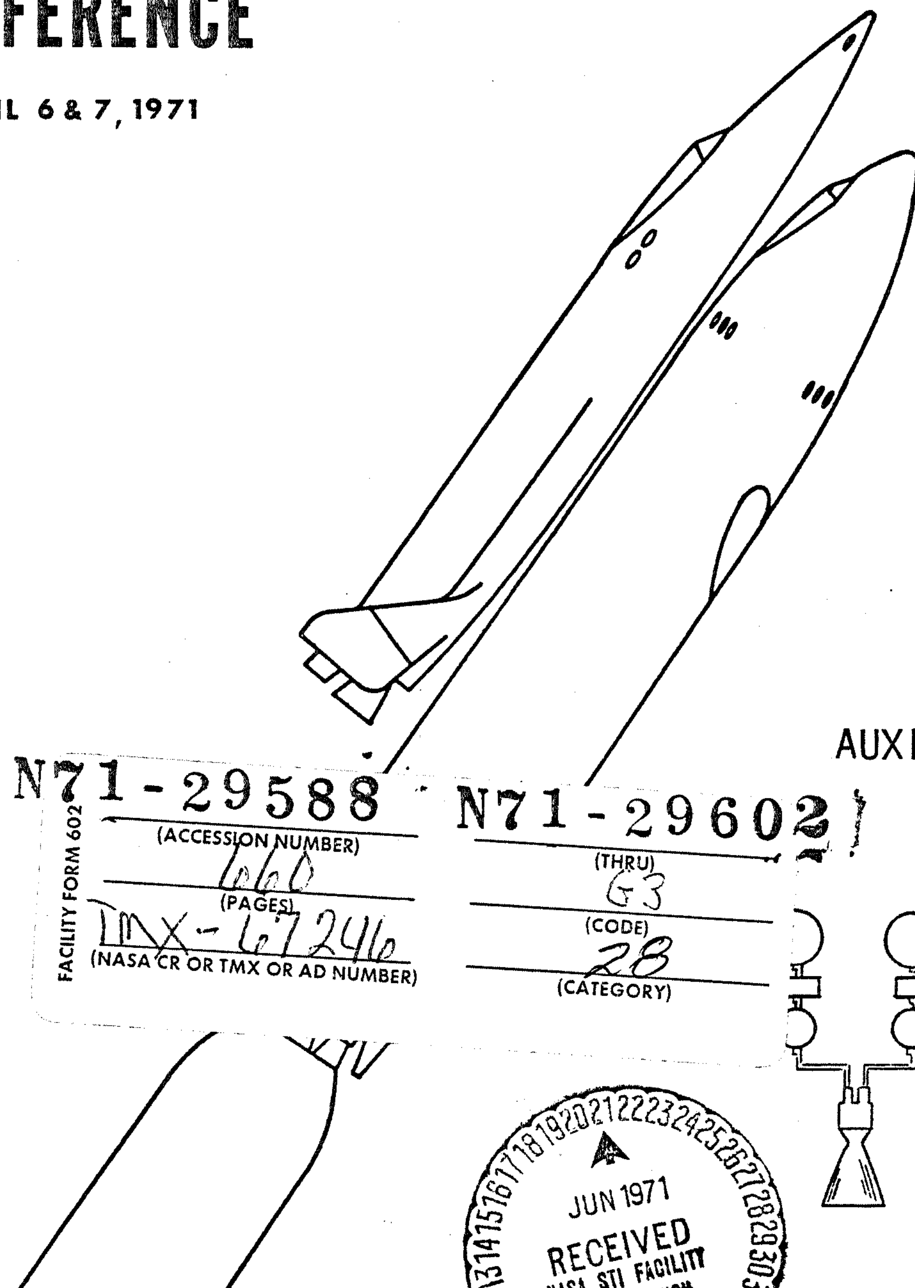
PROCEEDINGS

GEORGE C. MARSHALL SPACE FLIGHT CENTER

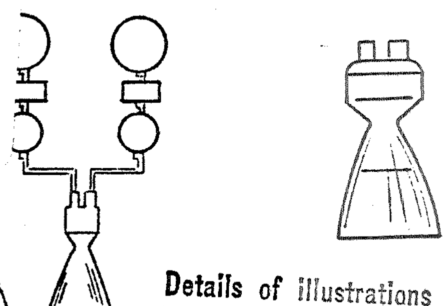
SPACE TRANSPORTATION SYSTEM PROPULSION TECHNOLOGY CONFERENCE

APRIL 6 & 7, 1971



VOLUME II

AUXILIARY PROPULSION



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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

PROCEEDINGS
SPACE TRANSPORTATION SYSTEM
PROPULSION TECHNOLOGY CONFERENCE

APRIL 6-7, 1971

MARSHALL SPACE FLIGHT CENTER

VOLUME II

SESSION II AUXILIARY PROPULSION SYSTEM E. JACOBS, CHAIRMAN

April 28, 1971

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PROPULSION TECHNOLOGY CONFERENCE PAPERS

VOLUME I MAIN PROPULSION

1.	"Final Results of the XLR-129 Program"	P&W
2.	"Two-Phase Flow LH_2 Pump Inducers"	Rocketdyne
3.	"Saturated LH_2 Turbopump Operation"	MSFC
4.	"Characteristics of Feed System Instabilities"	Martin Marietta
5.	"Combustion Oscillations Damping Devices"	P&W
6.	"Minimum Pressure Loss in High Velocity Flow Duct Systems"	S. R. I.
7.	"Engine Onboard Checkout System Study"	Martin Marietta
8.	"Low Frequency Analysis of Rocket Engines Using Compressible Propellants"	Aerojet
9.	"Titanium Pump Impeller Fabrication and Testing"	Rocketdyne
10.	"Advanced Thrust Chamber (Non-tubular)"	Rocketdyne
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VOLUME II AUXILIARY PROPULSION

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4.	"Space Shuttle ACPS Shutoff Valve"	Marquardt
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6.	"Ignition Devices for ACPS"	Aerojet
7.	"Spark and Auto Ignition Devices for ACPS"	Rocketdyne
8.	"Catalytic Ignition/Thruster Investigation"	TRW
9.	"Auxiliary Propulsion Subsystem Investigations"	MSFC
10.	"Injector Performance, Heat Flux and Film Cooling in O_2/H_2 Engines"	MSC
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12.	"Auxiliary Propulsion Subsystem Definition Study"	MDAC
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PROPULSION TECHNOLOGY CONFERENCE PAPERS

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1.	"Auxiliary Power Unit Design Studies"	Airesearch
2.	"Auxiliary Power Unit Design Studies"	Rocketdyne
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4.	"Recent Developments in High Performance Insulation Purge Systems for Shuttle Application"	MDAC
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VOLUME II

AUXILIARY PROPULSION

Auxiliary Propulsion Panel
Ed Jacobs, Chairman

Gentlemen.

Let me start by pointing out that the Auxiliary Propulsion Panel covers both the Attitude Control System and the Orbital Maneuvering System. As you may have noticed on Mr. Thomson's funding chart, the Auxiliary Propulsion work has been running approximately $5 million a year.

During the past year the primary effort on the Attitude Control Thruster work has been devoted to developing the components such as the valves, ignition system and cooling concepts. In the coming year the results of these programs will be used to establish an overall thruster concept that will meet the Space Shuttle requirements.

There were four system studies devoted to the definition of the Attitude Control System, two of which were high pressure; that is, utilizing thrusters of approximately 300 psi chamber pressure, and two low pressure systems; that is, utilizing thrusters of approximately 15 psi chamber pressure. Based on the results of these studies, the high pressure system has been selected for further study.

At the present time Requests for Proposals are being released for the system critical components such as turbopumps, heat exchangers, gas generators, regulators, and control systems. It is planned that these critical components will eventually be assembled for a breadboard system test to demonstrate the feasibility of the system.

There is not yet much underway on the Orbital Maneuvering System, but a Request for Proposal was recently released to take a more in-depth look at this system.

As you will hear in papers to be presented tomorrow, there has also been a significant amount of work conducted in-house both at the Manned Spacecraft Center and here at the Marshall Space Flight Center.

And with that brief summary, I would like to present the first speaker, Mr. Gary Falkenstein, who will discuss the work at Rocketdyne on Hydrogen-Oxygen ACPS Engines.

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"HYDROGEN/OXYGEN ACPS ENGINES"

G. L. FALKENSTEIN

ROCKETDYNE

TECHNICAL MANAGER

P. N. HERR

LEWIS RESEARCH CENTER

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TITLE: HYDROGEN - OXYGEN APS ENGINES

CONTRACT: NAS3 - 14352

PRESENTER: G. L. FALKENSTEIN, PRINCIPAL ENGINEER

COMPANY: ROCKETDYNE, DIV. OF NORTH AMERICAN ROCKWELL

NASA PROJ. MANAGER: P. HERR

THE SPACE SHUTTLE AUXILIARY PROPULSION SYSTEM (APS) WILL REQUIRE A REUSABLE, LONG LIFE, HIGH PERFORMANCE ENGINE OPERATING ON GASEOUS HYDROGEN/GASEOUS OXYGEN PROPELLANTS. THIS PROGRAM HAS THE OBJECTIVE OF ESTABLISHING THE TECHNOLOGY BASE FOR SUCH ENGINES.

SPACE SHUTTLE AUXILIARY PROPULSION SYSTEM
HIGH AND LOW PRESSURE THRUSTERS

NAS3-14352

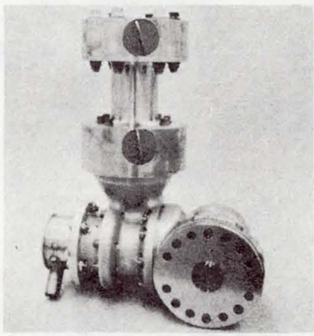
393

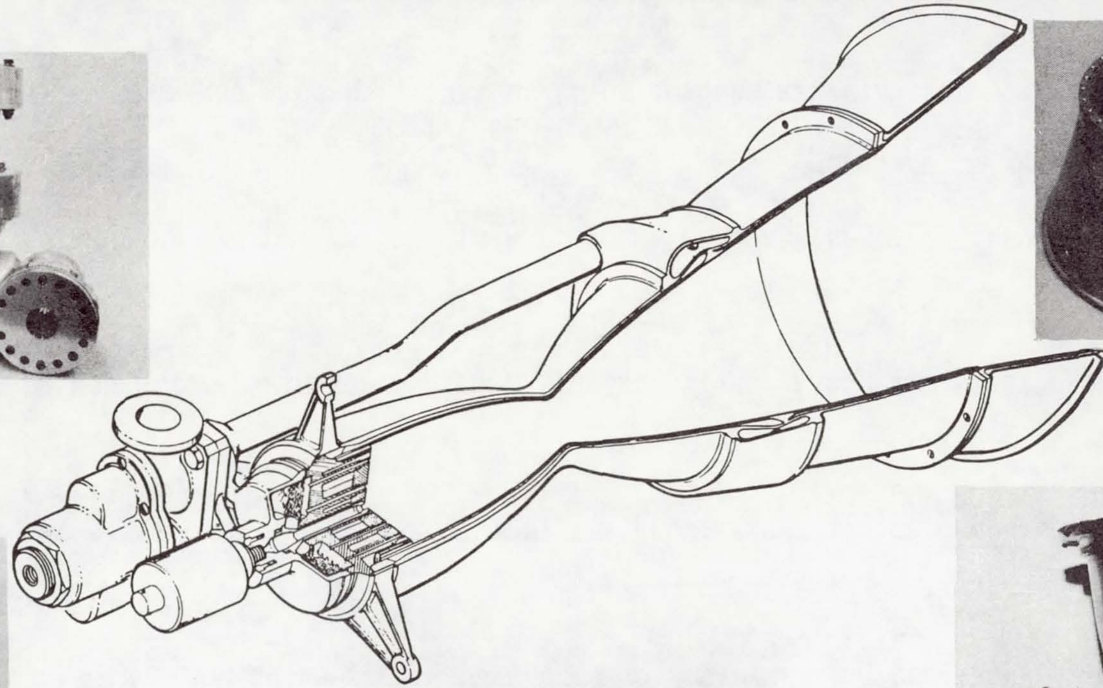
OBJECTIVE: INVESTIGATE THE USE OF GASEOUS HYDROGEN - GASEOUS OXYGEN PROPELLANTS IN AUXILIARY PROPULSION ENGINES INCLUDING EVALUATION OF THRUST CHAMBER COOLING CONCEPTS, GAS-GAS INJECTOR DESIGNS, INJECTOR CHAMBER INTERACTIONS, AND THE PULSING AND STEADY-STATE PERFORMANCE OF HIGH-PRESSURE AND LOW-PRESSURE THRUSTERS.

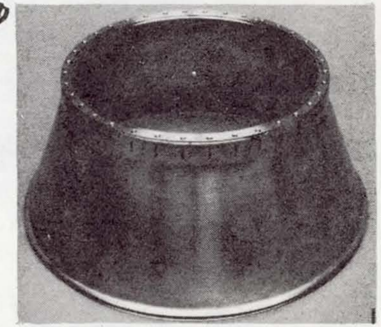
IN ORDER TO CONDUCT THE EXTENSIVE INJECTOR, THRUST CHAMBER AND ASSEMBLY TEST EFFORT, VALVES AND IGNITERS WERE DEVELOPED. A J-2S BALL VALVE WAS MODIFIED BY REPLACING THE PNEUMATIC ACTUATOR WITH A HYDRAULIC ACTUATOR TO OBTAIN A RESPONSE TIME OF 30 MILLI-SECONDS. BOTH CATALYTIC AND SPARK IGNITERS WERE DESIGNED AND BUILT. IGNITER CHECKOUT TESTS AND INTEGRATED INJECTOR/IGNITER TESTS WERE RUN WITH BOTH IGNITER TYPES. BASED ON FASTER RESPONSE AND BETTER IGNITIBILITY CHARACTERISTICS, THE SPARK IGNITER WAS SELECTED TO COMPLETE THE THRUST CHAMBER TEST EFFORT.

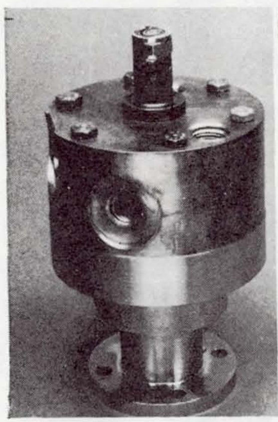
HIGH PRESSURE APS ENGINE

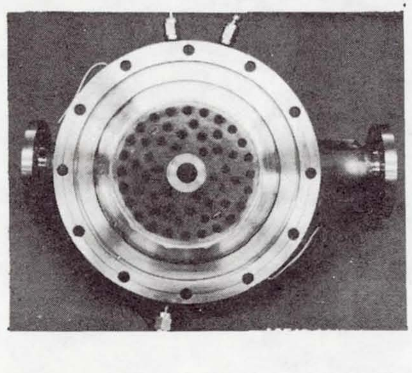
THRUST = 1500 LBS	MIXTURE RATIO = 4.0
CHAMBER PRESSURE = 300 PSIA	INLET PRESSURE = 375 TO 400 PSIA
NOZZLE EXPANSION RATIO = 40:1	INLET TEMPERATURE = 540 R

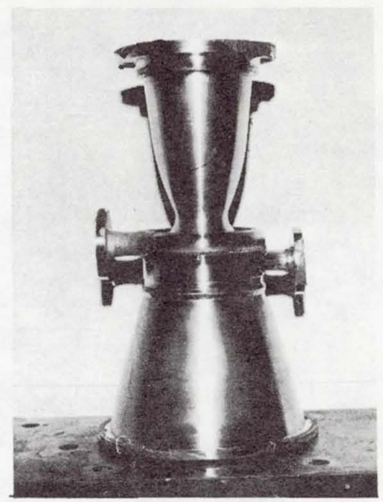












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THE APS ENGINE WILL BE USED FOR LIMIT CYCLE CONTROL, RE-ENTRY ALTITUDE CONTROL, ATTITUDE MANEUVERING, AND TRANSLATIONAL ΔV MANEUVERS. THIS USAGE MAKES AN UNLIMITED DUTY CYCLE HIGHLY DESIRABLE, RANGING FROM 50 LB-SEC IMPULSE BITS TO STEADY FIRING DURATIONS OF 1000 SECONDS. FOR 100 MISSIONS OVER A TEN YEAR PERIOD, THE REQUIREMENT IS FOR ONE MILLION PULSE CYCLES AND 50 HOURS TOTAL FIRING DURATION. THE RELATIVELY LARGE AMOUNT OF PRO-PELLANT EXPENDED IN THE APS SYSTEM RESULTED IN A PERFORMANCE GOAL (SPECIFIC IMPULSE) FOR THE PROGRAM OF GREATER THAN 435 SECONDS. THIS LEVEL OF IMPULSE EFFICIENCY ($\sim$ 92 PERCENT) IS FAIRLY HIGH FOR SMALL, LONG LIFE ENGINES.

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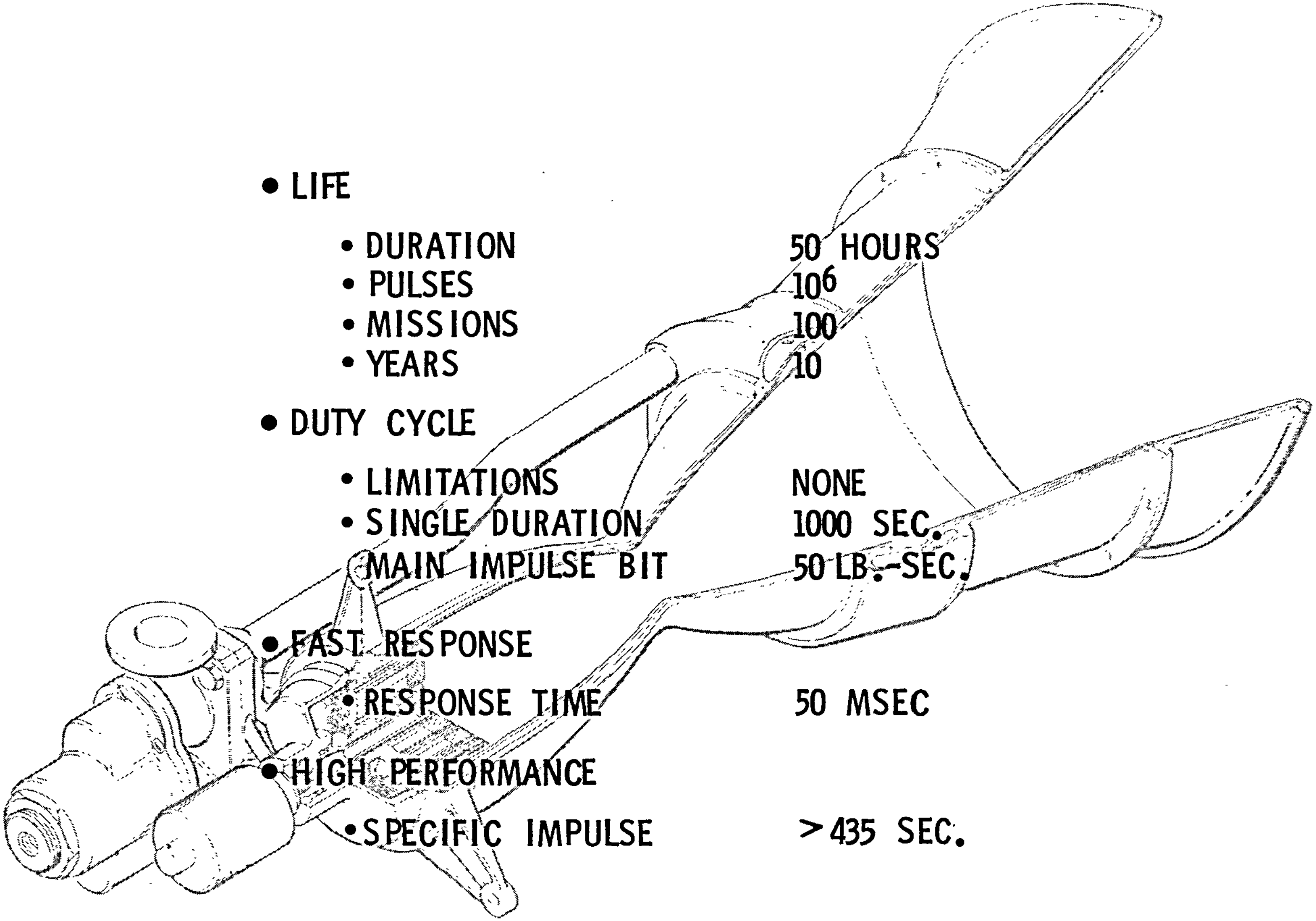
REQUIREMENTS



- LIFE
  - DURATION — 50 HOURS
  - PULSES — 10^6
  - MISSIONS — 100
  - YEARS — 10
- DUTY CYCLE
  - LIMITATIONS — NONE
  - SINGLE DURATION — 1000 SEC.
  - MAIN IMPULSE BIT — 50 LB.-SEC.
- FAST RESPONSE
  - RESPONSE TIME — 50 MSEC
- HIGH PERFORMANCE
  - SPECIFIC IMPULSE — $>$435 SEC.

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EVALUATION OF COOLING CONCEPTS LED TO THE CONCLUSION THAT ACTIVE TECHNIQUES WERE DESIRABLE TO MEET THE UNLIMITED DUTY CYCLE AND REUSE REQUIREMENTS, THE BURIED ENVELOPE TEMPERATURE OF 820F AND THE REQUISITE LIFE. FURTHER ANALYSIS AND DESIGN STUDY, IDENTIFIED TWO DESIGNS AS MOST PROMISING. THE HIGHER PERFORMANCE DESIGN USES REGENERATIVE CHAMBER COOLING FROM AN EXPANSION RATIO OF 3:1 SUPPLEMENTED BY A SMALL AMOUNT OF FILM COOLING. THE NOZZLE IS DUMP COOLED TO AN EXPANSION RATIO OF 18:1 AND UTILIZES A FILM COOLED NOZZLE EXTENSION TO THE 40:1 EXIT PLANE. THE SECOND CONCEPT ACHIEVES A LOWER FUEL INLET PRESSURE BY RUNNING THE CHAMBER COOLANT IN PARALLEL WITH THE INJECTOR FUEL. THE COOLANT IS THEN DUMPED DIRECTLY INTO THE CHAMBER AS FILM COOLANT. NOZZLE COOLING IS IDENTICAL TO THE REGENERATIVE ENGINE. WITH THE NOZZLE FILM COOLANT CONSERVATIVELY (HIGH VALUE) SPECIFIED, THE CHAMBER MIXTURE RATIO IS 4.7 FOR AN OVERALL MIXTURE RATIO OF 4.0.

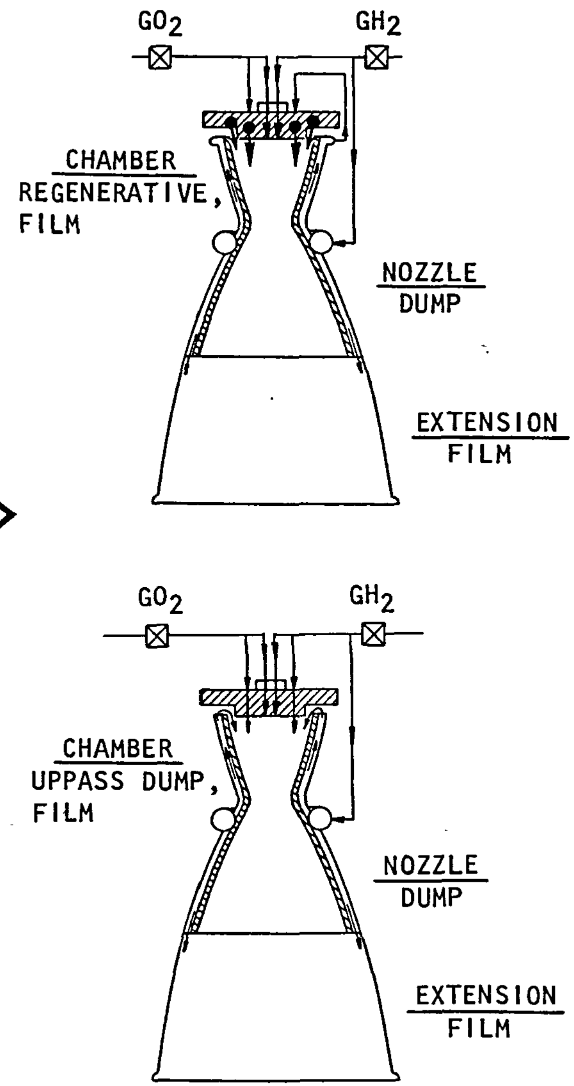
THRUST CHAMBER COOLING CONCEPTS

PASSIVE

- HEAT SINK
- ABLATION
- RADIATION

ACTIVE

- TRANSPIRATION
- FILM
- INTEREGEN
- DUMP
- REGENERATIVE



THREE LIFE-RELATED OPERATIONAL CASES ARE USED TO EVALUATE THRUST CHAMBER CAPABILITIES. THE FIRST, A FULL STEADY-STATE CYCLE, ASSUMES AN INITIALLY AMBIENT ENGINE FIRED UNTIL STEADY TEMPERATURES AND TEMPERATURE GRADIENTS HAVE BEEN ACHIEVED. AFTER SHUTDOWN THE ENGINE IS ALLOWED TO COOL BACK DOWN TO AMBIENT TEMPERATURE. A PULSE CYCLE IS DEFINED AS A SHORT (50 TO 200 MILLISECOND) FIRING PRIOR TO ENGINE COOLDOWN. THE PULSE DURATION IS SUFFICIENTLY LONG TO ALLOW TEMPERATURE GRADIENTS TO BE FULLY ESTABLISHED. FOR BOTH TYPES OF CYCLES, ALL STRAINS INCLUDING THERMAL, PRESSURE, AND THRUST ARE INCLUDED IN THE ANALYSIS SINCE OVER 90% OF THE STRAIN IS THERMALLY INDUCED. ACCUMULATED TIME AT THE STEADY-STATE OPERATING CONDITION IS ALSO INCLUDED SO AS TO CONSIDER CREEP RUPTURE CHARACTERISTICS.

