		8
9																										9
10																										10
11																										11
12																										12
13																										13
14																										14
15																										15
16																										16



II. EXPANDABLE NOZZLES

1. Purpose

The purpose of this project is to develop and test efficient packaging of high expansion ratio nozzle exit cones for rocket propulsion systems.

2. Approach

The program objectives will be accomplished by a three-phase (theoretical, fabrication, and experimental) effort. Work will be concentrated on the simulation of the nozzle conditions of the Titan II second-stage engine with both metallic and non-metallic nozzle expansion cone skirt extensions. Thrust vectoring will be investigated on an existing engine mounted on a two-axis thrust stand. A series of flight simulation tests in a wind tunnel will be conducted with scale models using various forebody shapes.

A. PROGRESS DURING REPORT PERIOD

The design has been initiated for the engine, nozzles, and deployment systems to be tested at the Arnold Engineering Development Center (AEDC). Because of the availability of the new J-4 tunnel at AEDC, the tentative approach is to use an LR-91-5 thrust chamber assembly that will operate at full Titan II conditions with nozzles up to an area ratio of 200:1.

Preliminary planning meetings for the simulated flight test program have been held with personnel from the Aeronautical Division of the Downey Plant. The locations being considered for these tests are the Jet Propulsion Laboratory, Cornell Aero Labs, and Arnold Engineering Development Center.

II, A, Progress During Report Period (cont.)

The test stand design and modifications for the thrust vector control program have been initiated for the Aerojet-General Proving Grounds at Azusa.

B. NEXT REPORT PERIOD

The activities that have been initiated will continue and a model nozzle of 12,000-lb thrust to demonstrate deployment of gas pressure, and structural stabilization will be completed.

Plans will be finalized for wind tunnel tests for external flow investigation and the AEDC J-4 tunnel for simulated altitude firings of large nozzles at high chamber pressure. Lark-engine material tests will be conducted.



### III. COMBUSTION INSTABILITY SCALING CONCEPTS

#### A. INTRODUCTION

##### 1. Purpose

The Combustion Instability Scaling Concept Program is an effort to predict analytically the longitudinal modes of instability of a subscale rocket engine; to determine from testing of this engine the longitudinal mode stability limits; to compare these results; and finally to utilize the results for determining the occurrence of transverse modes of instability for larger rocket engines.

##### 2. Approaches

This program is concerned with the stability of rocket engines as it is affected by chamber length, chamber pressure, and propellant mixture ratio. An empirical analysis will be attempted to determine the axial combustion distribution of this engine, which is required by the theoretical analysis.

For each test, a chamber pressure and a mixture ratio will be specified and held constant throughout the test. Once the thrust chamber has reached steady-state operating conditions, the chamber length will be increased during the test to study the stability effects resulting from the variation in chamber length. Two firings will be required for each operating point to keep the duration approximately 5.0 sec for each test. For the next test, the chamber pressure and/or the mixture ratio will be changed, the motor will be returned to its initial chamber length (a length to diameter ratio approximately equal to unity), and the test series continued.

The principle of the variable chamber length will be used for the combustion distribution tests along with the introduction of a water-cooled aerofoil

III, A, Introduction (cont.)

into the chamber at the exit plane to determine the energy distribution. This foil will measure nine temperatures and eight stagnation pressures at the leading edge and eight static pressures on the bottom of the foil.

B. PROGRESS DURING REPORT PERIOD

1. Thrust Chamber Assembly Design

Preliminary design and analysis was begun on the concept. The 6.0-in. diameter, cylindrical thrust chamber assembly to be used for this test series will operate within a nominal thrust range of 5000 to 10,000 lb at sea level. The chamber will be water-cooled and will operate for durations of 4.0 to 5.0 sec for each test. The hypergolic propellant combination for this test series will be nitrogen tetroxide and Aerozine 50.

The injector pattern will be the central portion of the 2SIN-0 pattern which was used on the YLR91-AJ-5 second-stage engine of Titan II. This unbaffled injector has a like-on-like impingement spray pattern.