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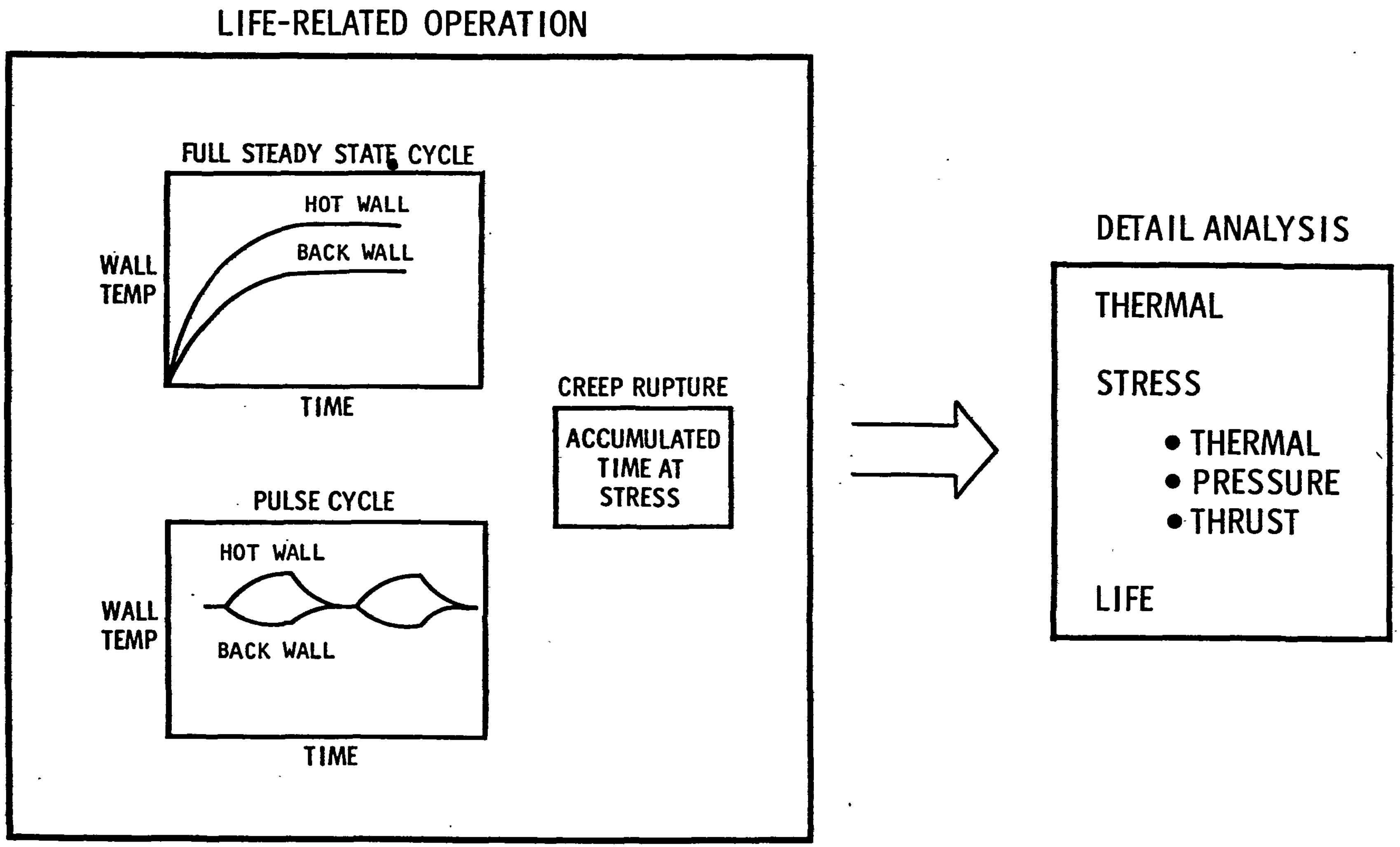
CHAMBER DESIGN ANALYSIS



THE DETAILED ANALYSIS RESULTED IN THE CONCLUSION THAT VERY LOW STRAIN DESIGNS ARE REQUIRED. THIS CAN ONLY BE ACHIEVED WITH LOW TEMPERATURE GRADIENTS. LOW HEAT FLUX INJECTORS AND CHAMBER MATERIALS WITH A HIGH THERMAL CONDUCTIVITY ARE REQUIRED. COPPER ALLOYS ARE MOST ATTRACTIVE AS HOT WALL MATERIALS. A RELATIVELY HIGH MATERIAL STRENGTH AT THE OPERATING TEMPERATURE IS ALSO REQUIRED. CAREFUL ANALYSIS SHOWED NARloy-Z (A COPPER, SILVER ZIRCONIUM ALLOY) TO BE MOST ATTRACTIVE. THIS ALLOY HAS BEEN DEVELOPED BY ROCKETDYNE AS A HIGH STRENGTH, INSPECTABLE ALLOY FOR LONG LIFE, LOW COST, REUSABLE REGENERATIVE THRUST CHAMBER APPLICATIONS.

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CHAMBER DESIGN CRITERIA

LIFE CRITERIA

- LOW STRAIN DESIGN
- LOW THERMAL GRADIENTS
- HIGH MATERIAL STRENGTH

COMPONENT DESIGN

INJECTOR

- LOW CHAMBER HEAT FLUX

CHAMBER MATERIAL

- NAR loy-Z LINER
- HIGH CONDUCTIVITY
- HIGH STRENGTH

CHAMBER DESIGN

- LOW OPERATING TEMP

THE PRIMARY OBJECTIVE OF THE INJECTOR EFFORT WAS THE ACHIEVEMENT OF HIGH PERFORMANCE WITH AN INJECTOR MEETING THE HEAT FLUX REQUIREMENTS REQUIRED FOR CHAMBER LIFE. THE INJECTOR EVALUATION EFFORT WAS INITIATED BY SCREENING POTENTIAL ELEMENT PATTERNS FOR APPLICABILITY TO THE APS ENGINE. PARALLEL COLD FLOW TESTS FOR PERFORMANCE POTENTIAL AND DESIGN STUDIES WERE USED TO SELECT THE TWO MOST PROMISING CONFIGURATIONS. THE CONCENTRIC TUBE PATTERN HAS BEEN USED DURING EARLIER GAS-GAS STUDIES AND GAVE HIGH PERFORMANCE WITH A LOW HEAT FLUX CHARACTERISTIC. THE TRISLOT PATTERN (NON-CIRCULAR TRIPLET) HAS A HIGHER PERFORMANCE POTENTIAL AND HAS THE POTENTIAL OF A MORE EASILY FABRICATED AND DURABLE DESIGN. FOLLOWING A MORE DETAILED ELEMENT COLD FLOW EFFORT WHICH EMPHASIZED LOW HEAT FLUX WITH HIGH PERFORMANCE, THE TWO INJECTOR TYPES WERE FABRICATED. HOT-FIRE TESTING SHOWED THE CONCENTRIC TUBE INJECTOR TO BE SUPERIOR, REFLECTING ITS MORE MATURE DESIGN STATUS. THE CONCENTRIC TUBE INJECTOR WAS SHOWN TO GIVE THE DESIRED HIGH PERFORMANCE, YET HAVE THE LOW HEAT FLUX CHARACTERISTICS REQUIRED FOR LONG CHAMBER LIFE. ADDITIONAL TRISLOT INJECTOR DEVELOPMENT SHOULD BE ACCOMPLISHED.

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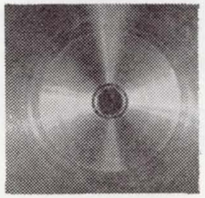
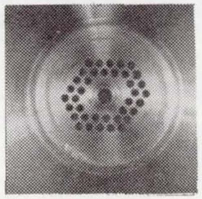
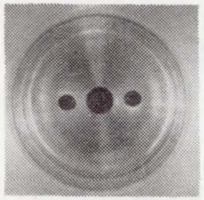
INJECTOR STUDIES

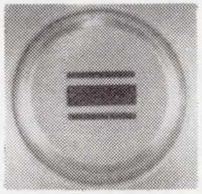
CONCEPT EVALUATION

COLD FLOW SCREENING

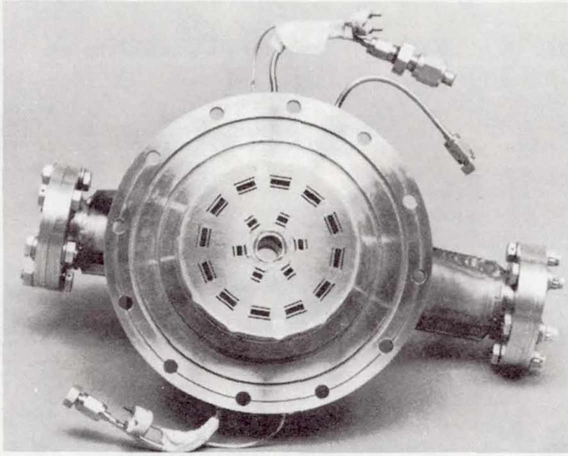





DETAIL DESIGN

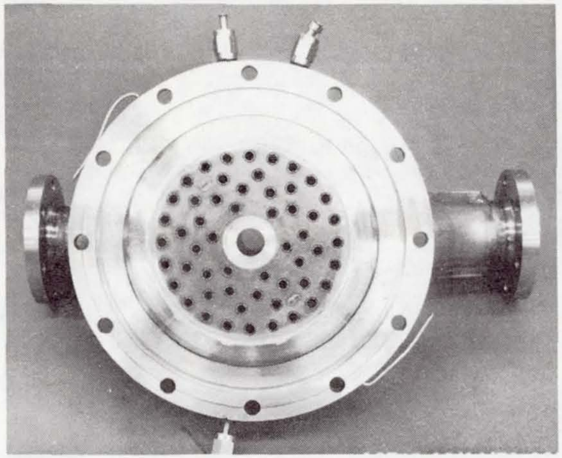
COLD FLOW OPTIMIZATION




TRISLOT



CONCENTRIC TUBE



ALL OBJECTIVES ACCOMPLISHED

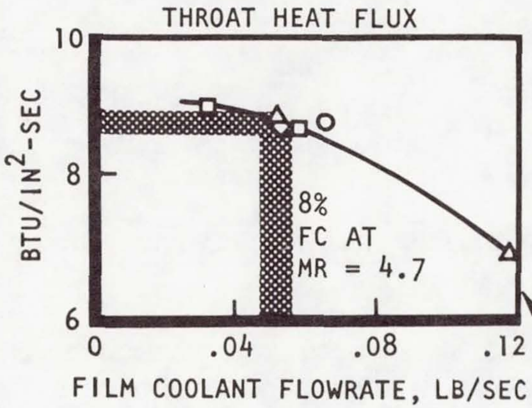
- HIGH PERFORMANCE
- LOW CHAMBER HEAT FLUX
- CHAMBER COMPATABILITY
- DURABILITY

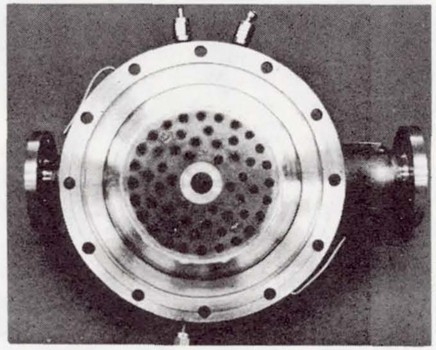
CONCENTRIC TUBE INJECTOR SELECTED

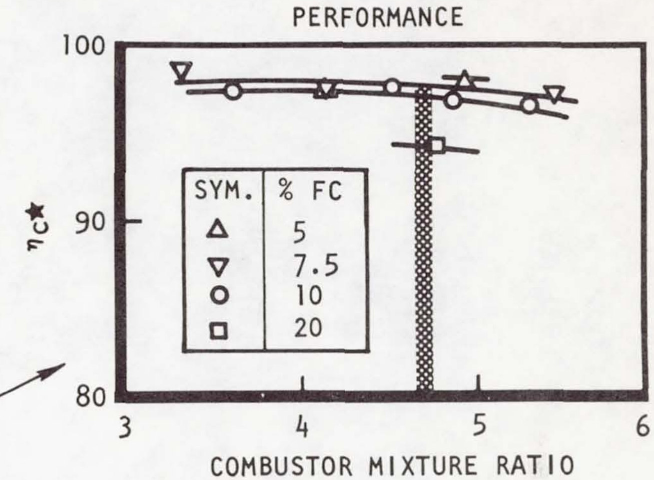
Rocketdyne
North American Rockwell

THE CONCENTRIC TUBE INJECTOR AS IT APPLIES TO THE REGENERATIVE ENGINE HAS A C* EFFICIENCY OF 97.7% WITH 8% FILM COOLANT. THIS FILM COOLANT PERCENTAGE WAS SELECTED TO GIVE THE REQUIRED HEAT FLUX CHARACTERISTICS AS TYPIFIED BY A THROAT HEAT FLUX OF LESS THAN 9 BTU/IN^2-SEC. A WATER COOLED THROAT WITH AN ABLATIVE CHAMBER LINER WERE USED TO VERIFY CHAMBER COMPATIBILITY AND INJECTOR DURABILITY. DURATION TESTS OF 10 - 30 SECONDS WERE MADE WITH AND WITHOUT INJECTOR AND FILM COOLANT. ABLATION WAS UNIFORM WITH ONLY MINOR EROSION. MEASURED INJECTOR FACE TEMPERATURE WAS QUITE LOW, ONLY 200R HIGHER THAN THE INLET HYDROGEN TEMPERATURE. THIS VERIFIES THE DURABILITY OF THE BASIC INJECTOR DESIGN AND LENDS CONFIDENCE THAT THE LONG DURATION CHAMBER TEST SERIES WILL BE ACCOMPLISHED SUCCESSFULLY.

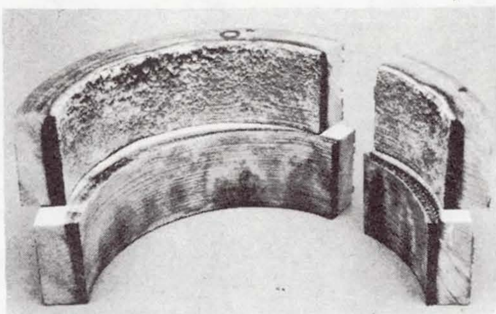
REGENERATIVE ENGINE INJECTOR

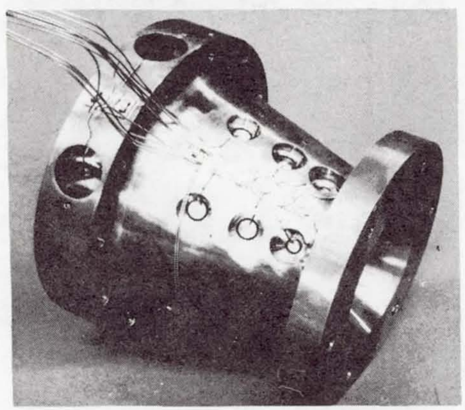


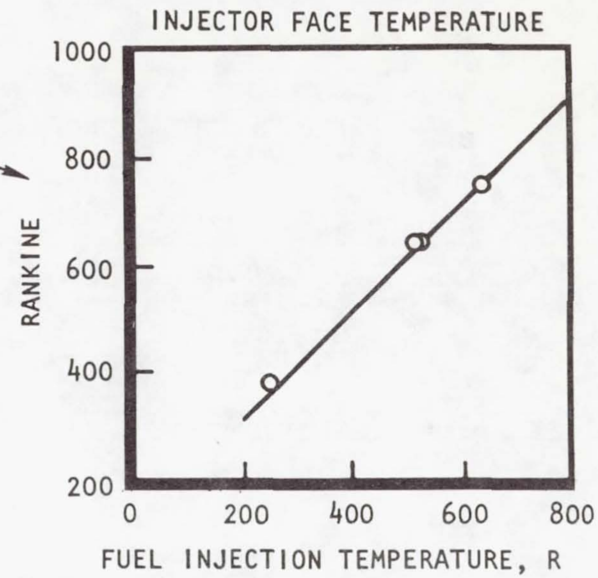




- HIGH PERFORMANCE
- LOW CHAMBER HEAT FLUX
- CHAMBER COMPATIBILITY
- INJECTOR DURABILITY



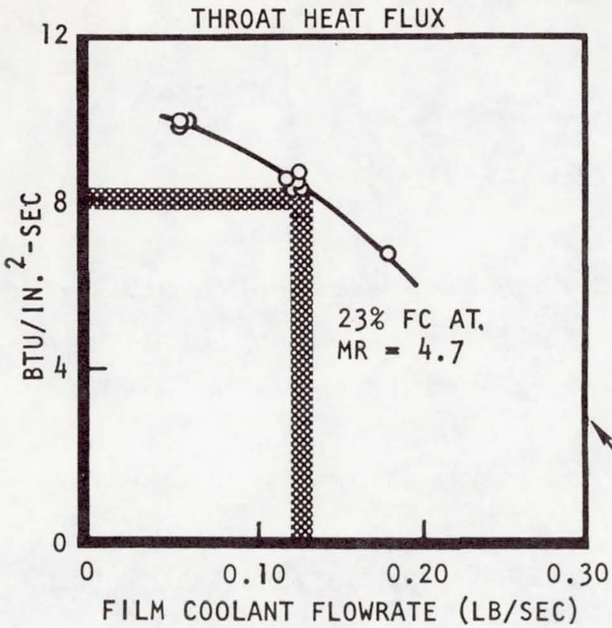


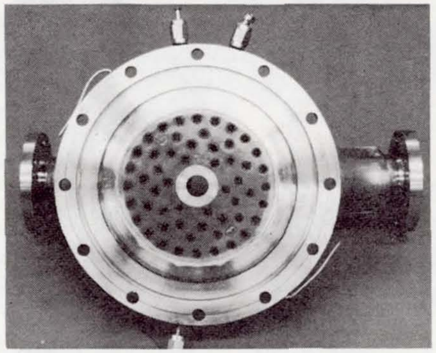


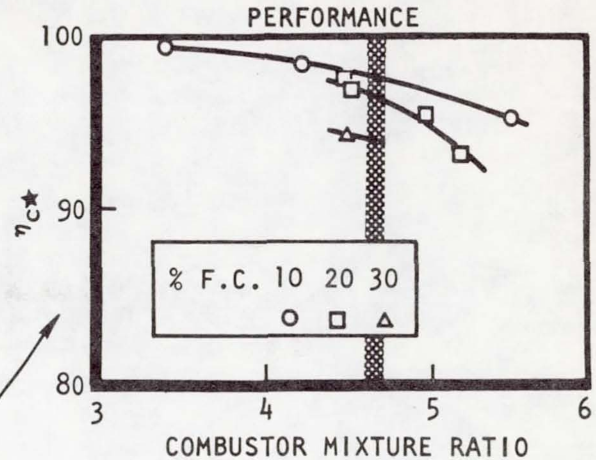
Rocketdyne
North American Rockwell

THE DUMP COOLED ENGINE REQUIRES MORE FILM COOLANT THAN WITH THE REGENERATIVE ENGINE AND PERFORMANCE (C* EFFICIENCY) DROPS TO 96.3%. THIS IS STILL A HIGH PERFORMANCE FOR SMALL LONG LIFE ENGINES. TO ACHIEVE THE REQUIRED HEAT FLUX CHARACTERISTICS, APPROXIMATELY 8 BTU/IN^2-SEC AT THE THROAT, REQUIRES 23% FILM COOLANT. NO CHANGE IN PERFORMANCE WAS NOTED OVER A CHAMBER PRESSURE RANGE OF 100 - 500 PSIA. AS EXPECTED THROAT HEAT FLUX VARIED WITH CHAMBER PRESSURE TO THE 0.8 POWER.

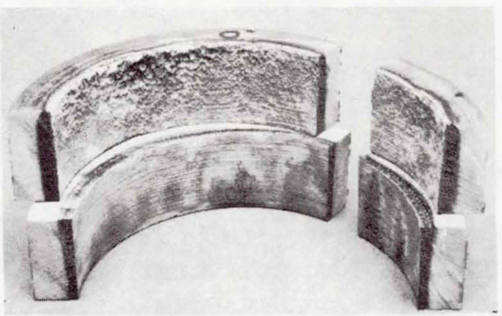
UPPASS DUMP ENGINE INJECTOR

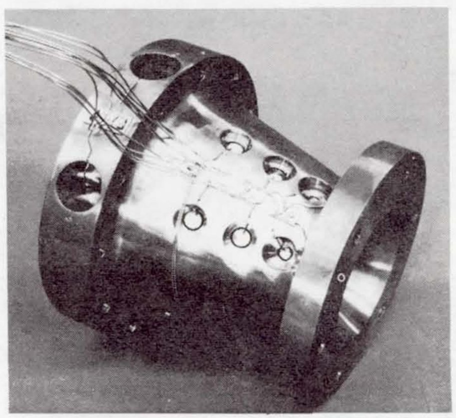


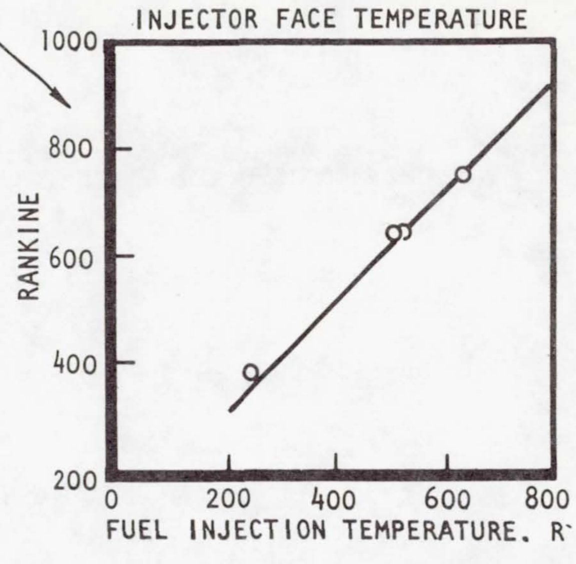




- HIGH PERFORMANCE
- LOW CHAMBER HEAT FLUX
- CHAMBER COMPATIBILITY
- INJECTOR DURABILITY







Rocketdyne
North American Rockwell

THE EXPERIMENTAL DATA WERE OBTAINED USING A FACILITY DESIGNED FOR HIGH RESPONSE GAS-GAS THRUSTERS. A 32 FOOT LONG BY 15 FOOT CAN CONTAINS THE HIGH PRESSURE TEST STAND AND A SIMILAR LOW PRESSURE TEST STAND USED TO ACCOMPLISH 15 PSI THRUSTER EXPERIMENTAL WORK NOW TERMINATED. THE GASEOUS FEED SYSTEM IS DESIGNED TO CONTROL TO A CONSTANT INLET PRESSURE SIMULATING A VEHICLE SYSTEM. THUS, REPRESENTATIVE FLOW TRANSIENTS WERE OBTAINED. CAPACITY HEAT EXCHANGERS ARE USED TO CONTROL INLET TEMPERATURE OVER THE 200 - 800R TEMPERATURE RANGE. A THRUST MEASUREMENT SYSTEM WAS CUSTOM DESIGNED TO OBTAIN HIGH RESPONSE THRUST MEASUREMENT (FREQUENCY RESPONSE GREATER THAN 300 H_z). HIGH PERFORMANCE IMPACT METERS ARE USED TO MEASURE TRANSIENT FLOW RATES. THE HYPERFLOW SYSTEM USED TO MAINTAIN ALTITUDE PRESENTLY HAS A DURATION CAPACITY OF 1000 SECONDS.

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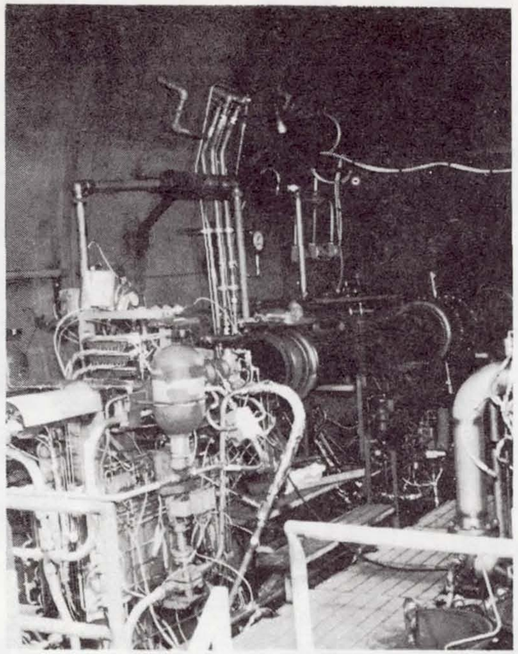
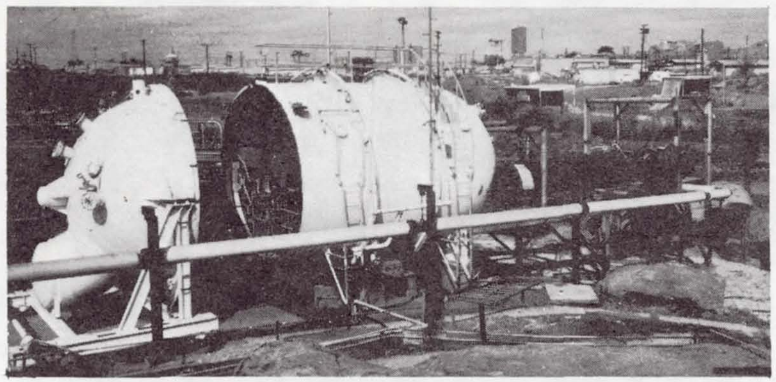
APS ENGINE FACILITY



HIGH RESPONSE
PERFORMANCE
MEASUREMENTS

ALTITUDE CAPABILITY

- <0.5 PSIA
- 1000 SECOND DURATION



SIMULATED VEHICLE FEED SYSTEM

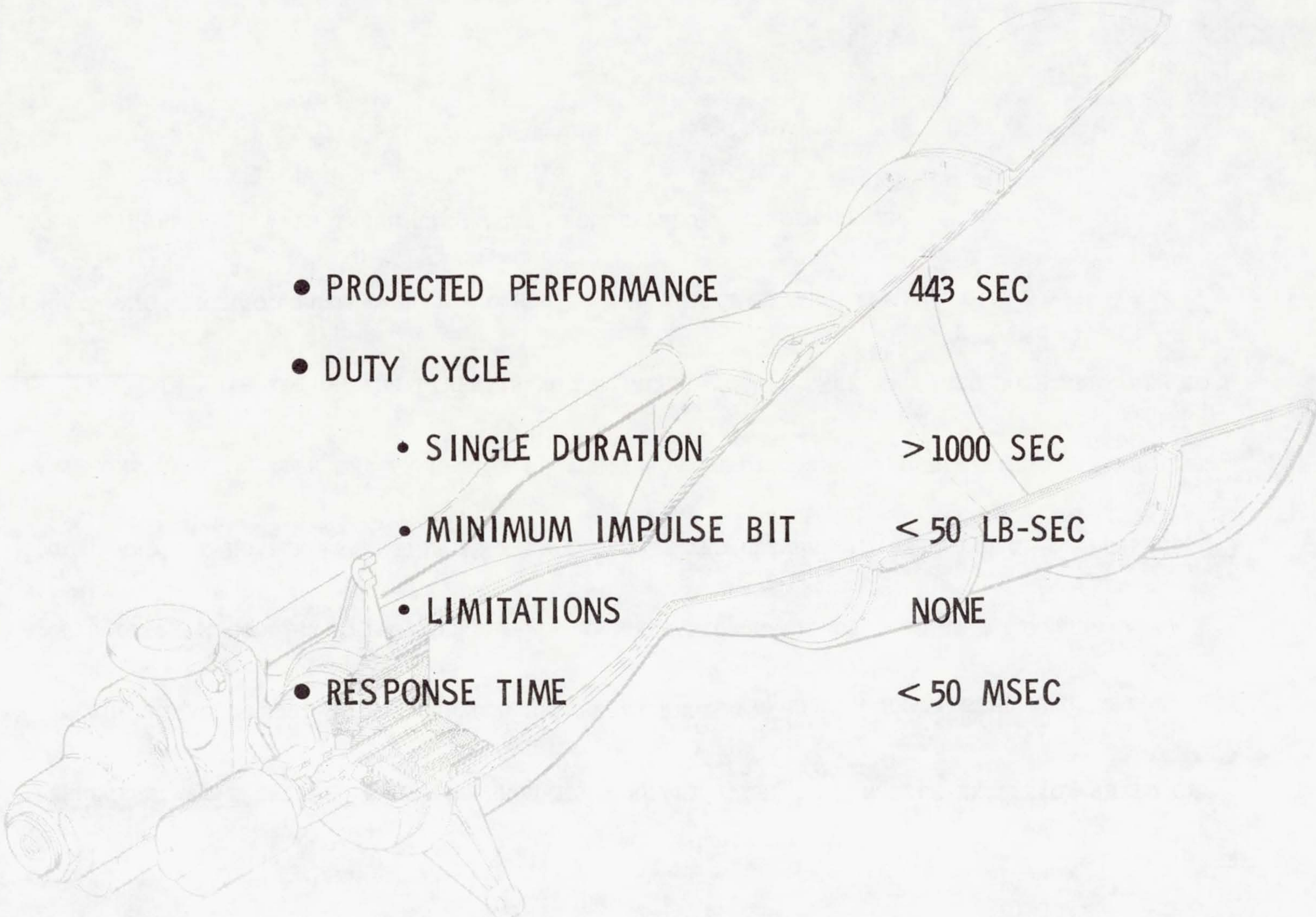
- INLET PRESSURE CONTROL
- 200 TO 800 R GASEOUS FEEDS

Rocketdyne
North American Rockwell

THE REGENERATIVE ENGINE WAS DESIGNED AND ANALYZED IN DETAIL USING THE EXPERIMENTALLY OBTAINED HEAT FLUX DATA. BASED ON THE MEASURED INJECTOR PERFORMANCE AT THE HEAT FLUX LEVELS REQUIRED TO MEET THE LIFE GOALS, THE ENGINE WILL HAVE A SPECIFIC IMPULSE OF APPROXIMATELY 443 SECONDS. THE LIFE AND RESPONSE GOALS WILL BE EXCEEDED.