Instead of the conventional one-element rocket exhaust nozzle, a zero-length nozzle will be utilized. This is a flat plate across the exit of the chamber with equally sized, equally spaced orifices with the sum area equivalent to the area of a conventional one-element nozzle. The purpose of this zero-length nozzle is twofold. First, the flat plate across the exit improves the acoustic reflections because of the better definition of the reflection plane when compared to that of a standard exhaust nozzle. Second, the many orifices ensure that the cross section of the exhaust gas flow pattern at any one plane is relatively uniform with respect to the gas dynamics and gas properties.

III, B, Progress During Report Period (cont.)

The combustion chamber will be rigidly mounted to a support frame with recirculating ball bearing race assemblies on two of the supports, and hydraulic actuation cylinders connected to the others. Axial movements will be indicated by linear potentiometers. The hydraulic cylinders will move the combustion chamber during the test from the minimum chamber length to the maximum chamber length position. The injector will be restrained from the front of the chamber. Sealing between the injector and the chamber wall will be accomplished by either a Harrison K-seal or a Wiggins Bar-X-seal.

The chamber length will always be increased during testing to assure that the injector to chamber seals will be in progressive contact with a clean, cool surface rather than a surface that has been pitted, heated or charred by exposure to the exhaust gases.

2. Instrumentation

Two flush-mounted, water-cooled transducers will be incorporated in these tests to measure the high-frequency pressure oscillations in the chamber. One will be mounted in the exit plane of the chamber and the other will measure pressure oscillations at the injector face; therefore, the locations of these transducers will afford measurements at the boundaries of the combustion chamber. These transducers are capable of measuring frequencies up to 10,000 cps with good fidelity.

C. NEXT REPORT PERIOD

Preliminary design of the thrust chamber assembly will be continued and detail design will be initiated. Analysis efforts will begin on the theoretical calculations of instability zones.

III, B, Progress During Report Period (cont.)

The combustion chamber will be rigidly mounted to a support frame with recirculating ball bearing race assemblies on two of the supports, and hydraulic actuation cylinders connected to the others. Axial movements will be indicated by linear potentiometers. The hydraulic cylinders will move the combustion chamber during the test from the minimum chamber length to the maximum chamber length position. The injector will be restrained from the front of the chamber. Sealing between the injector and the chamber wall will be accomplished by either a Harrison K-seal or a Wiggins Bar-X-seal.

The chamber length will always be increased during testing to assure that the injector to chamber seals will be in progressive contact with a clean, cool surface rather than a surface that has been pitted, heated or charred by exposure to the exhaust gases.

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C. NEXT REPORT PERIOD

Preliminary design of the thrust chamber assembly will be continued and detail design will be initiated. Analysis efforts will begin on the theoretical calculations of instability zones.

TABLE III-1  
COMBUSTION INSTABILITY SCALING CONCEPTS MILEPOST CHART

LINE NO	CHANGE	PRESENT SCHEDULE	R I O R												LINE NO		
			DAY MO YR			Calendar Year 1963											
			Calendar Year 1964														
			J	J	J	A	A	S	O	N	D	J	F	M	A	M	
1		Phase I--Engineering Design															
2		A. Concept Completed															
3		B. Predesign Initiated															
4		Completed															
5		C. Final Design Initiated															
6		1. Mount Completed															
7		2. Thrust Chamber Completed															
8		3. Injector Completed															
9		4. Actuation System Completed															
10		Phase II--Fabrication Initiated															
11		1. Mount Completed															
12		2. Thrust Chamber Completed															
13		3. Injector Completed															
14		4. Actuation System Completed															
15		5. Calibration Completed															
1		Phase III--Testing Initiated															
2		1. Test Stand Build-Up Completed															
3		2. Calibration Firings Completed															
4		3. Test Program Completed															
5		Phase IV--Analysis Initiated															
6		1. Calculation of Theoretical															
7		Instability Zones Completed															
8		2. Completion of Comparison Between															
9		Theoretical Zones and Test Results															

Table III-1

IV. ABLATIVE THRUST CHAMBERS

A. INTRODUCTION

1. Purpose

The purpose of this project is to demonstrate the feasibility of large ablatively cooled thrust chambers for high performance, liquid rocket engines operating at chamber pressures up to 300 psia at an appropriate mixture ratio for extended durations.