275-815
3-71

REGENERATIVELY COOLED ENGINE



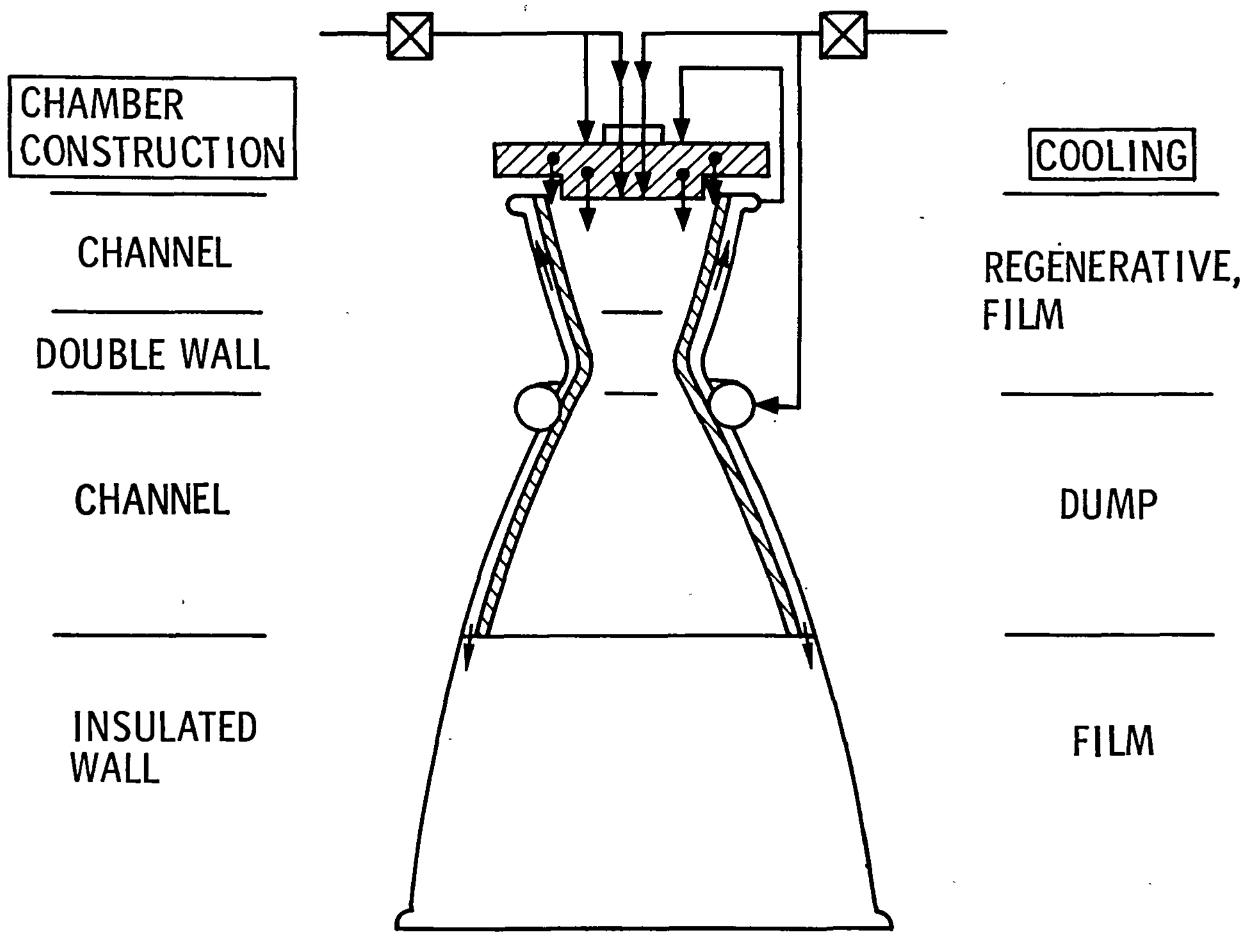
- PROJECTED PERFORMANCE — 443 SEC
- DUTY CYCLE
  - SINGLE DURATION — >1000 SEC
  - MINIMUM IMPULSE BIT — < 50 LB-SEC
  - LIMITATIONS — NONE
- RESPONSE TIME — < 50 MSEC

Rocketdyne
North American Rockwell

THE REGENERATIVE ENGINE IS COOLED FROM THE COOLANT INLET IN A NOZZLE EXTENSION RATIO OF 3 TO THE INJECTOR END OF THE CHAMBER REGENERATIVELY AND USES A SMALL SUPPLEMENTARY AMOUNT OF FILM COOLANT. THE NOZZLE FROM AN EXTENSION RATIO OF 3 TO 18 IS DOWNPASS COOLED AND THE NOZZLE EXTENSION IS FILM COOLED. THE CHAMBER IS ESSENTIALLY OF CHANNEL WALL CONSTRUCTION WITH THE EXCEPTION OF THE THROAT REGION WHERE DOUBLE WALL CONSTRUCTION IS USED. IN THIS REGION THE LINER IS NOT ATTACHED TO THE BACK WALL AND BOTH ARE FREE TO EXPAND AND CONTRACT FREE FROM THE OTHER. THE NOZZLE EXTENSION IS A THIN STAINLESS STEEL SINGLE WALL WITH INSULATION ADDED FOR ENVELOPE TEMPERATURE.

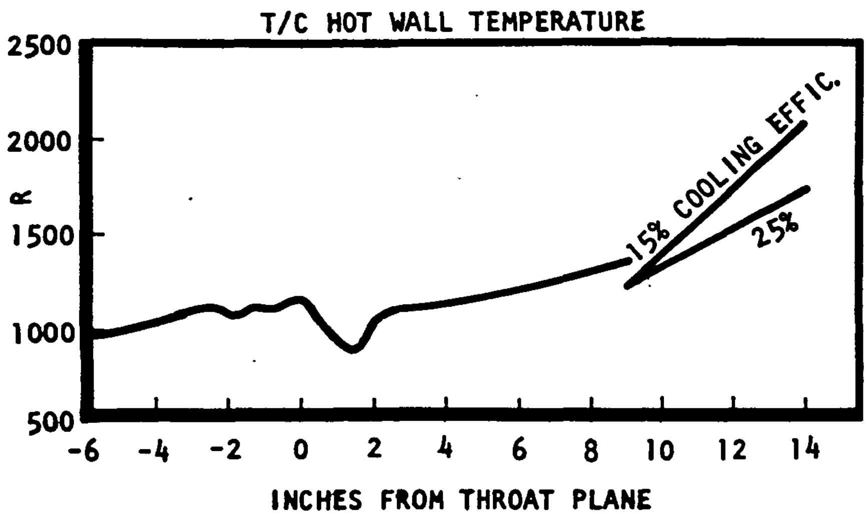
275-859
3-71

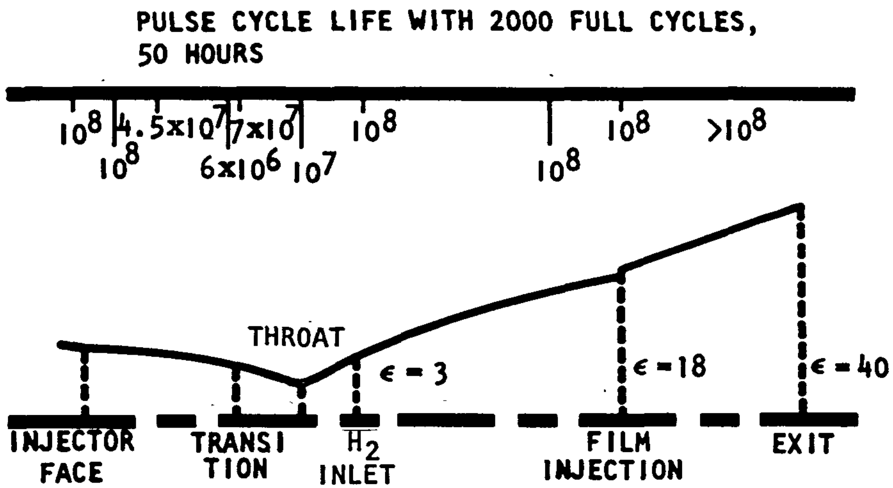
REGENERATIVE CHAMBER DESIGNS

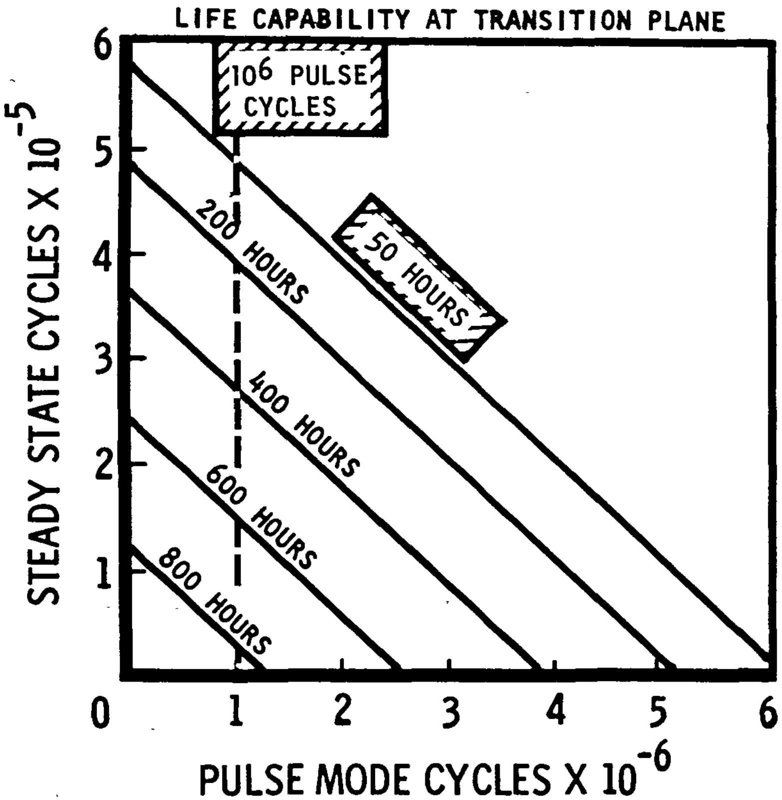


DETAILED ANALYSIS OF THE REGENERATIVE CHAMBER USING THE EXPERIMENTAL HEAT FLUX DATA SHOWED A MAXIMUM CHAMBER WALL TEMPERATURE OF 1200R AT THE THROAT PLANE. AT THE EXIT OF THE DUMP COOLED SECTION A TEMPERATURE OF APPROXIMATELY 1300R OCCURS. THE MINIMUM LIFE PLANE OF THE CHAMBER IS IN THE CHAMBER WALL SECTION AT THE TRANSITION TO THE DOUBLE WALL THROAT SECTION. AT THIS PLANE A CAPACITY OF 6,000,000 PULSE CYCLES IN CONJUNCTION WITH 2000 FULL STEADY STATE CYCLES AND A 50 HOUR TOTAL DURATION IS PREDICTED. FURTHER AT THE TRANSITION PLANE WITH 1,000,000 PULSE SECONDS AND A 50 HOUR DURATION, THE STEADY STATE CYCLE CAPABILITY IS APPROXIMATELY ONE-HALF MILLION.

REGENERATIVELY COOLED THRUST CHAMBER







Rocketdyne
North American Rockwell

THE LIFE MARGIN CAPABILITY WAS DETERMINED FOR THE CHAMBER FIRED UNDER VARIOUS OFF-LIMIT CONDITIONS. THE TABLE SHOWS THE LIFE CAPABILITY AT THE LIFE LIMITING POINT IN THE CHAMBER. UNDER NOMINAL CONDITIONS THE HOT WALL TEMPERATURE AT THE TRANSITION IS 1080 R AND THE LIFE CAPABILITY IS 6,000,000. RUNNING THE CHAMBER AT 500 PSIA INCREASED THE TEMPERATURE BY 50° R. EVEN AT THIS ELEVATED PRESSURE THE LIFE IS ALMOST ONE MILLION CYCLES. BY SLIGHTLY INCREASING THE FILM COOLANT RATE OF THE EXISTING CHAMBER DESIGN, ONE MILLION PULSE CYCLE LIFE CAN BE ACHIEVED WITH A MINIMUM OF PERFORMANCE DEGRADATION. AT 300 PSIA CHAMBER PRESSURE WITH THE INLET PRESSURES MIS-MATCHED SO THAT A MIXTURE RATIO OF 5 EXISTS THE TRANSITION TEMPERATURE IS INCREASED TO 1140 R, AND ONLY A MINOR DEGRADATION IN LIFE CAPABILITY OCCURS. WITH THE PROPELLANT INLET TEMPERATURES AT 800 R, REPRESENTATIVE OF A MAXIMUM HEAT SOAKBACK TO THE PROPELLANT, THE WALL TEMPERATURE IS 1290 R. AGAIN, ONLY A MINIMUM DEGRADATION OF LIFE OCCURS. TO EVALUATE STREAKING AN INCREASE IN THE HEAT FLUX OF 25% FROM THE INJECTOR END OF THE CHAMBER TO THE COOLANT INLET WAS ASSUMED. THIS INTENSITY IS REPRESENTATIVE OF COMPLETELY LOSING FILM COOLANT AND IN ADDITION HAVEING A DAMAGED INJECTOR ELEMENT IN THE OUTER ROW OF THE INJECTOR FACE. WITH THIS CONDITION THE TRANSITION TEMPERATURE IS 1230 R AND THE LIFE CAPABILITY IS ALMOST ONE MILLION PULSE CYCLES.

REGENERATIVE ENGINE MARGIN CAPABILITY

OFF-LIMIT CONDITION	TRANSITION TEMPERATURE, R	LIFE CAPABILITY CYCLES
NOMINAL	1080	6×10^6
P_C = 500 PSIA	1130	7.5×10^5
MR = 5	1140	4.7×10^6
INLET TEMP. = 800 R	1290	5.3×10^6
STREAK - Q/A UP 25%	1230	7.5×10^5

NOMINAL CONDITIONS
• P_C = 300 PSIA • MR = 4.0 • INLET TEMP. = 540 R

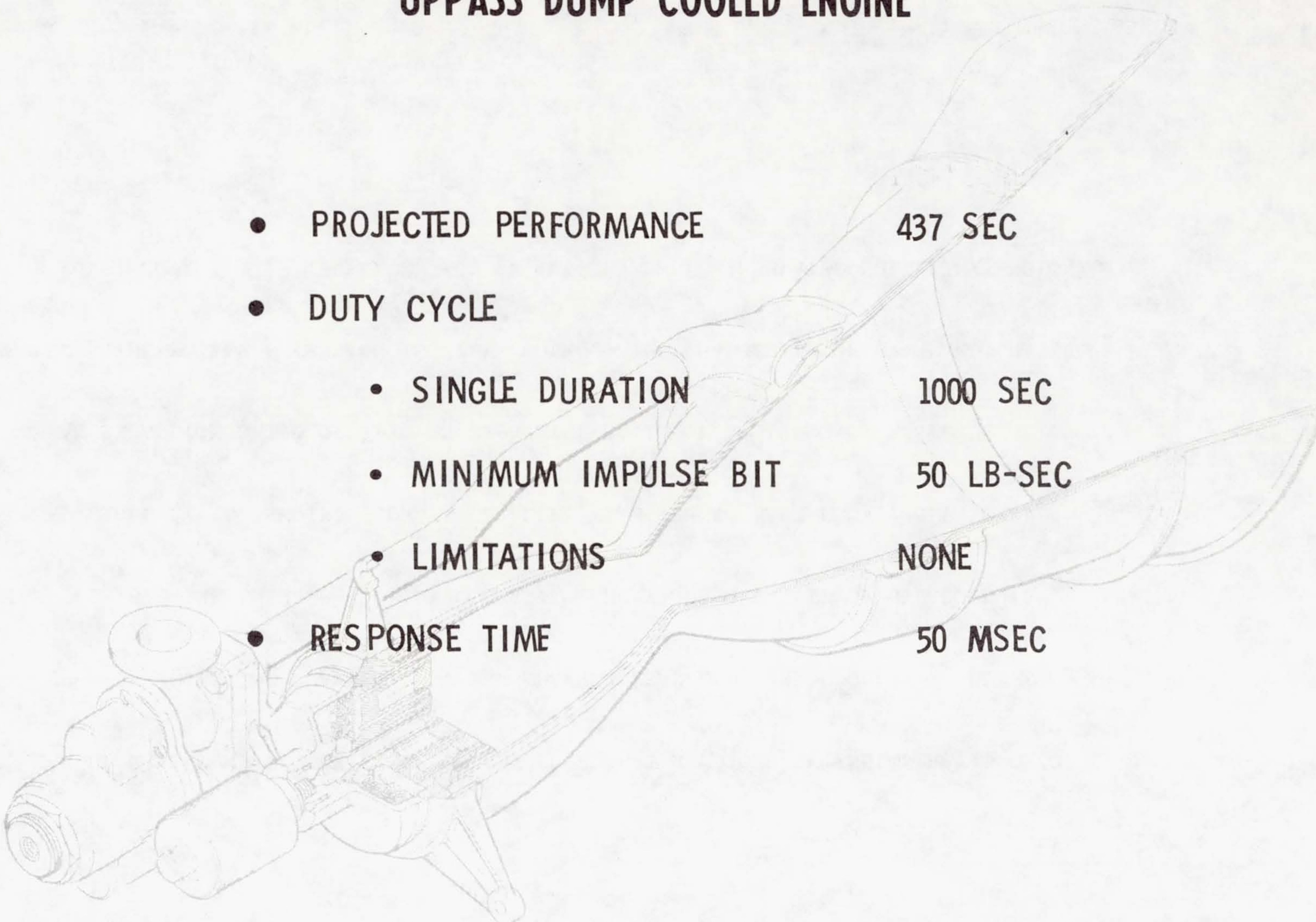
LIFE CAPABILITY
PULSE CYCLES WITH • 2000 STEADY STATE CYCLES • 50 HOURS

Rocketdyne
North American Rockwell

THE UP-PASS DUMP COOLED ENGINE WAS DESIGNED AND ANALYZED IN A MANNER SIMILAR TO THE REGENERATIVE ENGINE. A PERFORMANCE OF 437 SECONDS SPECIFIC IMPULSE IS PREDICTED.

THE LIFE AND RESPONSE GOALS WILL BE EXCEEDED.

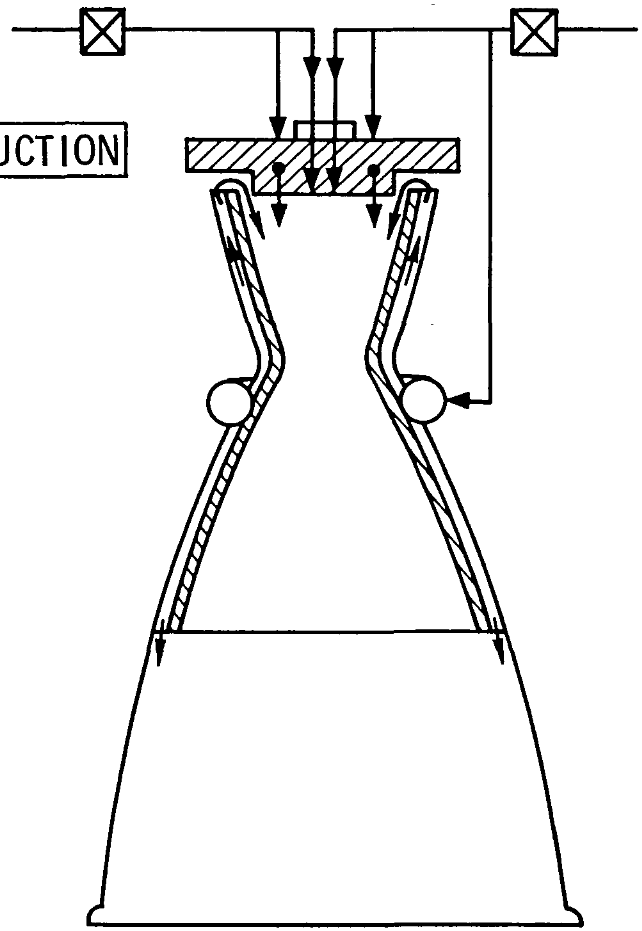
UPPASS DUMP COOLED ENGINE



- PROJECTED PERFORMANCE — 437 SEC
- DUTY CYCLE
  - SINGLE DURATION — 1000 SEC
  - MINIMUM IMPULSE BIT — 50 LB-SEC
  - LIMITATIONS — NONE
- RESPONSE TIME — 50 MSEC

IN THE UP-PASS DUMP ENGINE, THE COOLANT ENTERS AT A NOZZLE EXTENSION RATIO OF 3 AND UP-PASS COOLS THE CHAMBER. AT THE CHAMBER HEAD END, THE COOLANT IS DUMPED DIRECTLY INTO THE CHAMBER AS FILM COOLANT. THE MAJOR QUANTITY OF HYDROGEN IS FLOWED DIRECTLY TO THE INJECTOR IN PARALLEL WITH THE COOLANT FLOW. THE NOZZLE FROM AN EXPANSION RATIO OF 3 TO AN EXPANSION RATIO OF 18 IS DOWN PASS DUMP COOLED. THE NOZZLE EXTENSION IS FILM COOLED. THE CHAMBER IS OF STANDARD CHANNEL WALL CONSTRUCTION WITH THE EXCEPTION OF THE THIN WALL INSULATED NOZZLE EXTENSION.

UPPASS DUMP CHAMBER DESIGN

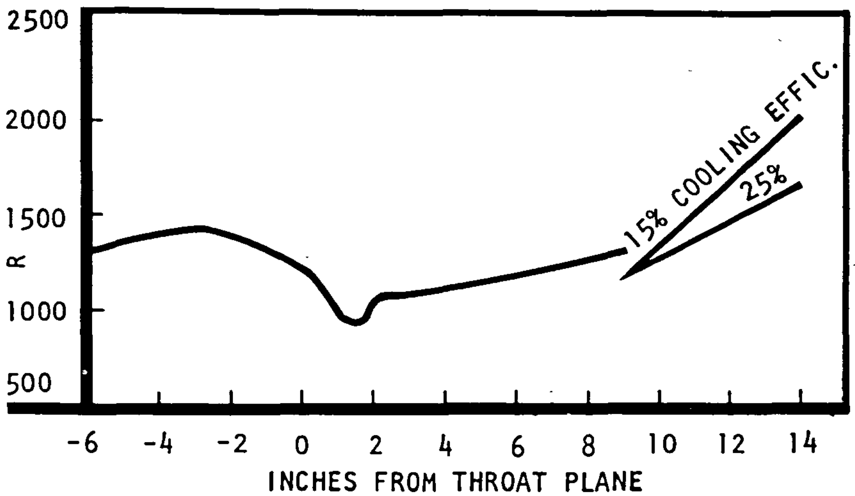


CHAMBER CONSTRUCTION	COOLING
CHANNEL	UPPASS DUMP FILM
	DUMP
INSULATED WALL	FILM

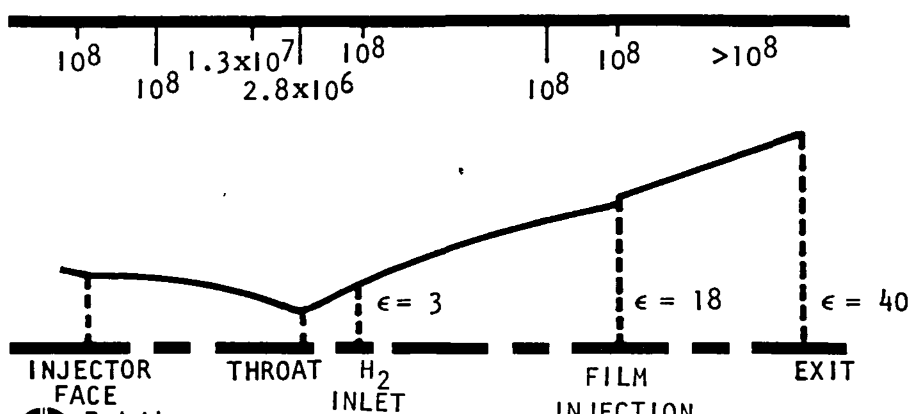
DETAILED ANALYSIS OF THE CHAMBER BASED ON THE EXPERIMENTAL HEAT TRANSFER RESULTS SHOWS SIMILAR HOT WALL CHARACTERISTICS IN THE THROAT REGION TO THE REGENERATIVE DESIGN. NEAR THE INJECTOR THE WALL TEMPERATURES ARE HIGHER BECAUSE THE SMALL AMOUNT OF COOLANT INCREASES MORE RAPIDLY IN TEMPERATURE. HOWEVER HEAT FLUX LEVELS ARE LOW IN THIS SECTION OF THE CHAMBER SO THAT THE STRAIN LEVEL IS LOW AND A HIGH CYCLE LIFE IS OBTAINED. THE LIFE LIMITING PLANE IN THE CHAMBER IS AT THE THROAT AND HAS THE CAPABILITY FOR 2.8 MILLION PULSE CYCLES. IN ADDITION TO 2000 STEADY-STATE CYCLES AND A 50 HOUR DURATION. THIS DESIGN HAS THE CAPABILITY OF APPROXIMATELY 100,000 FULL STEADY-STATE CYCLES IN ADDITION TO ONE MILLION PULSE CYCLES AND AN OPERATING DURATION OF 50 HOURS.