2. Approaches

Six thrust chambers fabricated under a previous contract (AF 04(647)-652/SA33) will be used with injectors developed on the Apollo service module engine program. A water-cooled transition section is required. The thrust chambers utilize the building block concept wherein molded, reinforced plastic rings are bonded together within a thin-walled steel shell (Figure IV-1). They are described in Report 652/SA 4-2.2-F-1, Volume 1, dated 28 June 1963.

B. PROGRESS DURING REPORT PERIOD

The pacing item of the ablatively chamber feasibility program is the water-cooled transition section which will allow the ablatively chamber to be fired with an Apollo injector. The heat transfer analysis and design of this transition section has been initiated and should be completed in July.

The test history of the existing Apollo injectors is currently being investigated to determine a possible candidate injector that will satisfy the necessary technical requirements of the ablatively chamber feasibility program.

IV, B, Progress During Report Period (cont.)

Two additional steel shells for the ablative chambers are being fabricated. The additional shells will serve two purposes: (1) if one of the shells designated for reuse were damaged on the first firing, the firing of the last two chambers would be delayed, and (2) the final assembly of the last two chambers can proceed without delays caused by waiting for shells to become available. The design of these two steel shells has been revised to allow for the diametrical age that was experienced on the original four shells during welding. The end flanges of the shells were strengthened and simplified to reduce out-of-phase distortion, and the method of welding has been changed to achieve this end. The ablative chambers are being completed as necessary. The chambers that have been fabricated are:

<u>Part No.</u>	<u>Chamber Material</u>	<u>Throat Material</u>
284528-9	Silica roving-modified phenolic resin.	Same as chamber. Cylindrical throat configuration.
278120-149	Silica roving-modified phenolic resin.	Pyrolytic deposit of tantalum carbide on graphite. Asbestos phenolic insulation.
278120-259	Asbestos-phenolic.	ATJ graphite. Asbestos phenolic insulation.
278120-189	Asbestos-phenolic.	Pyrolytic deposit of tantalum carbide on graphite. Asbestos phenolic insulation.
284528-29	Silica roving-elastomer modified phenyl silane resin.	Same as chamber. Cylindrical throat configuration.
278120-219	Silica roving-modified phenolic resin.	ZTA graphite. Asbestos phenolic insulation.

IV, B, Progress During Report Period (cont.)

Specific Identification of Materials

<u>Material</u>	<u>Manufacturer</u>
Silica roving-modified phenolic resin, FM 5073	U.S. Polymeric
Asbestos phenolic, Thermomatt 193	Johns-Manville
Silica roving-elastomer modified phenyl-silane resin, X-2017	U.S. Polymeric
Pyrolytic tantalum carbide coating on graphite	High Temperature Materials, Inc.
ATJ graphite and ZTA graphite	National Carbon Company

The first chamber (P/N 284528-9) is ready for immediate firing. The second, third, and fourth assemblies listed are installed in their metal shells and require only interior contouring before firing. Because the 91-3 injector will not be used in the ablative chambers program as originally planned, the recessed lip on the forward end of the ablative chambers will no longer be required and will be removed from the applicable drawings.

Injector checkout tests are currently scheduled during October and November with ablative chamber testing immediately following during December and January. Table IV-1 shows the milestone schedule for the ablative chamber project.

C. NEXT REPORT PERIOD

Upon completion of the transition section design, fabrication orders will be initiated. The most promising Apollo injector will be obtained and reworked if necessary to be available for checkout tests with the transition section.

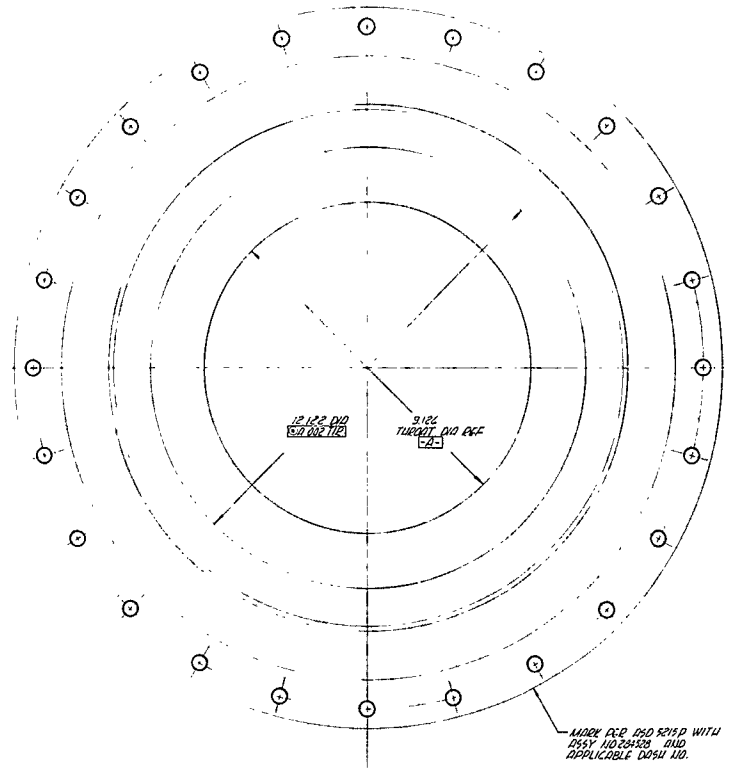
TABLE IV-1

ABLATIVE THRUST CHAMBERS FEASIBILITY MILEPOST CHART

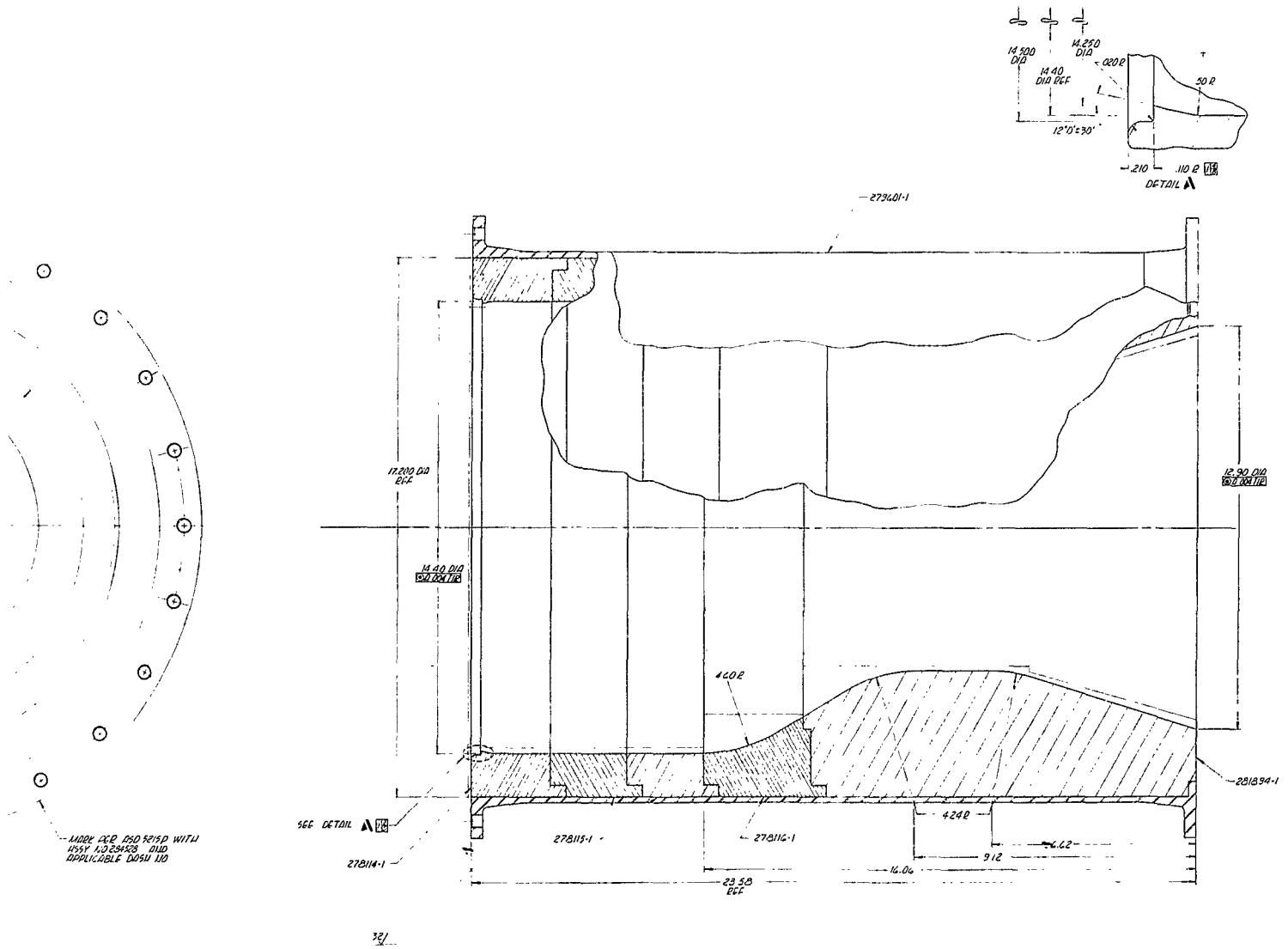
LINE NO.	CURRENT CHANGE	PRESENT SCHEDULE			P R I O R I T Y	Calendar Year 1963							Calendar Year 1964					LINE NO.
		DAY	MO	YR		J	J	A	S	O	N	D	J	F	M	A	M	
1	Phase I - Test Preparation																	1
2	A. Design of Ablative Chamber				A													2
3	B. Fabrication of Ablative Chamber																	3
4	1. Chamber 284528-9				A													4
5	2. Chamber 278120-14	31	8	63					▲									5
6	3. Chamber 278120-259	31	8	63					▲									6
7	4. Chamber 278120-189	31	8	63					▲									7
8	5. Chamber 284528-29	31	8	63					▲									8
9	6. Chamber 278120-219	31	8	63					▲									9
10	C. Design of Metal Shell				A													10
11	D. Fabrication of Metal Shell	31	8	63					▲									11
12	E. Design of Transition Section	30	9	63						▲								12
13	1. Heat Transfer Analysis	31	8	63					▲									13
14	F. Fabrication of Transition Section	30	10	63							▲							14
15	G. Selection of Injectors	31	8	63					▲									15
16	H. Injector Fabrication	30	9	63						▲								16
1	I. Uncooled chamber Acquisition				A													1
2	J. Selection of Instrumentation	30	9	63						▲								2
3	K. Selection of firing Parameters	15	11	63							▲							3
4	L. Ablative Chamber Assembly	31	12	63								▲						4
5	M. Installation of Instrumentation	30	11	63								▲						5
6	Phase II - Firing Tests																	6
7	A. Injector Check-out	30	11	63								▲						7
8	B. Ablative Chamber Test	31	1	63									▲					8
9																		9
10	Phase III - Data Evaluation																	10
11	A. Data Reduction	21	2	64														11
12	B. Chamber Evaluation																	12
13	1. Performance	16	3	64														13
14	2. Materials Evaluation	16	3	64														14
15	3. Comparison with Subscale Tests	15	4	64														15
16																		16

Table IV-1

NOTES  
1. REMOVE ALL BURRS AND SHARP EDGES  
2. SURFACES PER MIL-STD-B-11.4.1.1-17D-12  
3. CLEAN PER AGC-46350, LEVEL II  
4. PRESERVE + PACKAGE PER AGC-46387, CLASS I



1



Uncooled Combustion Chamber, Cylindrical Throat Configuration

Figure IV-1

2

V. PROGRAM REPORTING

Table V-1 shows the schedule of reports that will be submitted for this contract.



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