UPPASS DUMP COOLED THRUST CHAMBER

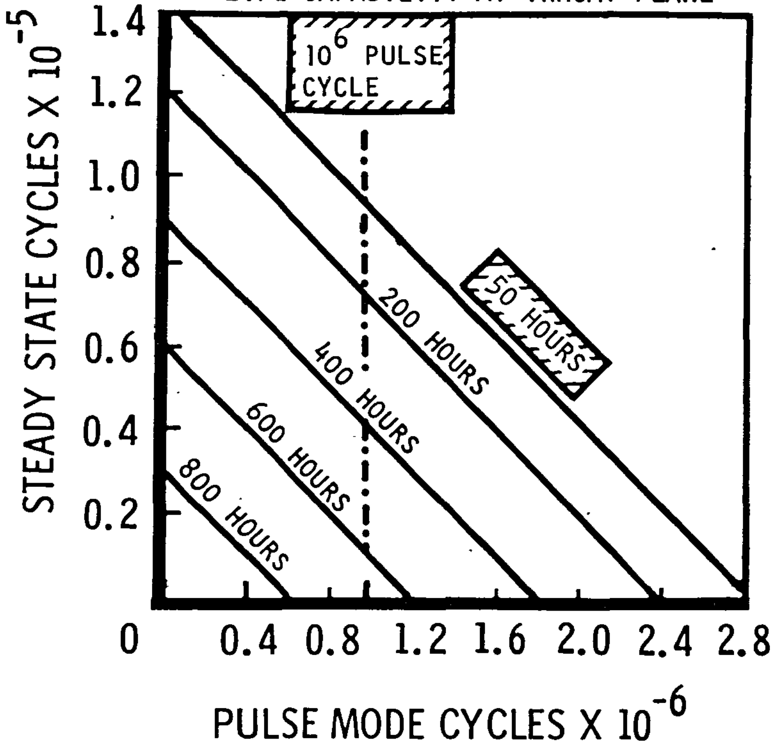
T/C HOT WALL TEMPERATURE



PULSE CYCLE LIFE WITH 2000 FULL CYCLES, 50 HOURS



LIFE CAPABILITY AT THROAT PLANE



Rocketdyne
North American Rockwell

THE MARGIN CAPABILITY OF THE UP-PASS DUMP ENGINE WAS ANALYZED IN A MANNER SIMILAR TO THAT OF THE REGENERATIVE ENGINE. INCREASING THE CHAMBER PRESSURE TO 500 PSI RESULTS IN THE CAPABILITY OF OVER ONE-HALF MILLION PULSE CYCLES. INCREASING THE PROPELLANT MIXTURE RATIO TO 5 AND THE PROPELLANT TEMPERATURES TO 800 R RESULT IN ONLY MINOR LIFE LOSSES. A STREAK IN THE CHAMBER EQUIVALENT TO COMPLETELY LOSING THE FILM COOLING EFFECT RESULTED IN A LIFE OF ONLY SLIGHTLY UNDER ONE-MILLION PULSE CYCLES. AS WITH THE CASE OF THE REGENERATIVE ENGINE THIS SHOWS THE INHERENT LONG LIFE CAPABILITIES OF THE PRESENT DESIGN.

UPPASS DUMP ENGINE
MARGIN CAPABILITY

OFF-LIMIT CONDITION	THROAT TEMPERATURE, R	LIFE CAPABILITY CYCLES
NOMINAL	1200	2.8×10^6
P_C = 500 PSIA	1250	5.1×10^5
MR = 5	1290	2.8×10^6
INLET TEMP. = 800 R	1390	2×10^6
STREAK - Q/A UP 25%	1380	7.3×10^5

NOMINAL CONDITIONS
• P_C = 300 PSIA • MR = 4.0 • INLET TEMP. = 540 R

LIFE CAPABILITY
PULSE CYCLES WITH • 2000 STEADY STATE CYCLES • 50 HOURS

Rocketdyne
North American Rockwell

THIS SERIES OF PICTURES IS OF THE UP-PASS DUMP CHAMBER BEING FABRICATED. THIS LOW COST WELL CONTROLLED PROCESS HAS BEEN UNDER DEVELOPMENT FOR SEVERAL YEARS AND HAS BEEN USED TO FABRICATE A NUMBER OF THRUST CHAMBERS IN THIS SIZE RANGE AND IN LARGER SIZES. AFTER PROPER INSPECTION A NARloy-Z LINER IS MACHINED AND SLOTTED USING HIGH PRECISION TEMPLATES. AN ELECTROFORMED NICKEL CLOSEOUT IS ADDED, THE OUTER DIMENSIONS ARE MACHINED, THE INLETS TO EACH SLOT ARE CUT, THE ELECTROFORMED WAX REMOVED, AND THE NICKEL IS ANNEALED. FINALLY, THE MANIFOLDS AND FLANGES ARE ADDED. THE CHAMBER IS NOW READY FOR INSTALLATION OF INSTRUMENTATION AND FOR TEST.

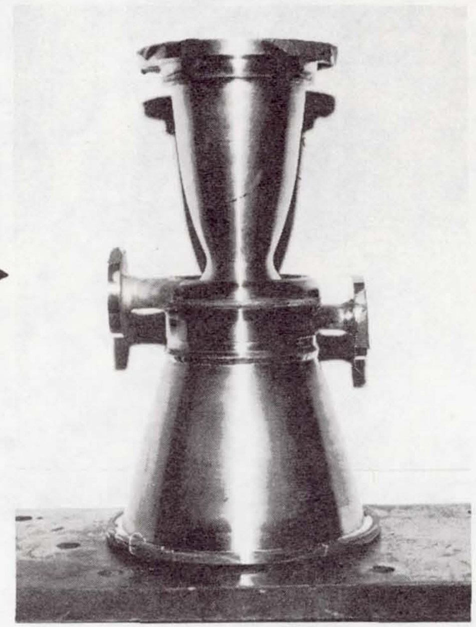
CHAMBER FABRICATION SEQUENCE

ELECTROFORM

CLOSEOUT

ADD

MANIFOLDS



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3-71

UPPASS DUMP ENGINE

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"HYDROGEN/OXYGEN ACPS ENGINES"

R. KLAUS

AEROJET

TECHNICAL MANAGER

J. W. GREGORY

LEWIS RESEARCH CENTER

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HYDROGEN/OXYGEN APS ENGINES

Contract NAS 3-14354

NASA/LeRC Program Manager	J. Gregory
Aerojet Program Manager	R. LaBotz (Speaker)
Aerojet Project Engineers	
High Pressure	L. Schoenman
Low Pressure	A. Oare

AEROJET LIQUID ROCKET COMPANY
SACRAMENTO, CALIFORNIA • A DIVISION OF AEROJET-GENERAL

AGCS 0100-71(REV 5/70)

This presentation covers both the high and low chamber pressure APS programs at Aerojet. The majority of the discussion will center on the High P_c program as the low pressure work was suspended at an early stage in the test program. At this point in time, the high pressure program has completed the injector characterization testing and is preparing to begin cooled chamber testing. Test results from the injector characterization tests on the high pressure program and a limited number of low pressure injector tests will be presented as well as a brief review of the cooled chamber designs. The igniter work performed as part of the engine contracts will not be discussed because of the similarity of this work with that being reported as part of the Aerojet ignition contract.

HYDROGEN/OXYGEN APS ENGINES

HIGH PRESSURE

- Injectors
- Performance
- Heat Transfer
- Combustor Stability
- Cooled Chambers

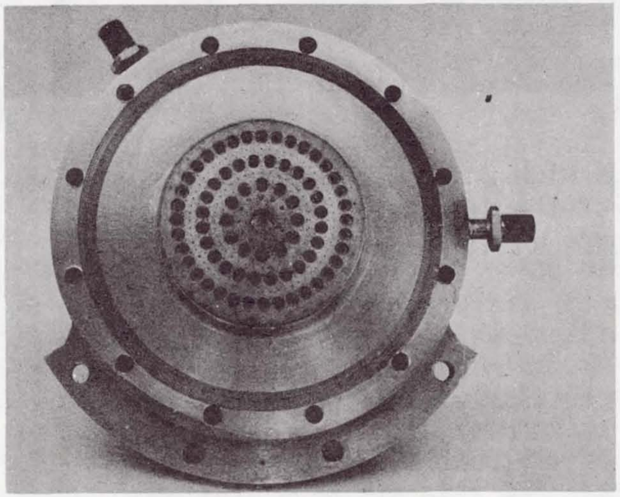
LOW PRESSURE

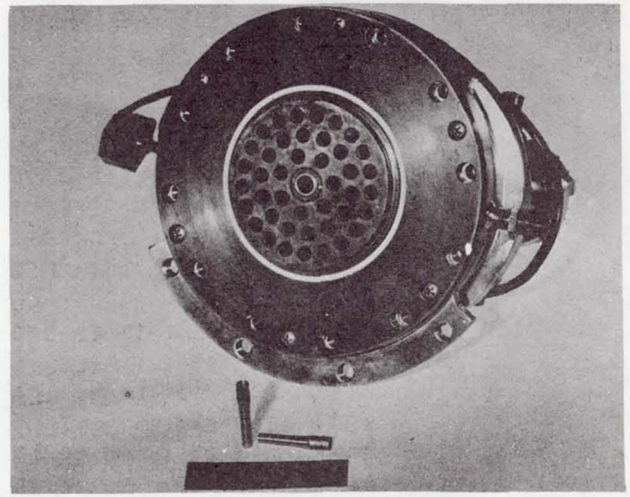
- Injectors
- Performance
- Heat Transfer

The two injector concepts tested during the High Pressure injector characterization testing are the coaxial injector and the impinging coaxial injector. The coaxial injector design is a conventional concentric element design employing a solid face plate and was run both with and without swirlers in the oxygen elements. The impinging coaxial injector is a relatively new injector design which employs a machined manifold structure with pattern-controlling photoetched face plates diffusion bonded to it. The basic impinging coaxial injector pattern employs axial oxygen streams which interact with fuel streams flowing parallel to the injector face. Two impinging coaxial patterns were tested: a triplet and an "I" triplet. The difference between these two patterns arises from the cross-sectional shape of the fuel streams as they impinge on the oxygen. Both the coaxial and impinging coaxial injector types are designed to accept center-mounted torch igniters.

The test program concentrated on the impinging coaxial design, as it was found to provide higher performance in a shorter chamber with lower heat fluxes than the coaxial. Face cooling on both injector types has been demonstrated to be consistent with the high cycle life requirements. Most tests were run with a 1.0 to 2.0 second duration, with thermal steady state (as indicated by measured injector face temperatures) being achieved in about 0.2 second. The high number of restarts accumulated on the impinging coaxial injector resulted from its use in the 2500-pulse test series reported in the ignition program presentation. Excellent injector durability was demonstrated in the injector characterization testing. The only injector damage encountered occurred during the stability bomb tests in which shrapnel from the exploding bomb imbedded in both the injector face and the combustion chamber wall.

HIGH PRESSURE INJECTORS





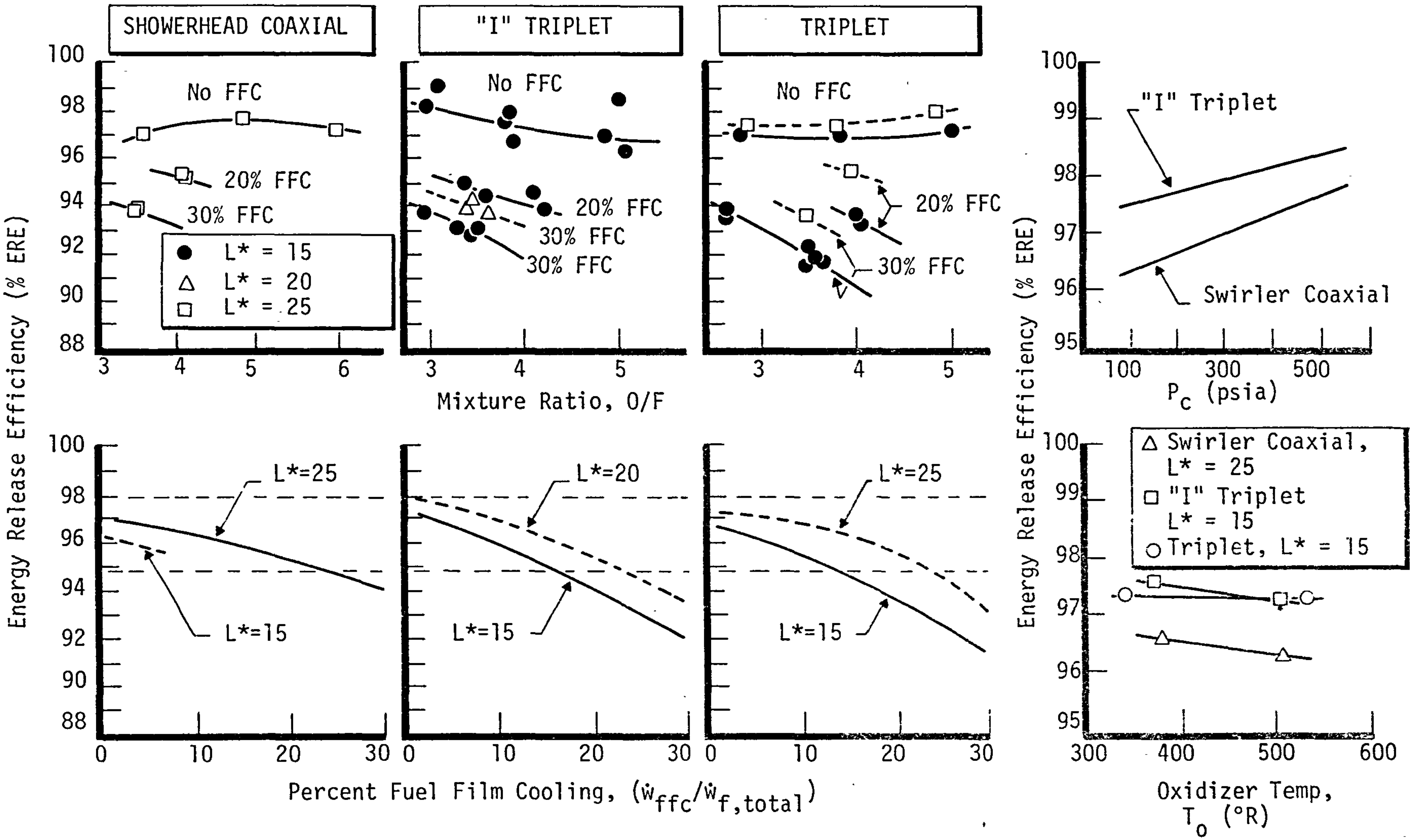
IMPINGING COAXIAL		COAXIAL
Impingement	Mixing Mechanism	Shear
72	No. Elements	42
Convection/Transpiration	Face Cooling	Conduction/Convection
Nickel	Face Material	Cu, Al
82	No. Tests	29
10 sec	Max Duration	2 sec
~2800	No. Restarts	29
100 → 500	P_c (psia)	100 → 500
3 → 6	MR	3 → 6

Some of the injector characterization test results are presented in the three plots of injector energy release efficiency (ERE) versus mixture ratio. These injector performance values are based on thrust measurements using the standard JANNAF methodology. The performance with film cooling was calculated as though the film coolant were passing through the injector, so the mixture ratio distribution loss (MRD) resulting from the use of film cooling shows up as a decrease in ERE. The data show the "I" triplet impinging coaxial design to give generally higher performance with shorter L* chambers than the other injectors. All three injectors shown exceed the contract goal of 97% ERE at MR = 4.0.

An important factor for chamber cooling is how much film cooling can be employed with a given injector and still meet the contract performance goal of 435 sec I_s. This is shown in the lower three plots in which the lower dashed horizontal line is the minimum ERE consistent with the I_s goal. These plots show, for example, that the "I" triplet injector in a 20 L* configuration can operate with 24% film cooling and still meet the performance goal. All of the lower plots are based on a P_c = 300 psia and a MR = 4.0.

The two right plots on the right side of the figure show the influence of chamber pressure and oxidizer temperature on performance. From these plots, it can be concluded that the Aerojet injector designs show performance improving with increasing pressure and unaffected by decreasing oxygen temperature. Tests with both propellants cold (350°R) also indicate no degradation from ambient performance.

HIGH PRESSURE INJECTOR PERFORMANCE



The heat flux data obtained with the triplet patterned impinging coaxial injector using copper heat sink chambers is shown for two different chamber L*'s. These curves show the axial heat flux distribution for various quantities of film cooling. The axial stations to the left of the throat are upstream, while those to the right of the throat are downstream. All film cooling data were obtained with a film cooling injection sleeve which extended 2-1/2 in. downstream of the injector face. These data and those on the following figure form the basis for the cooled chamber designs, particularly those employing regenerative cooling.

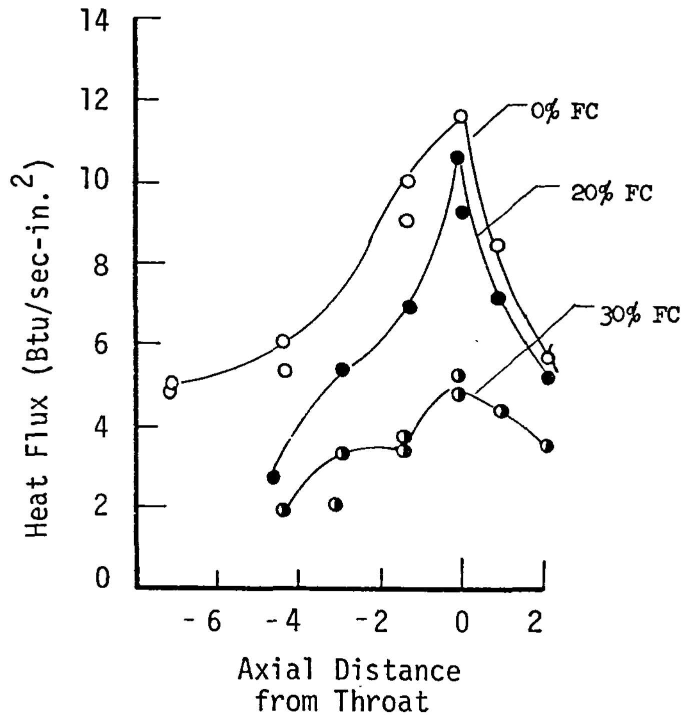
The throat heat fluxes without film cooling are markedly lower for the long chamber compared to the short chamber, probably reflecting the benefits of more boundary layer development and greater damping of combustion turbulence. The heat fluxes with film cooling are comparable for the two chamber lengths. It is interesting to note that, particularly with the shorter chamber, very little benefit is gained in terms of heat flux reduction from using small quantities of film cooling. Apparently the reduction in recovery temperature resulting from the presence of the hydrogen film coolant in the boundary layer is offset by the increased thermal conductivity and specific heat of the boundary layer, the net result being almost no change in heat flux.

HIGH PRESSURE HEAT TRANSFER SUMMARY

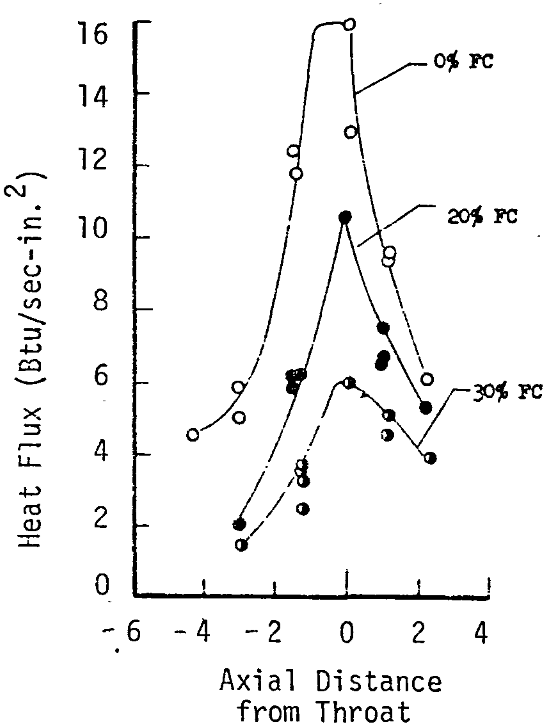
IMPINGING ELEMENT INJECTOR

2.5 in. Film Cooling Ring

L* = 25 in.



L* = 15 in.

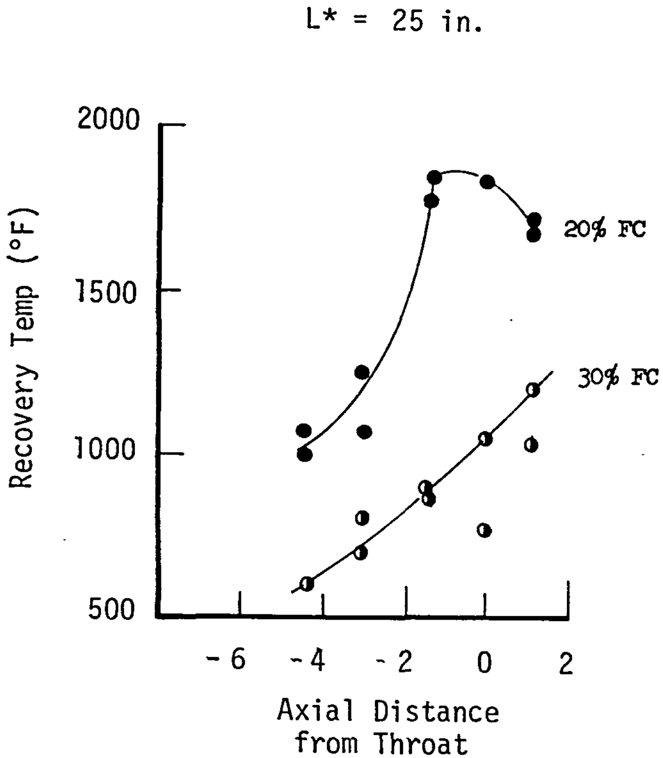


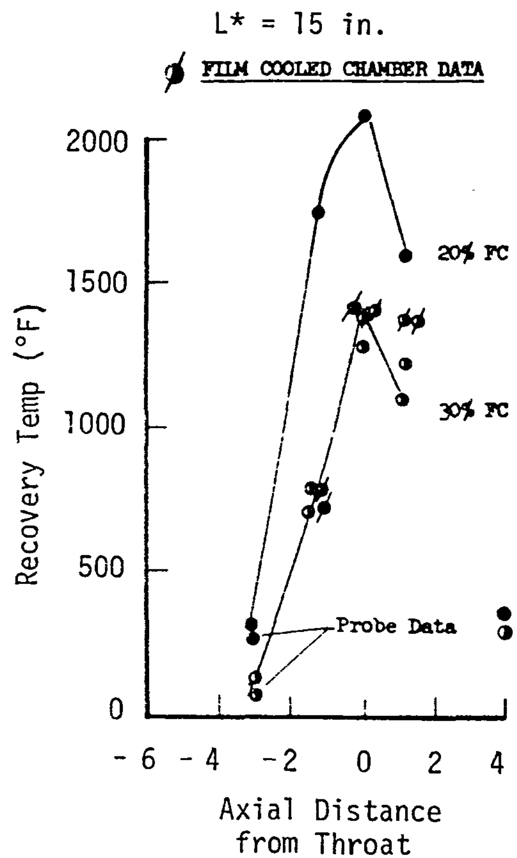
In the design of cooled combustion chambers which employ adiabatic wall designs, one is more interested in recovery temperatures than heat flux. Recovery temperature data for the same injector/chamber/film cooling ring combination presented in the previous figure are presented here. Again, the longer chamber is more favorable from the heat transfer viewpoint than the short chamber. The data indicate the feasibility of an adiabatic throat chamber using conventional superalloys. The general trend of recovery temperatures decreasing downstream of the throat is significant in that it implies that, for adiabatic wall designs, the film cooling requirements are established by the requirement of cooling the throat and that secondary injection of film coolant downstream of the throat is not required.

HIGH PRESSURE HEAT TRANSFER SUMMARY

IMPINGING ELEMENT INJECTOR

2.5 in. Film Cooling Ring





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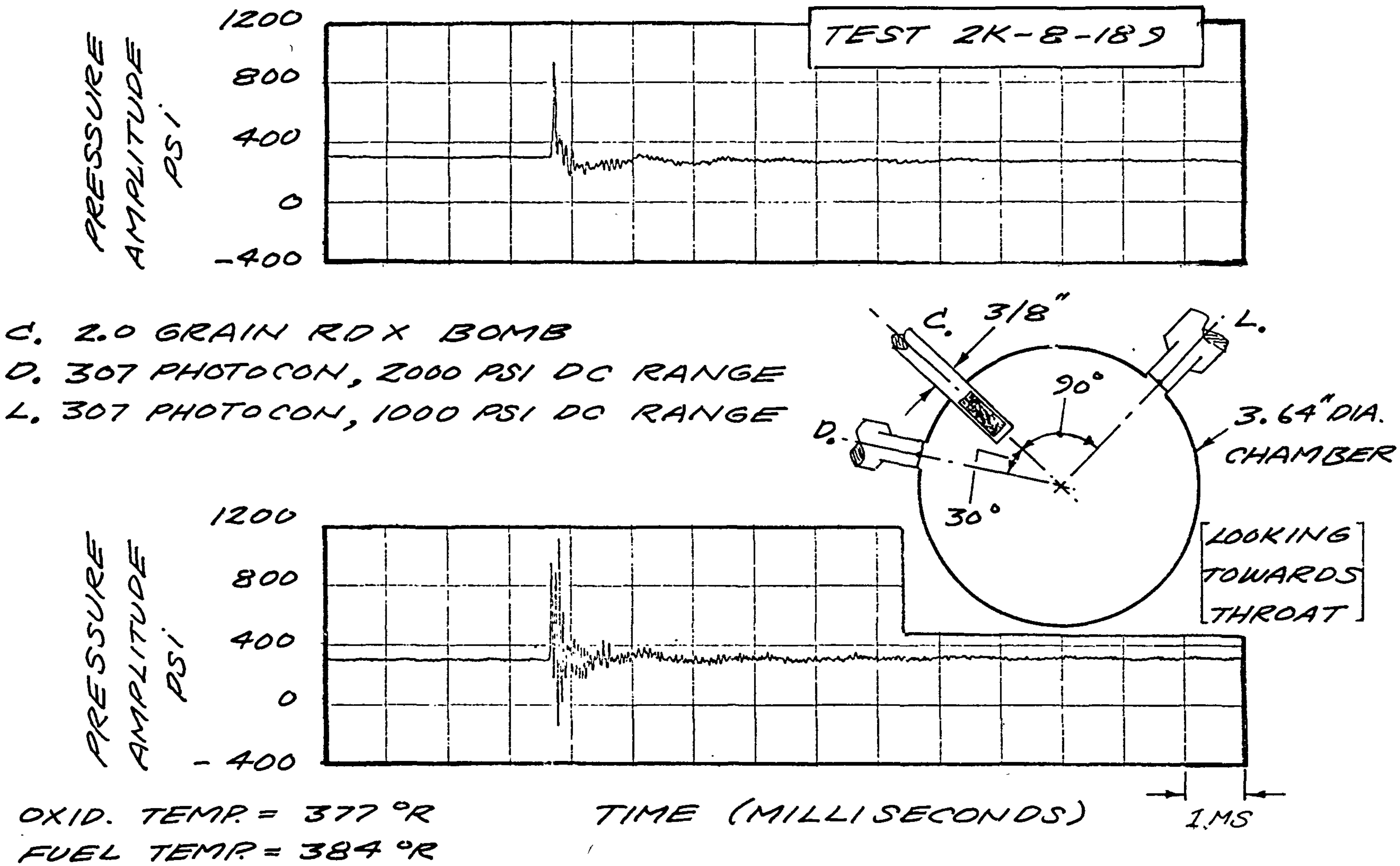
Limited combustion stability testing was conducted using an impinging coaxial injector with cold propellants. The propellant temperatures selected were those corresponding to the propellant temperatures entering the injector with the Aerojet cooled chamber designs when the thruster is being supplied $250^{o}R$ H_2 and $375^{o}R$ O_2. Four separate tests were conducted using nondirectional bombs in a chamber containing two flush-mounted high frequency response Photocon transducers. The injector was unbaffled and no acoustic resonators were employed.

The figure shows oscillograph traces from the two transducers for one of the tests, the time scale being 1 msec per division. It can be seen that, prior to the bomb discharge, the injector is very smooth running, the level of combustion noise being approximately $\pm 2\%$ of P_c. The bomb discharge introduces an overpressure in the combustion chamber of 200 to 300% of P_c, peak recorded pressures approaching 1200 psia. Within 0.0005 sec the large pressure disturbances have died out and in about 0.007 sec all evidence of the bomb discharge has disappeared. All four tests showed essentially the same behavior. While this cannot be constructed as a comprehensive stability demonstration, it does indicate a high level of dynamic stability at what constitutes the nominal low temperature operating condition.

COMBUSTION STABILITY BOMB TEST RESULTS



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AGCS 0100-71(REV 5/70)

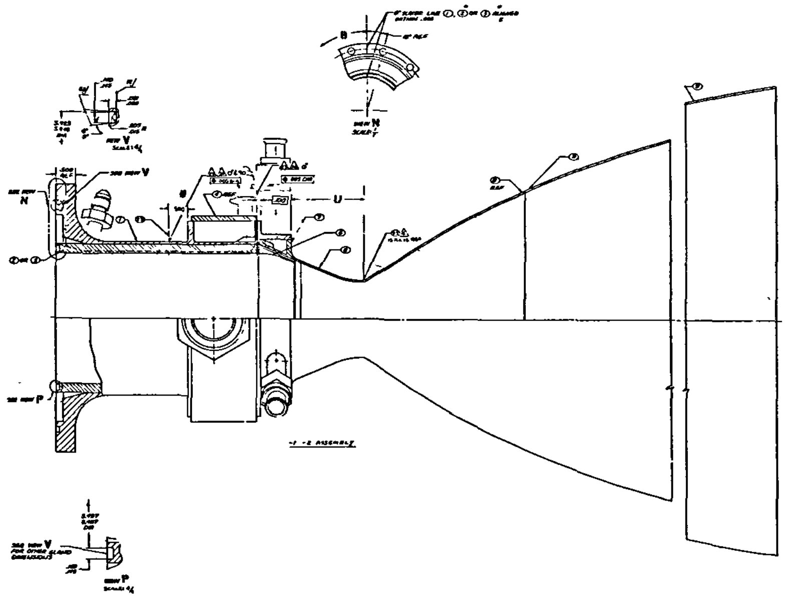
The first of Aerojet's cooled chamber designs employs a regeneratively cooled combustion chamber with a film-cooled throat and skirt. This design requires 22% of the fuel to be injected as film coolant through a film cooling sleeve in the converg-ing portion of the nozzle. The remainder of the fuel passes forward as regenerative coolant, through the cylindrical section of the chamber and into the injectors.

The thrust chamber is fabricated in three separate parts. The regeneratively cooled cylindrical portion is made of copper with machined axial slots forming the cooling passages. The passages are enclosed on the backside with a stainless steel sleeve. The throat section is spun thin-walled (0.050 in.) Haynes 188, which has the short film cooling sleeve brazed into it. The throat section and regeneratively cooled cylindrical section are joined by brazing. The skirt is spun from stainless steel and welded to the Haynes 188 throat. The Haynes 188 is required in the throat but not in the skirt because the peak stresses occur in the throat.

The advantages of this design are that it is lightweight, inexpensive, easily scarfed, has high response and low ΔP in the fuel circuit, and can tolerate extended exposure to high temperature diving re-entry. The predicted performance based on the measured injector performance is 439 sec I_s with a 40:1 nozzle at vacuum.

HIGH PRESSURE COOLED CHAMBER

FILM COOLED THROAT



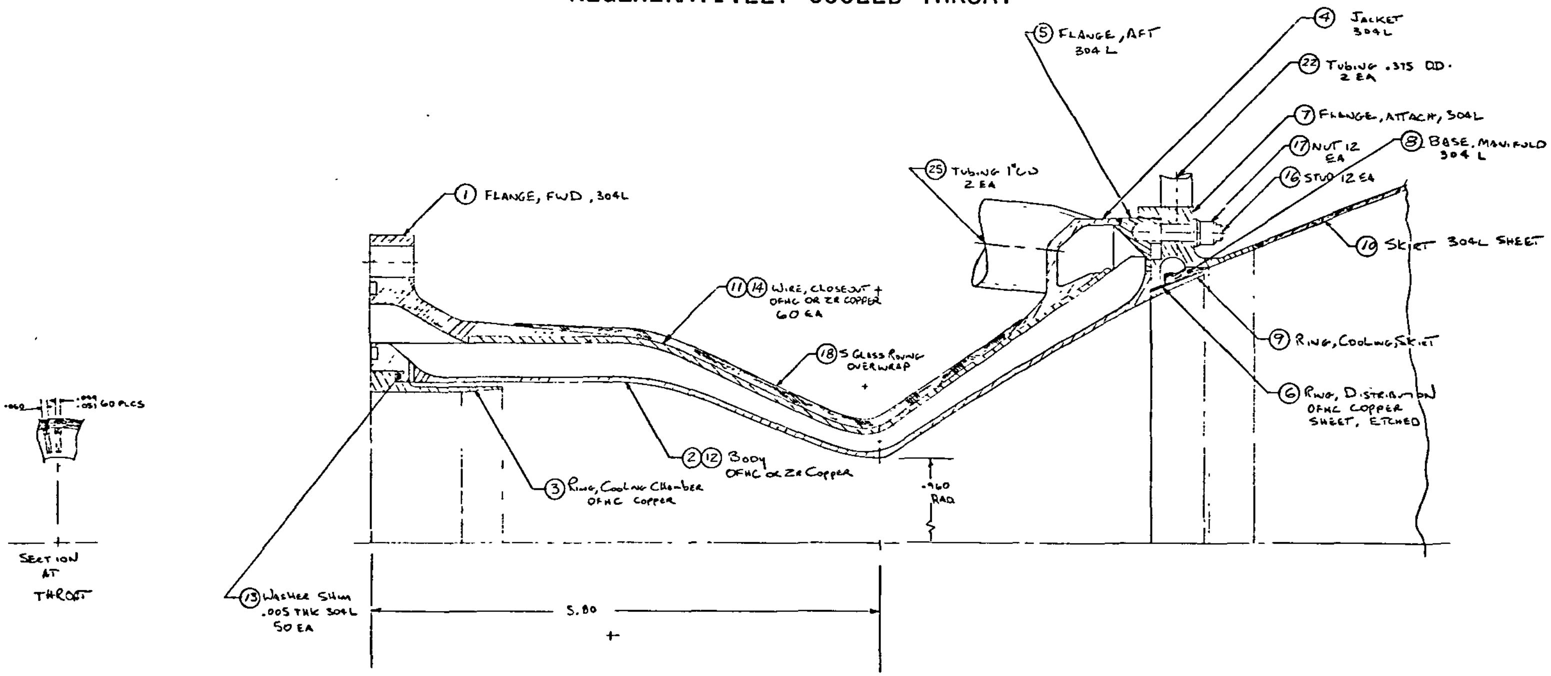
Required Fuel Film Cooling	22% of Fuel
Coolant ΔP	22 psi
I_s (98% ERE, ϵ = 40:1)	439 sec
Weight	15 lb

The second of Aerojet's cooled chamber designs is regeneratively cooled to an area ratio of 8.5:1 using a single up-pass system and film cooled downstream of that point. This design employs a total of 21% of the fuel as film coolant, 15% being injected at the injector face to enable the throat to meet the cycle life requirements and 6% being injected at the beginning of the film-cooled skirt. The regenerative section is made of slotted copper with the backside closeout being either electroformed copper or square copper wires brazed into the cooling slots. The film-cooled skirt is made of spun stainless steel.

This design is heavier than the film-cooled chamber and offers slightly higher performance (441 sec vs 439 sec). Its primary advantage relative to the film-cooled design is that it is probably more tolerant of brief excursions in operating conditions than the film-cooled design. It can tolerate limited re-entry heating and, like the film-cooled chamber, it is readily scarfed. The predicted coolant pressure drop (35 psi) will allow the engine to operate within its overall pressure schedule of 375 psia inlet pressure.

HIGH PRESSURE COOLED CHAMBER

REGENERATIVELY COOLED THROAT



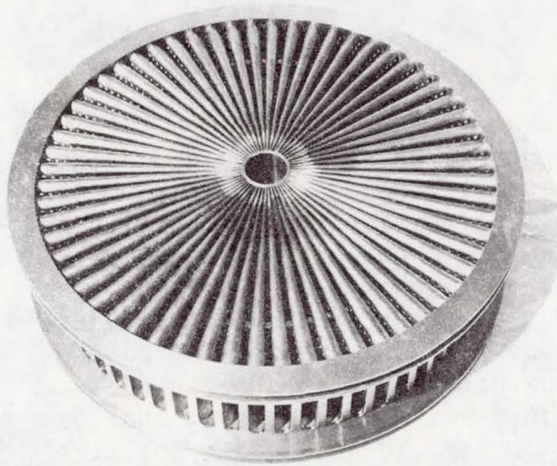
Required Fuel Film Cooling	15 + 6% of Fuel
Coolant ΔP	35 psi
I_s (98% ERE, ε = 40:1)	441 sec
Weight	27 lb

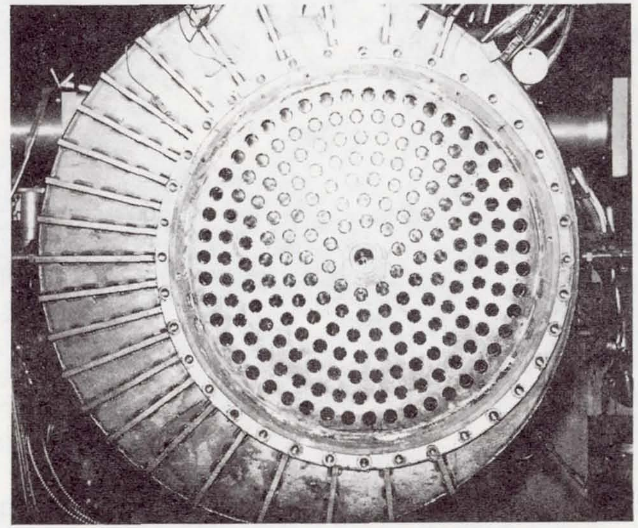
The two low pressure injector concepts employed in this program were a 200-element coaxial unit and a 3840-element vaned design. Because the low pressure portion of the program was terminated before the vaned injector fabrication was completed, only the coaxial injector was tested.

The coaxial injector oxidizer tubes were made of commercial tubing brazed in place. A solid copper face plate was used. Swirlers were used in all the oxidizer elements with the exception of the outer row which employed recessed oxidizer posts. The injector accepted a center-mounted torch igniter.

A total of 30 tests were conducted. All tests were run in a vacuum using a 5:1 nozzle with a maximum test duration of 10 seconds. The chamber employed was a thin-walled mild steel unit with circumferential and axial arrays of thermocouples. Film cooling rings of 0.5 in. and 2.7 in. length and designed for nominal flows of both 10% and 20% of the fuel were employed.

LOW PRESSURE INJECTORS

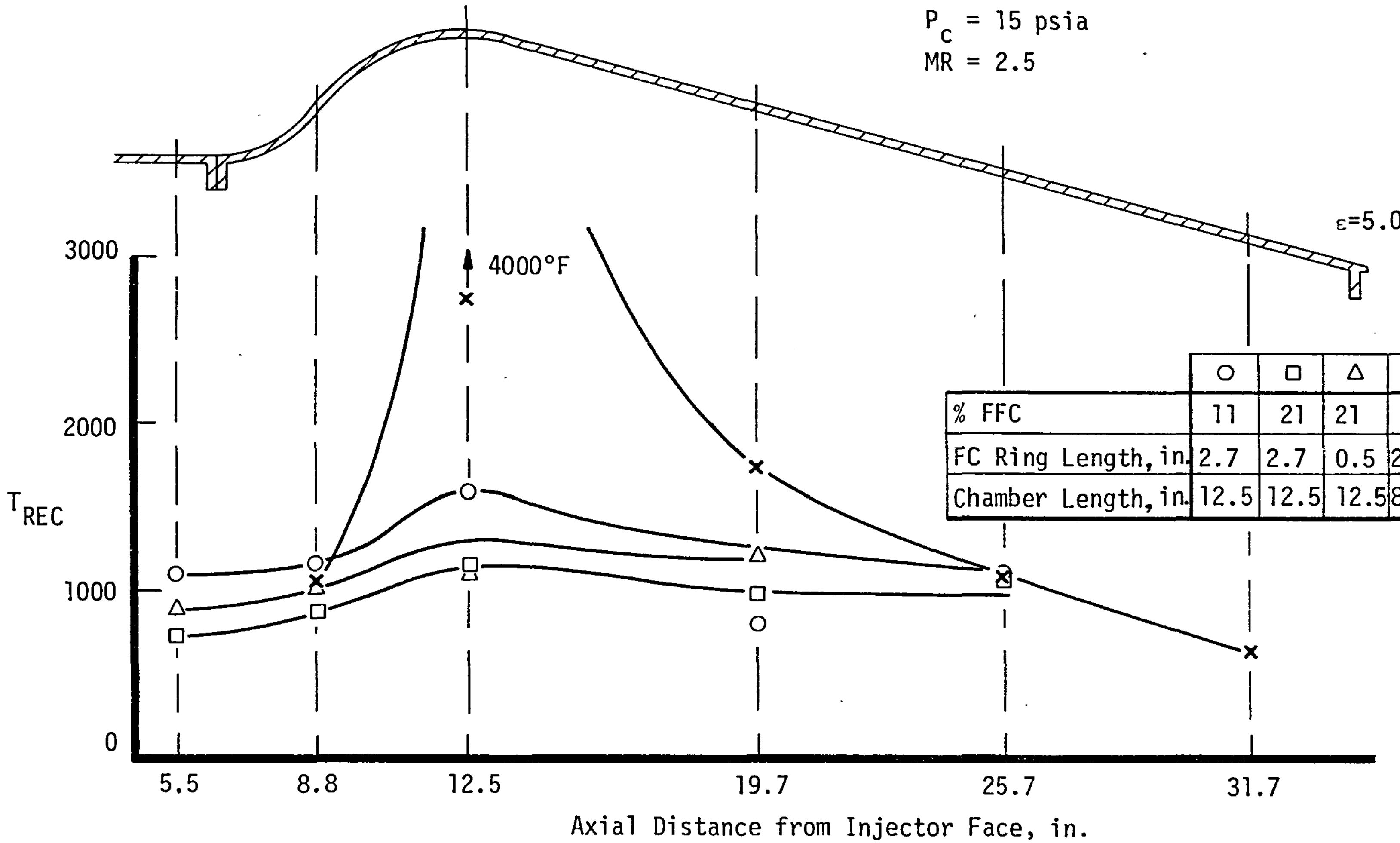




COAXIAL		VANED
Shear	Mixing Mechanism	Impingement
200	No. Elements	3840
Conduction/Convection	Face Cooling	Conduction/Convection
30	No. Tests	--
10	Max Duration	--
$10 \rightarrow 20$	P_c (psia)	--
$2 \rightarrow 4$	MR	--

The low pressure engine, with its low nominal MR, is particularly well suited to a film-cooled adiabatic wall chamber design. Data applicable to this type of chamber design are presented in the figure. This plot shows the adiabatic chamber wall temperature (°F) from near the injector out to the nozzle exit for various quantities of fuel film cooling. The chamber contour is shown above the plot. It is interesting to note that the data are consistent with those obtained on the high pressure engine in that the shortest chamber (8.5 in.) gives the highest throat temperatures and also that temperatures drop downstream of the throat. The conclusion one draws from these data is that with proper design one might construct a thin-walled chamber employing only 11% of the fuel as coolant and have peak wall temperatures of approximately 1500°F while, with 21% cooling, this temperature can be lowered to 1200°F.

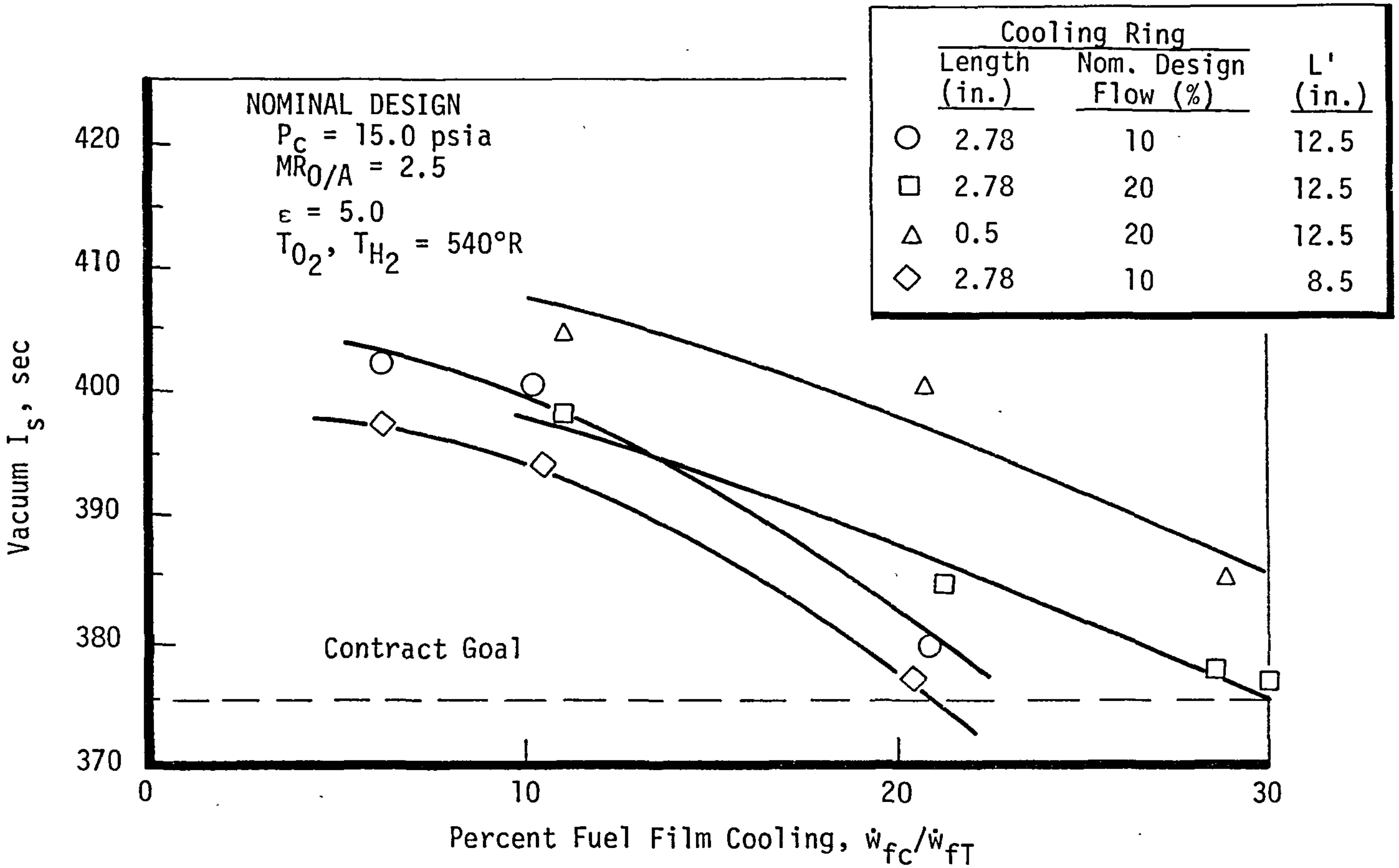
LOW PRESSURE CHAMBER HEAT TRANSFER



	○	□	△	×
% FFC	11	21	21	11
FC Ring Length, in.	2.7	2.7	0.5	2.7
Chamber Length, in.	12.5	12.5	12.5	8.5

This plot summarizes the flow pressure performance data obtained at the nominal engine operating point of P_c = 15 psia, MR = 2.5. The data are presented in terms of measured I_s versus percent of fuel used as film coolant. It is interesting to note that every data point falls above the contract performance goal of 375 sec I_s. By combining the data of this plot with the heat transfer data given in the previous figure, it is possible to obtain a realistic performance estimate for a lightweight, adiabatic wall, film-cooled low pressure engine. The condition of chamber length and film cooling ring, which gave the 1500°F maximum wall temperature at 11% film cooling, is represented by the circles. This gives a measured I_s of 399 sec. If the 1200°F operating condition were desired, the performance curve to be used is given by the triangles. At 21% film cooling, this gives an I_s of 396 sec, again well in excess of the goal of 375 sec.

LOW PRESSURE ENGINE PERFORMANCE



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"HIGH PRESSURE REVERSE FLOW ACPS ENGINE"

J. SENNEFF

BELL AEROSPACE COMPANY

TECHNICAL MANAGER

S. M. COHEN

LEWIS RESEARCH CENTER

The High Pressure Reverse Flow APS Engine program is in progress to assess the capability of the reverse flow engine concept for Space Shuttle APS application. Subsequent figures will present the engine configuration and the program accomplishments to date.

The contract requirements include operation with gaseous H_2 and O_2 over a mixture ratio of 3.0 to 5.0, 1500 lbs thrust at a nominal chamber pressure of 300 psia and a nozzle area ratio of 40:1. The nominal propellant feed temperature is 540°R. Tests are also required at 800 and 300°R propellant feed temperatures and chamber pressures from 100 to 500 psia. Specific goals of the effort are the attainment of 97% C* and 435 seconds specific impulse with a thrust chamber assembly capable of performing one million firings.

Previous development work at Bell Aerospace demonstrated the reverse flow engine concept with a combustion efficiency of approximately 95% at a mixture ratio of 4:1. The initial test firings of the program have concentrated on oxidizer and fuel injection modifications to improve the percentage of theoretical C* achieved.

The development effort is limited to the thrust chamber assembly. Commercially available engine valving and ignition systems are employed for the thrust chamber testing.

HIGH PRESSURE REVERSE FLOW ACPS ENGINE

CONTRACT NO. NAS 3-14353

COGNIZANT AGENCY - LEWIS RESEARCH CENTER

CONTRACTOR - BELL AEROSPACE COMPANY

NASA PROJECT MANAGER - S. COHEN

BELL PROJECT MANAGER - J. SENNEFF

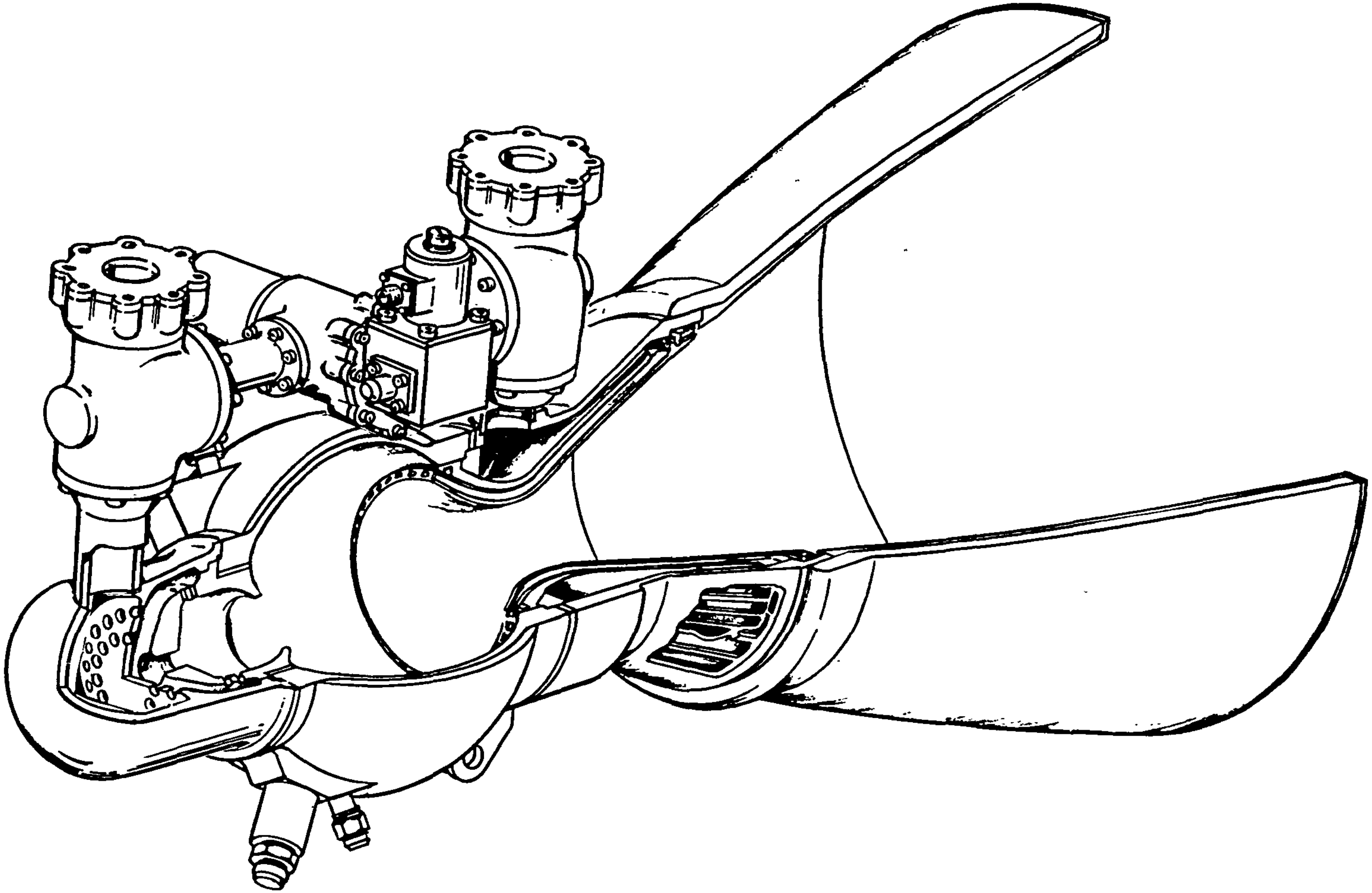
Bell Aerospace Company DIVISION OF textron

The rendering of a flight configuration reverse flow engine and the following figure show the unique propellant injection scheme. The combustion chamber is spherical with the oxidizer inlet in the conventional injector position. The O_2 gas passes through a baffle plate and feeds a single, relatively large diameter, swirl cup. The O_2 enters the combustion chamber in a divergent sheet through a large diameter swirl cup outlet. The fuel flows through regenerative cooling passages in the nozzle throat which are terminated in the convergent portion of the nozzle. The fuel is injected rearward forming a fuel film for the combustion chamber before being turned into the oxidizer flow at the head end of the chamber. The rearward injection of the fuel film is responsible for the term "Reverse Flow".

The ignition system includes a surface gap spark plug mounted in the combustion chamber wall toward the head end. O_2 augmentation of the plug is provided.

The flight configuration engine regeneratively cooled nozzle section extends to an area ratio of 10:1. The insulated nozzle extension is dump cooled with a small percentage of the H_2 feed to the engine. The dump cooling is injected through slots in the aft end of the nozzle section.

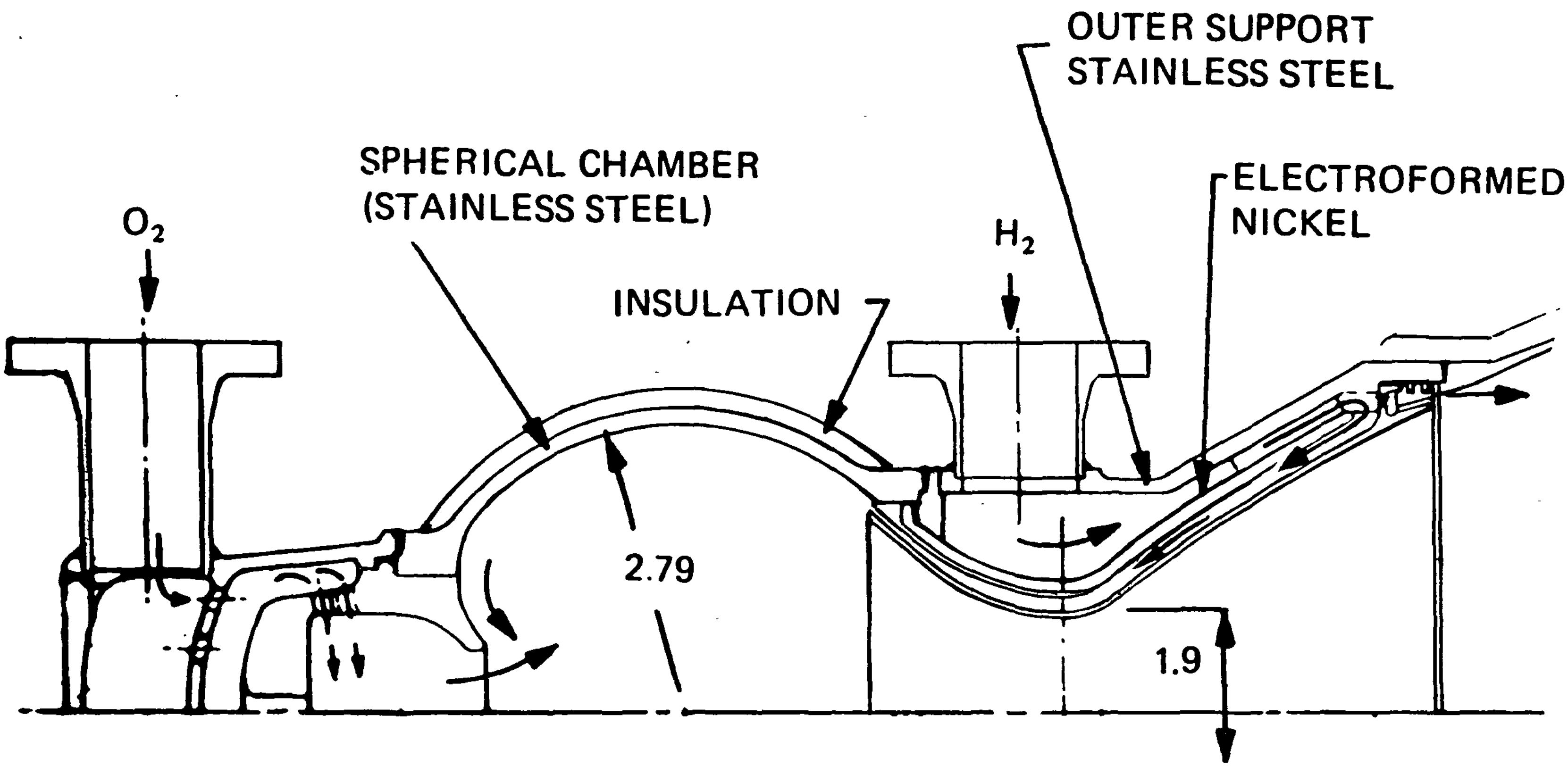
REVERSE FLOW ENGINE



The reverse flow engine has several advantages relative to engines with conventional injector schemes:

1. The fuel and oxidizer are injected at different locations allowing a significant simplification in injector design. The injector joints are few in number, all are welded and all are fully inspectable.
2. The combustion chamber walls are film cooled with the total H_2 combustion flow. The externally insulated combustion chamber shell reaches steady state temperatures well within the capability of conventional materials of construction.
3. The relatively low temperature single wall combustion chamber allows flexibility for the installation and positioning of the ignition system.
4. The nozzle section liner longitudinal thermal stresses are independent of the combustion chamber improving the life cycle capability of the liner.

1500 LBF REVERSE FLOW FLIGHT TYPE THRUST CHAMBER ASSEMBLY



The photograph shows a 1000 lb. thrust gaseous O_2/H_2 reverse flow engine which was recently prepared for altitude testing at NASA Marshall Space Flight Center, Huntsville.

The 1000 lb. thrust assembly was employed for sea level test firings at Bell in 1969 to demonstrate the reverse flow engine at relatively high chamber pressure. The test program included operation from 35 to 150 psia with heat sink test hardware followed by testing from 150 to the nominal chamber pressure of 250 psia with a regeneratively cooled nozzle section. Pulse mode operation (200 ms) and 120 second duration firings were conducted. The maximum combustion chamber wall temperatures recorded during steady state operation at a mixture ratio of 4.0 and Pc of 250 psia were 1200°F.

The original reverse flow engine testing was carried out at 25 lb. thrust and a chamber pressure of 50 psia with gaseous O_2/H_2 and F_2/H_2 propellants. Additional development work of the 25 lb. F_2/H_2 configuration was conducted under Air Force contract F04611-70-C-0026. A F_2/H_2 engine demonstrated 98-99% of shifting C* at an O/F of 10:1. Steady state combustion chamber temperatures of 2100°F maximum were observed during tests of up to 32 seconds duration.

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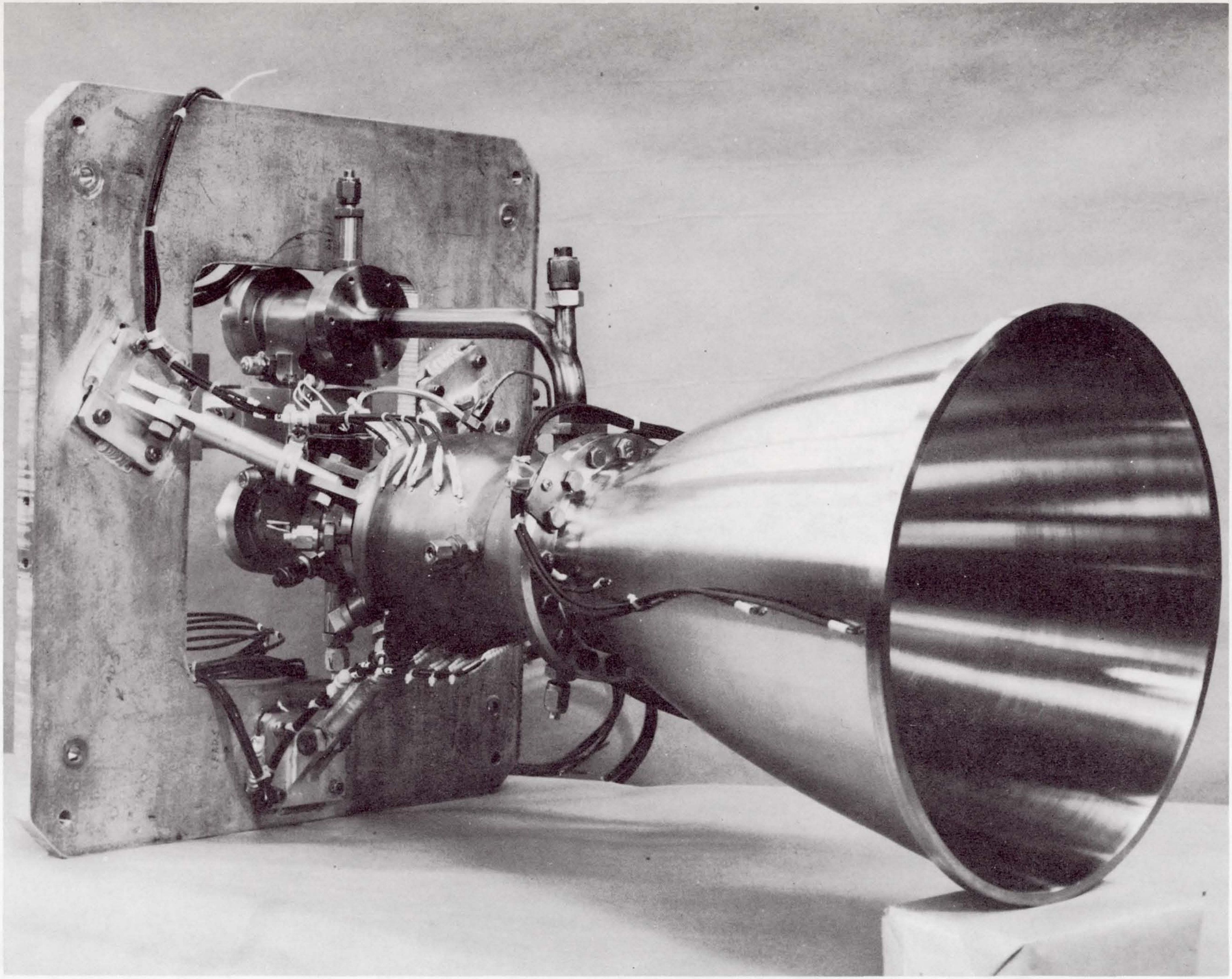


The effort accomplished to date under contract NAS 3-14353 is summarized above. Subsequent pages will provide additional details of the test firing results, thrust chamber analyses and the hardware to be tested at simulated altitude conditions.

1500 LB THRUST PROGRAM ACCOMPLISHMENTS

- $>$ 97 % c* DEMONSTRATION
- ANALYSES OF THRUST CHAMBER DESIGN SHOWS LIFE CYCLE REQUIREMENTS EXCEEDED
- HARDWARE FABRICATION FOR CHAMBER TESTING AT ALTITUDE IN PROGRESS

HEAT SINK THRUST CHAMBER ASSEMBLY

The heat sink thrust chamber assembly employed for the injector testing to date consists of five major subassemblies:

1. A stainless steel spherical chamber cooled with 100% of the fuel. A flush mounted igniter assembly with oxidizer augmentation is incorporated at the forward position of the chamber.
2. A stainless steel vortex cup oxidizer injector
3. A stainless steel fuel manifold
4. A copper fuel injector
5. A conical copper nozzle with a sea level expansion ratio of 3.17

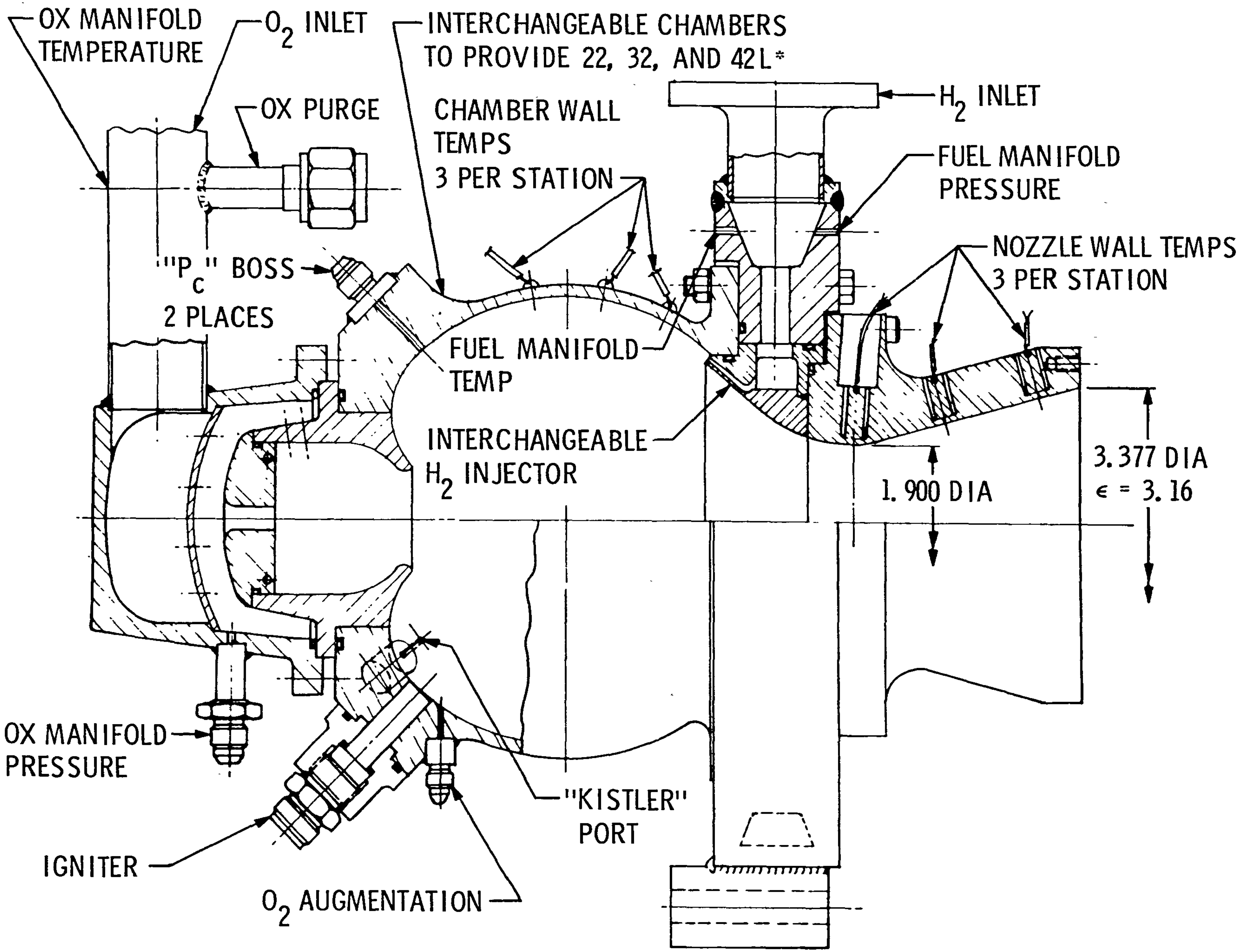
This hardware is designed for complete flexibility to provide for interchangeability of the spherical chambers, oxidizer injectors and fuel injectors.

Heat flux data is obtained in the nozzle at three (3) axial thermocouple stations and at each of two (2) circumferential locations.

Spherical chamber wall temperatures are obtained by twelve (12) thermocouples welded to the outer surface.

The basic "heat sink" feature is provided only in the fuel manifold, fuel injector and nozzle. The spherical chamber, oxidizer injector, and igniter assembly are reusable for the scheduled cooled chamber tests at simulated altitude conditions.

HEAT SINK THRUST CHAMBER



The injector test variables of the reverse flow engine include the geometrical characteristics of the oxidizer swirl cup, the fuel injection velocity and chamber characteristic length. The earlier 1000 lb. thrust reverse flow engine tests had established a baseline configuration. Variations were made above and below the baseline values.

The fuel injection mach number was adjusted by changing the height of the fuel injection slots. Fuel injectors were fabricated with design mach numbers of 0.40, 0.50 and 0.56 at rated fuel flow and fuel feed temperature, 0.69 lbs/sec and 540°R respectively.

The oxidizer cup designs included the following:

Swirl Hole Area, A_s, in^2, 0.294, 0.315, 0.421, 0.519
Outlet Diameter, D_2 inches, 1.189, 1.427, 1.660
Center Flow, % 1.5, 5.0, 10.0, 15.0

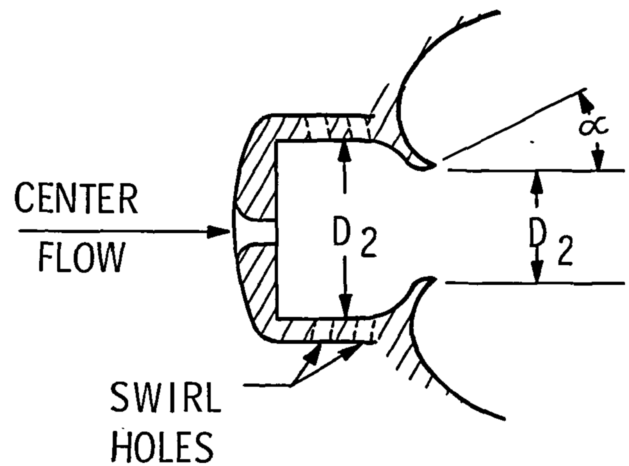
The values of A_s and D_2 (and the cup inside diameter) combine to give a measure of the angle of the hollow cone oxidizer flow, . The angle is reduced for increasing percentages of center flow. Cups were designed with the following values of the D_1, D_2/A_s ratio.

α	Angle	α	Angle
4.35	23.0°	8.61	34.0°
5.22	25.0	9.19	35.4
6.44	28.0	10.73	39.4
6.51	28.4		

The α - angle relationship is based on empirical swirl cup data for water flow. Gas flow predictions were checked at an α of 6.44.

Three spherical chambers were built to provide operation with L*'s of 22, 32 and 42 inches.

INJECTOR TEST HARDWARE VARIABLES	SYMBOL
• FUEL INJECTION VELOCITY	M_N
• OXIDIZER CUP SWIRL HOLE AREA	A_S
• OXIDIZER CUP OUTLET DIAMETER	D_2
• OXIDIZER INJECTION ANGLE	$\alpha - \frac{D_1 D_2}{A_S}$
• OXIDIZER CUP CENTER FLOW	CF
• CHAMBER CHARACTERISTIC LENGTH	L*



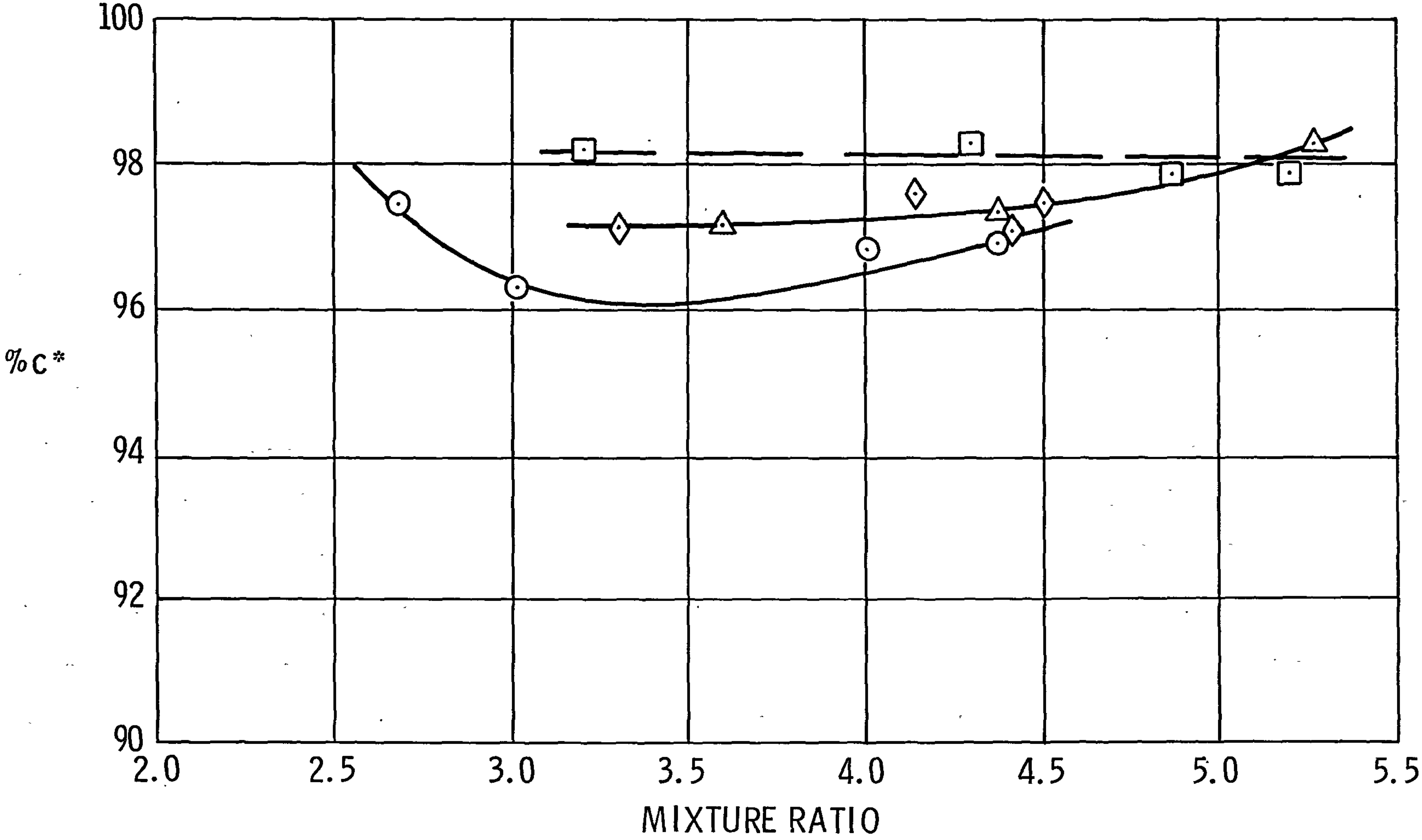
Each configuration of hardware variations was operated over a range of mixture ratio with the heat sink thrust chamber assembly. Thrust was measured to help confirm the recorded values of chamber pressure. Good agreement was obtained between the test data derived nozzle C_f and the sea level C_f calculated on the basis of the theoretical value and the nozzle losses. Swirl meters which had been calibrated with O_2 and H_2 gas were employed for flow measurement.

The test hardware configurations which provided performance consistent with the program requirements are shown in the figure. The data was obtained with the highest nominal value of fuel injection Mach Number, 0.56. The oxidizer configurations providing % C* above 97% have pressure drops which exceed the program work statement goal of 90 psi. The 6.44/.56/5% unit is the primary candidate for altitude testing because of its lower heat rejection as discussed on a later page.

% THEORETICAL c* VERSUS MIXTURE RATIO

$L^* = 32$, $P_c = 300$ PSIA

α	6.44	6.51	10.73	10.73
M_N	0.56	0.56	0.56	0.56
%CF	5	5	5	10
SYMBOL	⊙	△	⊡	◊



The variation of the % of theoretical shifting C* at a mixture ratio of 4.0, Pc = 300 psia, 5% center flow, 32 L*, versus α shows increasing % C* with higher values of α. The highest performance was obtained with a fuel injection Mach Number of 0.56. The same trend % C* vs α was noted with the 1000 lb. thrust unit.

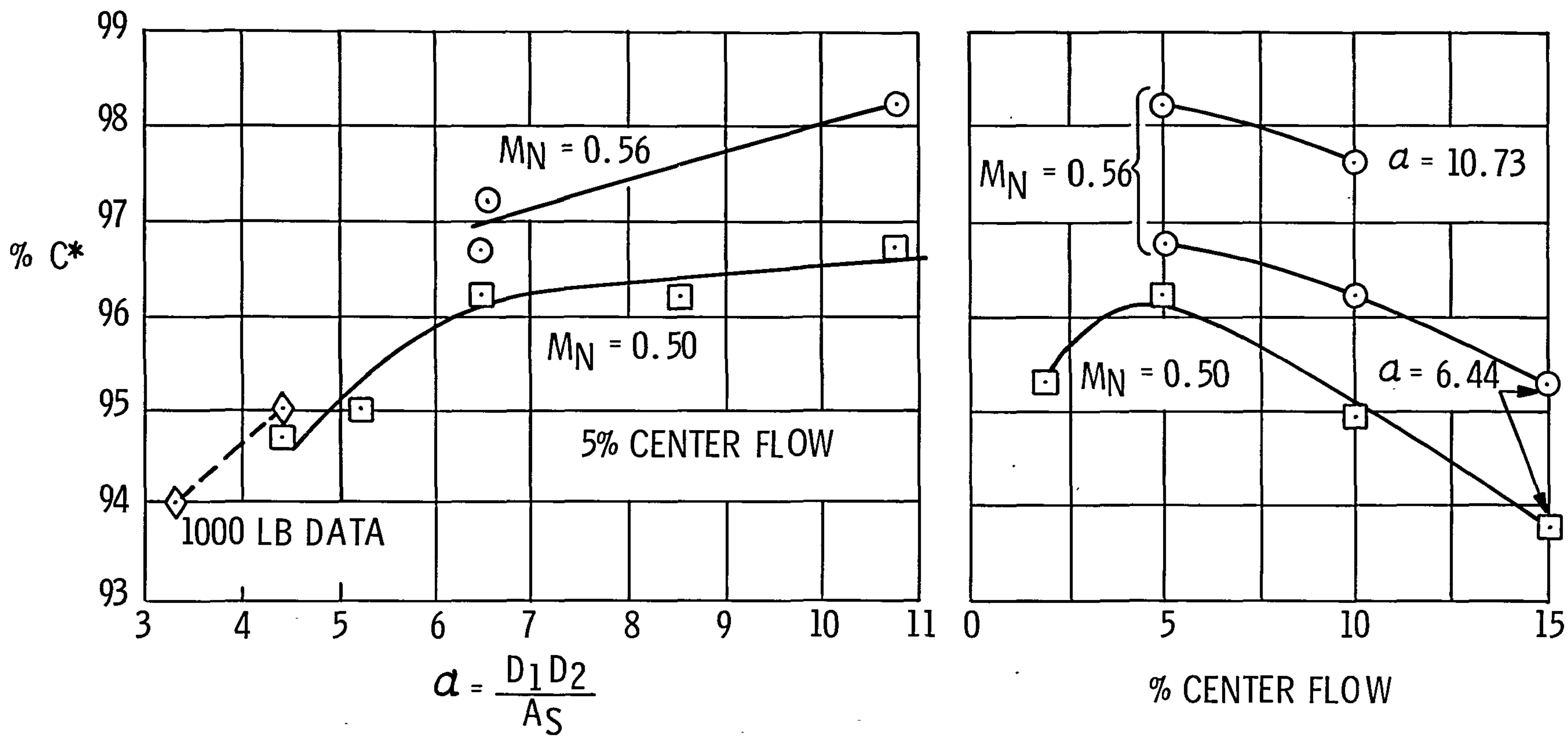
The oxidizer swirl cup center flow prevents hot gas circulation into the lower pressure region of the oxidizer vortex cup. The % C* shows a consistent fall off with increasing center flow above 5%. One set of data for operation with an α of 6.44, MN = 0.50 with 1.5% centerflow shows a decrease of % C*. The 5% optimum value of centerflow with the 1500 lb. thrust unit is consistent with the data obtained with the 1000 lb. thrust assembly.

% C* VERSUS TEST VARIABLES

O/F = 4.0

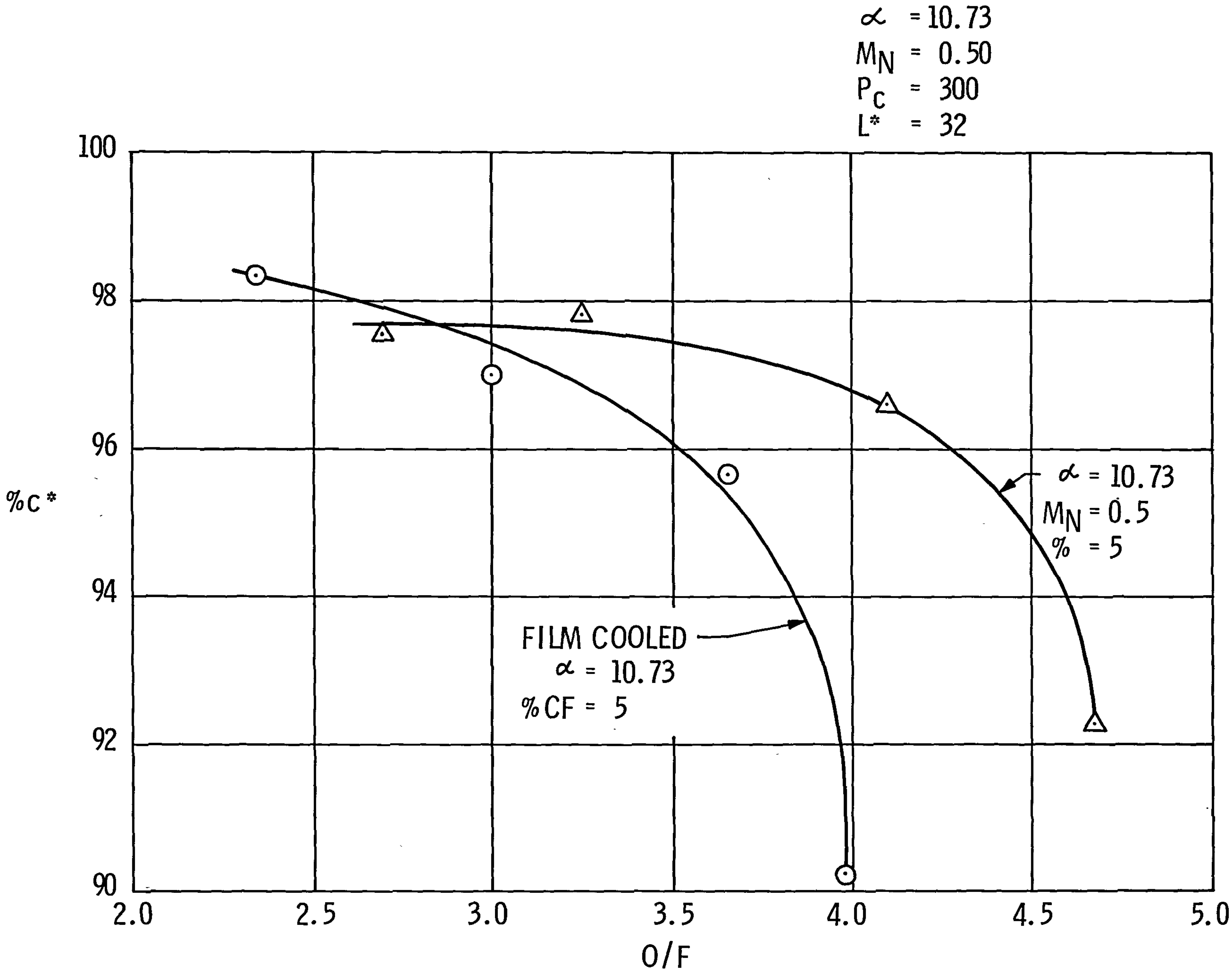
P_C = 300

L* = 32



The increase of % C* with increasing α for a given fuel injection Mach Number is limited by the interaction of the oxidizer sheet combustion zone and the H_2 near the point of fuel injection. The H_2 flow is partially blocked and a rapid reduction of % C* occurs as shown for tests of a 10.73 α configuration at lower fuel injection Mach Numbers.

% c* VERSUS O/F



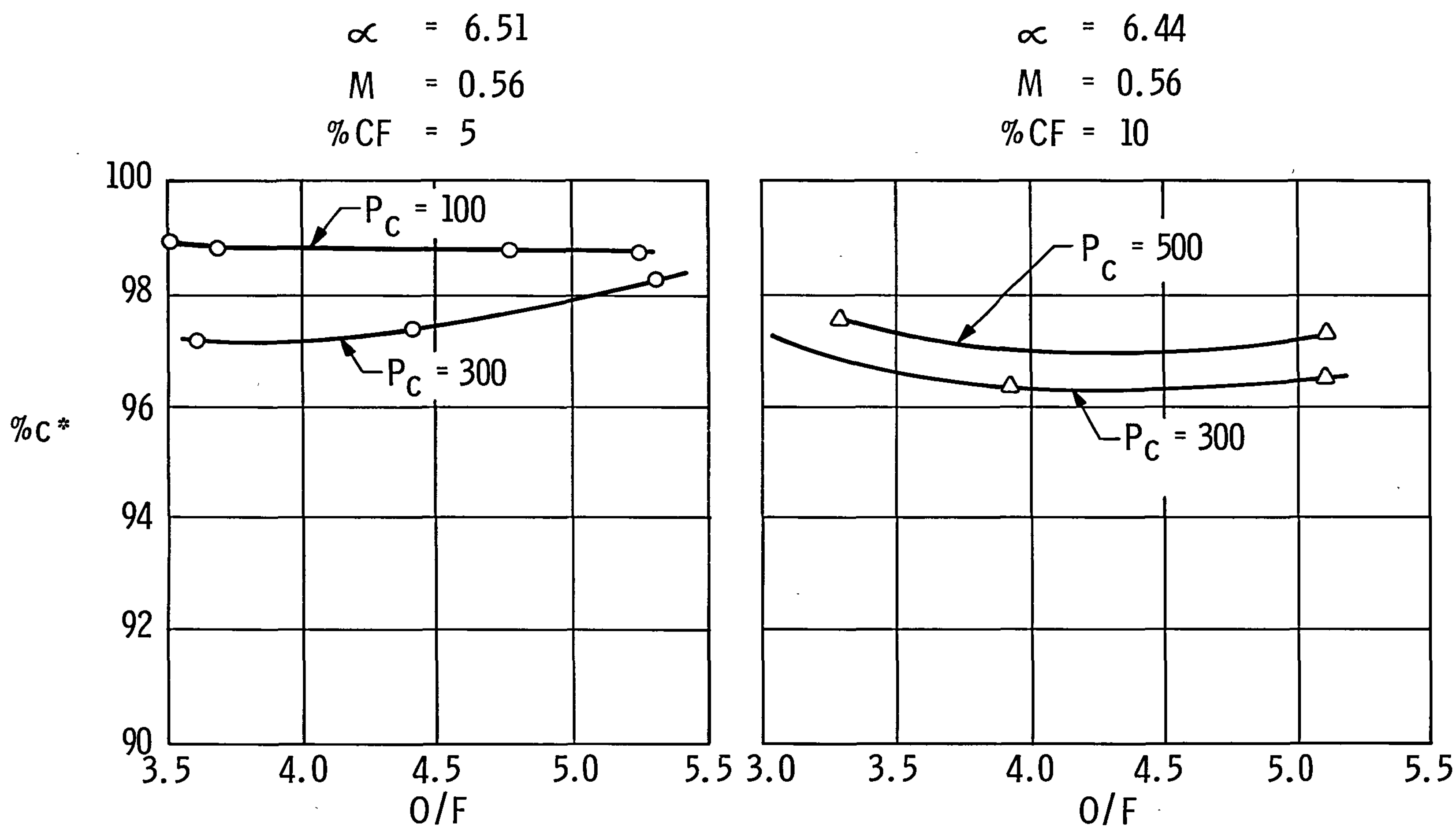
α = 10.73
M_N = 0.50
P_c = 300
L* = 32
%c*
100
98
96
94
92
90
2.0
2.5
3.0
3.5
4.0
4.5
5.0
O/F
FILM COOLED
α = 10.73
%CF = 5
α = 10.73
M_N = 0.5
% = 5

Tests conducted at low Pc showed an increase of % C* for a configuration of 6.51α , .56 Mach Number and 5% centerflow. The data for the same configuration at a Pc of 300 psia is included for comparison. The performance improvement at low Pc and the reduced fuel and oxidizer ΔP's at the reduced flow suggests further tests to explore the lower injection pressure drop operation. Other data comparisons suggest a high performance regime at low oxidizer ΔP.

Operation at 500 psia shows an increase of about 0.5% C* compared with operation at 300 psia with a 6.44α , .56 Mach Number, 10% centerflow configuration.

The low and high Pc tests were conducted with two injector configurations because of the relatively high ΔP of the α = 6.51 configuration at 500 psia.

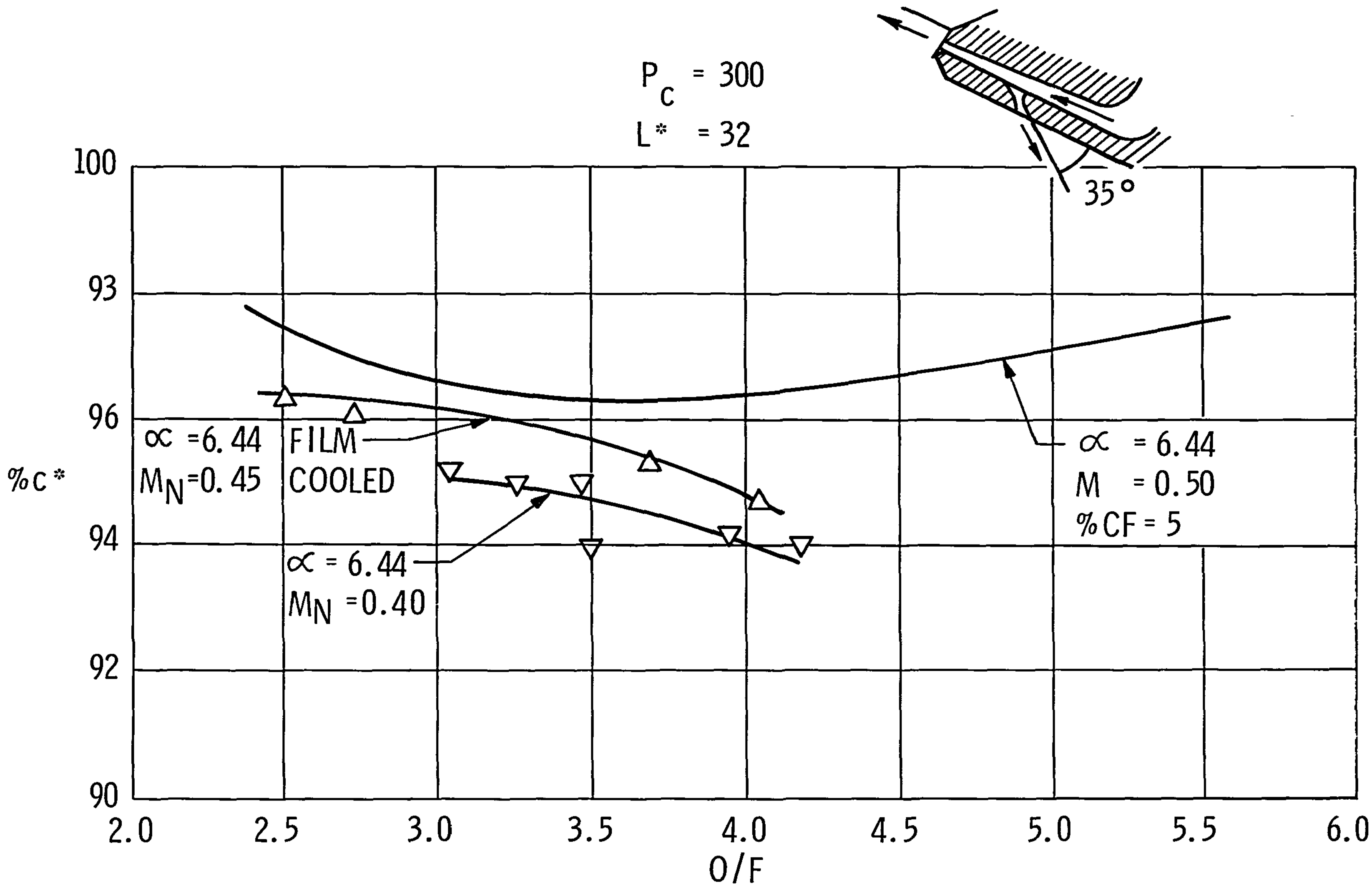
LOW AND HIGH P_c PERFORMANCE



The data for operation with film cooling is compared with the same configuration of α and % centerflow with a fuel injection Mach Number of 0.50. The film coolant slots were added by EDM approximately 0.20 inches downstream of the fuel injection orifices as shown in the sketch. The slots were calculated to provide a fuel injection velocity of 0.50 with 10% of the fuel passing through the film coolant slots. The slot crossectional shape and H_2 flow pattern resulted in a coolant flow greater than 10% with a resultant reduction of the H_2 injection velocity. The reduced H_2 injection velocity accentuated the performance fall off as noted earlier with the 10.73 α set up. The film coolant plot at an α of 6.44 follows the trend of data for tests made at a fuel injection of MN = 0.40 with α = 6.44 and 10% CF.

The film coolant effect on combustion efficiency was masked by a greater reduction in fuel injection velocity than desired. The film cooling tests will be repeated during thrust chamber firings at altitude scheduled to begin the end of April. The coolant slot design will reflect the experience with the sea level test units.

% c* VERSUS MIXTURE RATIO WITH FILM COOLING



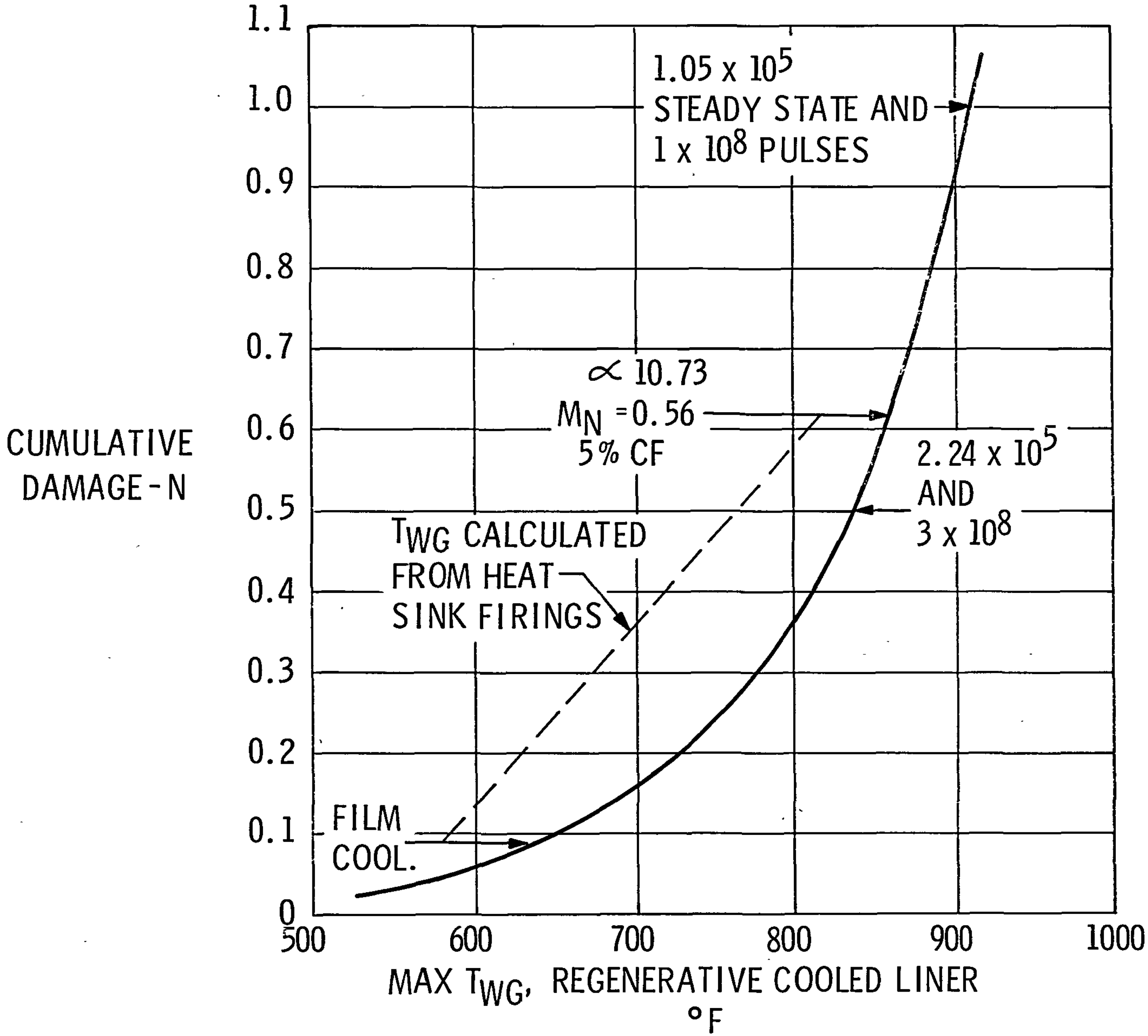
The plot of cumulative damage for a free standing zirconium copper liner (0.070 inches thick) versus the liner maximum gas side wall temperature was defined by calculation techniques described in some more detail on a later page. The cyclic capability of the liner for N=1 and 0.50 indicates that the design is limited by the 100,000 steady state firing capability rather than the 1 x 10^6 pulse requirement.

The range of T_{wg} calculated from the injector test firing data is indicated on the figure. The T_{wg} values were developed by calculation of the Q/A from the transient thermocouple data. That Q/A for the heat sink nozzle was employed for the calculation of T_{wg} for the .070 inch thick, regeneratively cooled liner.

The calculated T_{wg} values are for operation at a mixture ratio of 4.0. Data from a film cool test firing is included. The highest % C* configuration gave the highest heat rejection as noted on the figure.

The plot indicates that the higher performance injector configurations are within the limit of cumulative damage, N, less than 1.0. Also, the shape of the curve above a T_{wg} of 800°F suggests a reasonable margin for life cycle capability will require operation at or below N = 0.5. The plot also indicates a trade-off of injector performance and the performance loss with film cooling. Film cooling Isp loss will be evaluated during the scheduled thrust chamber cooling tests.

FREE STANDING AMZIRC NOZZLE LINER CUMULATIVE DAMAGE VERSUS T WALL GAS



The preceeding test data shows that fuel and oxidizer injector configurations have been defined to provide combustion efficiency % C* above contract goal with acceptable heat rejection characteristics. The injectors are available for the scheduled thrust chamber cooling tests at simulated altitude conditions.

A regenerative cooled nozzle section is being incorporated in the sea level test cell to permit evaluation of the injector configurations for longer duration firings to thermal equilibrium of the combustion chamber and nozzle. The firings will provide actual steady state wall temperatures. The sea level nozzle is fabricated with AMZIRC and the test program will confirm the altitude design release.

Lower pressure drop oxidizer injection schemes will be evaluated with 1500 lb thrust sea level test hardware. The continuing sea level testing is part of company sponsored reverse flow engine development work.

INJECTOR TEST STATUS

- HIGH PERFORMANCE CONFIGURATION DEFINED FOR TCA ALTITUDE TEST
- ADDITIONAL INJECTOR TESTING IN PROGRESS
  - OPERATION TO THERMAL EQUILIBRIUM AT SEA LEVEL WITH REGEN COOLED NOZZLE
  - LOWER PRESSURE DROP CONFIGURATION

1500 LBF COOLED THRUST CHAMBER ASSEMBLY
(WITH REGENERATIVE COOLED NOZZLE EXTENSION)

The baseline cooled chamber configuration for altitude testing consists of a reverse flow, fuel cooled spherical chamber, a vortex cup oxidizer injector, a regeneratively cooled nozzle section, (including fuel injector), and a "work horse" type regeneratively cooled nozzle extension. Dual flush mounted spark igniters are provided in the spherical chamber wall.

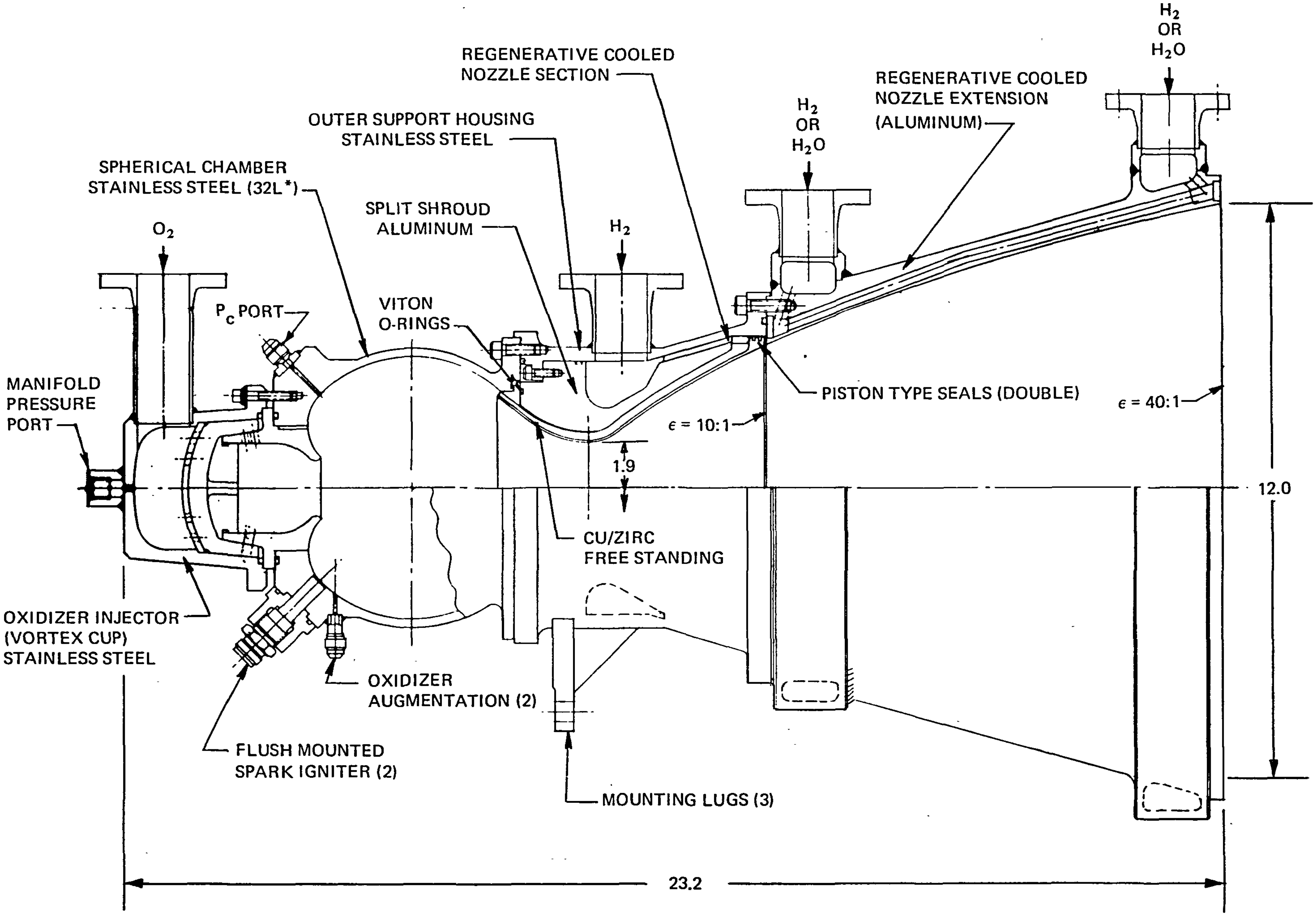
The nozzle section ends at an area ratio of 10:1 and the nozzle extension continues from an area ratio of 10:1 to 40:1. This nozzle extension can be regeneratively cooled by a separate supply of H_2O or gaseous H_2 and exhausted overboard. A fully regenerative H_2 cooled configuration can be evaluated by connecting a "jumper" line from the nozzle extension outlet to the nozzle section inlet.

The hardware combination will provide the capability of determining the performance and heat flux under the highest performance configuration (without film or dump cooling).

Interchangeable regenerative cooled nozzle sections incorporating film cooling and dump cooling will also be used, therefore allowing the evaluation of the effects of film and/or dump cooling.

Temperature data will be obtained on the spherical chamber wall, nozzle section liner wall and the regenerative cooled nozzle extension.

1500 LBF COOLED THRUST CHAMBER ASSEMBLY



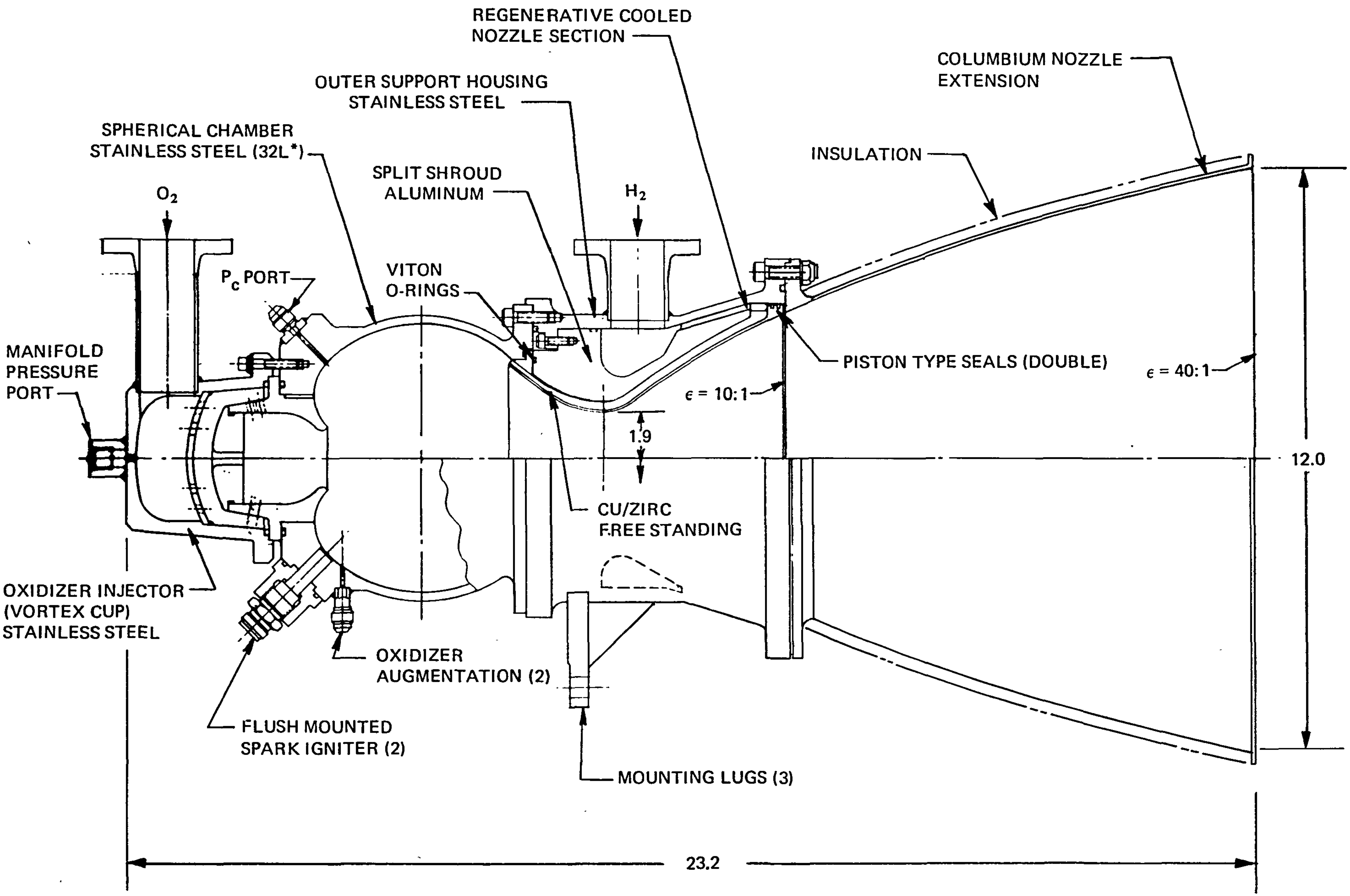
1500 LBS COOLED THRUST CHAMBER ASSEMBLY

(WITH UNCOOLED COLUMBIUM NOZZLE EXTENSION)

This cooled chamber configuration is identical to the baseline configuration except an uncooled columbium nozzle extension replaces the regeneratively cooled, aluminum nozzle extension. Columbium will be employed to give a high temperature margin for the tests. The nozzle extension will be used with the film cooled and/or dump cooled nozzle sections. Nozzle extension temperatures will be obtained by thermocouples welded to the exterior surface of extension.

The effect of film and/or dump cooling on performance and heat flux will be obtained on this more "flight type" configuration.

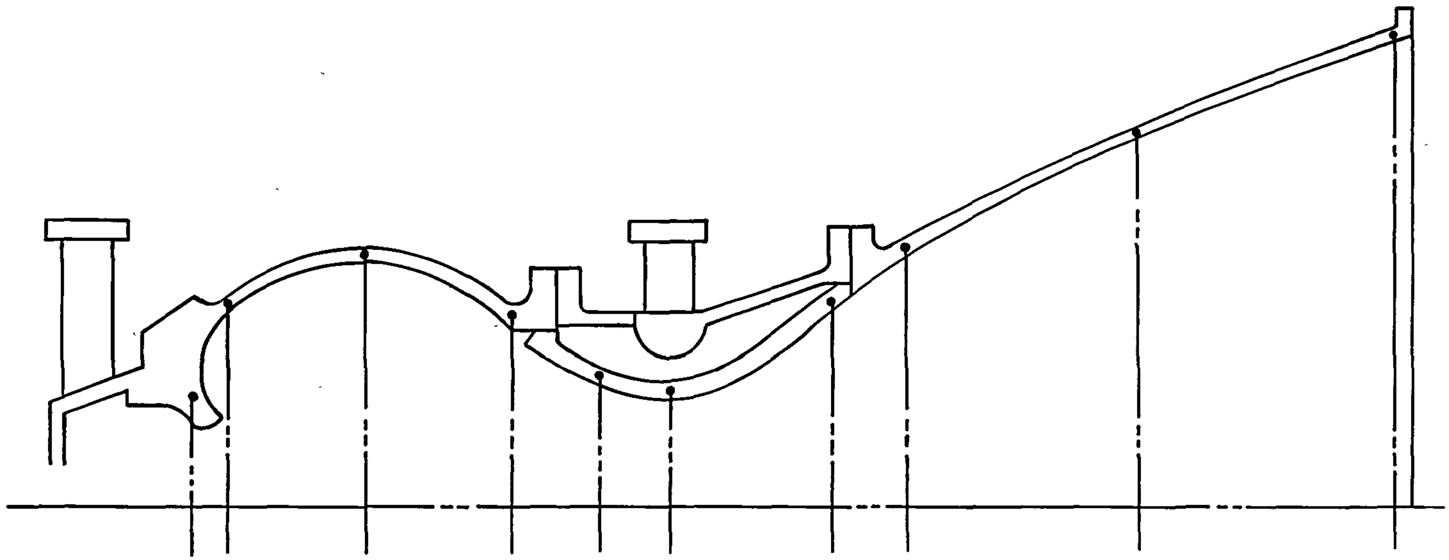
1500 LBF COOLED THRUST CHAMBER ASSEMBLY



The predicted operating temperatures for the entire thrust chamber assembly with a 10% dump cooled nozzle extension and employing 5% nozzle section film cooling are presented in the figure. The temperatures are consistent with a minimum engine pulse cycle life of 1×10^7 pulses and over 500,000 steady state firings. The temperature and life cycle estimates will be refined with the data from the engine altitude test firings.

<u>PREDICTED OPERATING CONDITIONS</u>

1500 LBF REVERSE FLOW COOLED CHAMBER ASSEMBLY



MAXIMUM TEMP °F   300 1200 1000 500 740 740 740 500 1750 2150 (10%)

MINIMUM CYCLE LIFE 10^7 10^7 10^7 10^7 10^7 10^7 10^7 10^7 10^7

(540°R PROPELLANT TEMPERATURE)

The estimated performance losses for the area ratio 40 thrust chamber assembly and an engine design with 5% film cooling and 10% effective dump cooling are approximately 4 to 6.5 seconds depending on the effectiveness of the effective dump cooling obtained with film cooling. For example, if 5% film cooling provides 5% effective dump cooling the actual dump cooling flow can be limited to 5% to achieve the desired maximum nozzle extension temperature. The % of theoretical C* with injectors tested to date with no film or dump cooling would provide an altitude I_{sp} of approximately 439 seconds based on calculated C_F losses with an optimized 85% bell nozzle. Therefore, the anticipated engine altitude performance is in the range of 432 to 435 seconds.

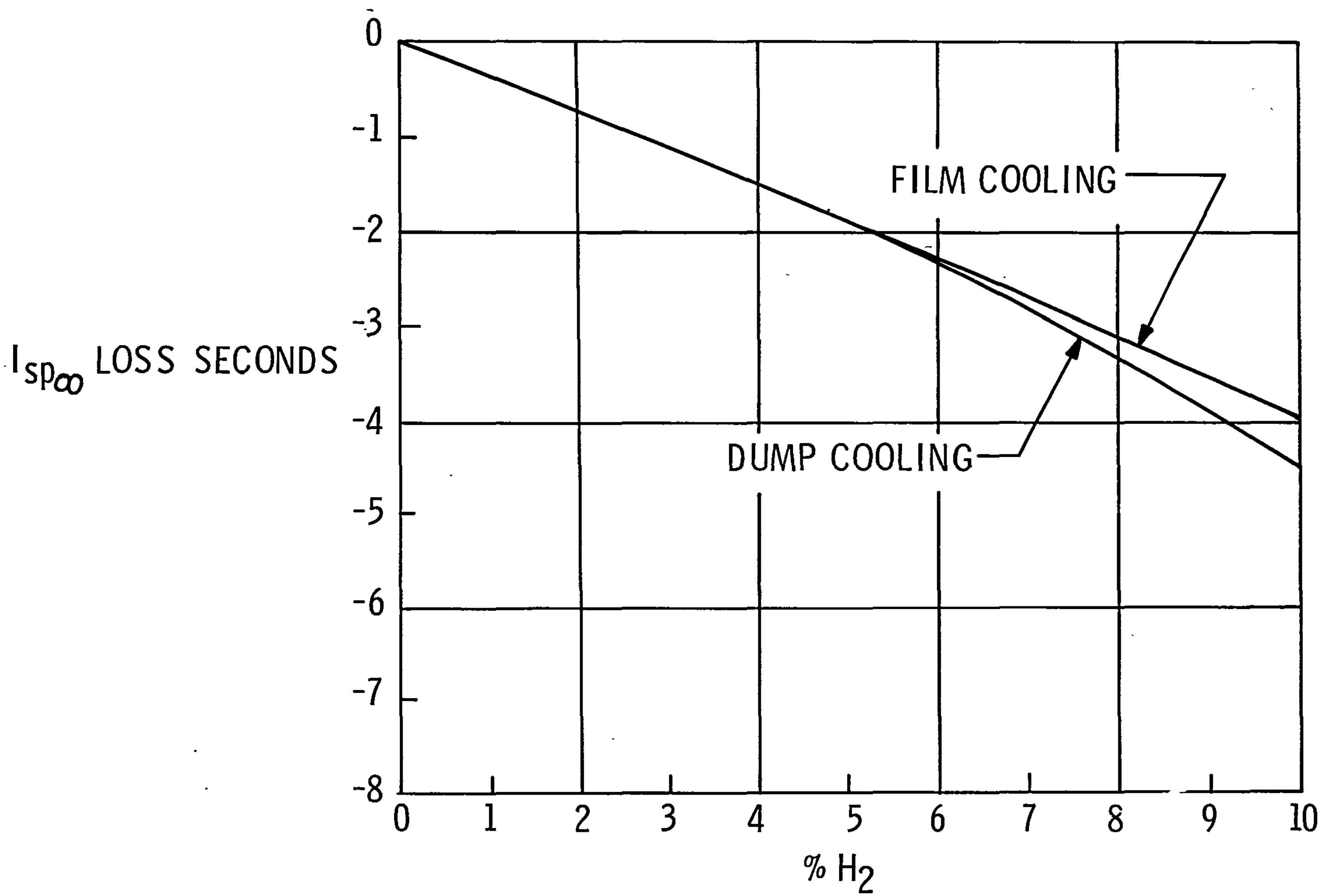
Calculations of the predicted performance loss with film cooling are continuing.

The nozzle section film coolant flow not only provides a large reduction in T_{wg} and an increase in cycle life but also a large calculated reduction in the regenerative coolant flom ΔP, approximately 50 psi.

ESTIMATED I_{sp_∞} LOSS FILM AND DUMP COOLING

ENGINE O/F = 4

INLET TEMPERATURE = 80 °F



IGNITION SYSTEM

The electrical ignition system has been used for over 150 injector evaluation tests. The exciter 79 millijoule storage and spark rate of 850 SPS @ 28 volts, was demonstrated to provide repeatable ignition. The ignition system employs oxidizer augmentation. The exciter and spark plug are Bendix Mode Nos. 10-81110-3 and 10-390150-1 respectively.

VALVE

Individual ball type propellant valves, employing a pneumatic actuator, are used for the engine testing. These valves have demonstrated opening response times of .014 seconds and closing response times of .014 seconds including the delay of the solenoid pilot valves. The pressure drop is below 10 psia at rated fuel and oxidizer flows. Tests have been conducted with oxidizer leads and with oxidizer or fuel overrides.

ADDITIONAL ENGINE HARDWARE

IGNITION SYSTEM

EXCITER - INDUCTIVE DISCHARGE

SPARK PLUG - SURFACE GAP

VALVE

PNEUMATICALLY ACTUATED BALL VALVE

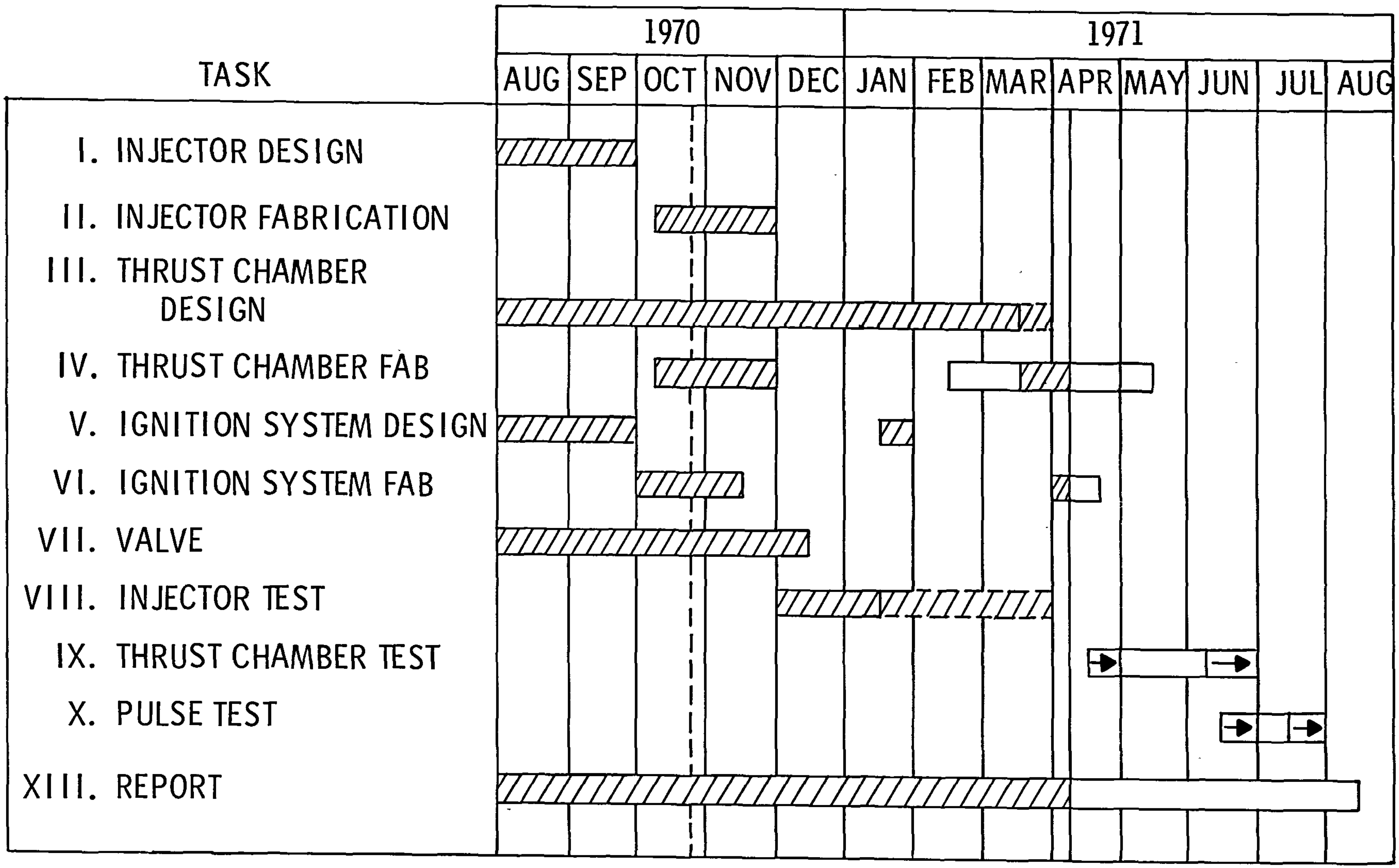
INDIVIDUALLY OR ELECTRICALLY COUPLED OPERATION

The effort accomplished against the contract tasks shows that the injector test effort has continued beyond its originally scheduled time span and some slippage of thrust chamber design and fabrication has occurred. The impact of the slippage against the remaining effort is small, currently estimated at 2 weeks.

The thrust chamber tests will be carried out under simulated altitude conditions. A baseline configuration engine incorporating a regenerative cooled nozzle section and a water cooled nozzle extension will be tested first to establish engine altitude I_{sp} without chamber film or extension dump cooling. The second series of tests will establish the performance, ΔP and cooling changes with a nominal 5% film cooling injected just aft of the fuel injection station of a regeneratively cooled nozzle. The third setup will include a regenerative cooled nozzle providing nozzle extension dump cooling of 7.5% to 10% of the H_2 flow. A water cooled nozzle extension will again be used for the dump cooling tests to establish the dump cooling I_{sp} loss and the change of heat flux to the extension. One of the preceeding three configurations will be operated at low and high propellant feed temperature. The testing will conclude with a series of firings to demonstrate durability of the throat sections and to establish the temperature profile of a dump/radiation cooled nozzle extension.

The series of tests will provide the specific impulse and other operating characteristics of the reverse flow engine and the operating characteristics with nozzle film and nozzle extension dump cooling. The hardware tested and the data obtained can be employed to establish the optimized designs of flight weight reverse flow engines capable of meeting both the contract Isp goal and life cycle requirements. The program will conclude with pulse testing at low, nominal, and high feed temperature and chamber pressure.

NAS-3-14353 PROGRAM SCHEDULE



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"SPACE SHUTTLE ACPS SHUTOFF VALVE"

H. WICHMANN

THE MARQUARDT COMPANY

TECHNICAL MANAGER

R. GREY

LEWIS RESEARCH CENTER

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CONTRACT FOR SPACE SHUTTLE AUXILIARY PROPELLANT VALVES

NAS 3-14349

- **OBJECTIVE:**
  - TO DEVELOP AND DEMONSTRATE DESIGN CRITERIA FOR THE FLIGHT TYPE GASEOUS HYDROGEN-GASEOUS OXYGEN PROPELLANT SHUT-OFF VALVES WHICH WILL BE UTILIZED ON THE THRUSTERS IN THE SPACE SHUTTLE AUXILIARY PROPULSION SYSTEM

NASA PROJECT MANAGER ----------- R. GREY

THE
MARQUARDT
CORPORATION

V8345-26B

The Program consists of five Tasks. Tasks I, II, and IV encompass analytical and design efforts, all fabrication and test efforts are included as Task III, and Task V has been designated as the reporting task.

The Task I effort consists of a trade-off study which analyzes various types of sealing closures, actuators and supporting parts to determine what types of components are most suitable for the APS Valve requirements. In recognition of the severe sealing requirements, this study is being supported experimentally by a sealing closure screening program.

During Task IIIA, eight different types of sealing closures are being fabricated and test evaluated by means of a rapid screening tester to determine their capability of meeting the 100 scc/hr maximum of Helium leakage requirement.

During Task II, preliminary design layouts of those valve concepts which have been determined most suitable for this application during Task I are prepared for both low pressure and high pressure propellant systems.

Task IIIB has been established to permit design, fabrication and test evaluation of boilerplate prototype valves of the valve concepts prepared in Task II.

During Task IV, final valve design layouts based on all previously generated data are prepared.

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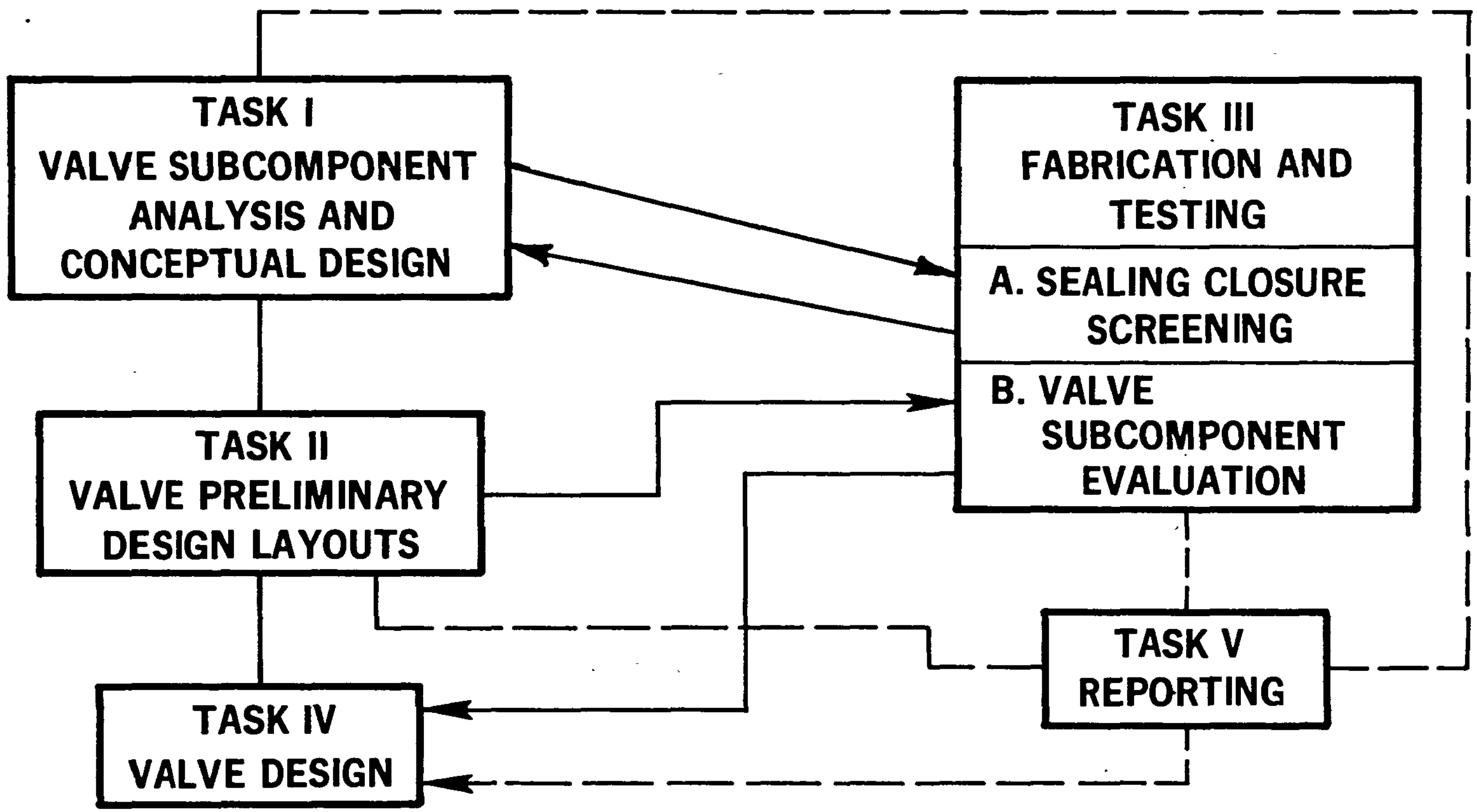
APS PROPELLANT VALVE TECHNOLOGY PROGRAM PLAN

FLOW DIAGRAM



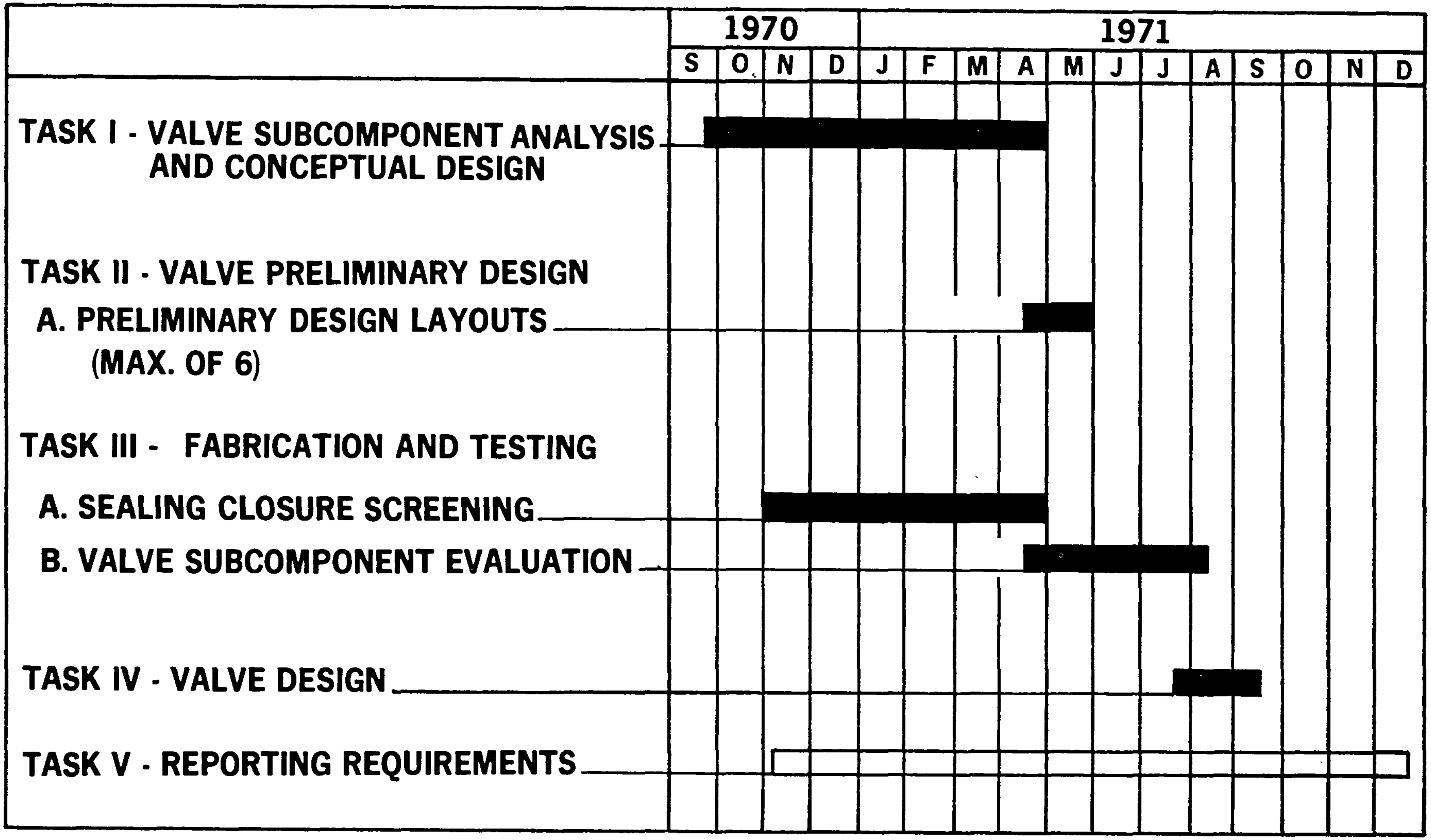
THE MARQUARDT COMPANY

UNCLASSIFIED

V8345-32

Efforts are presently in progress in support of Tasks I and IIIA. Both of these Tasks are expected to be completed at the end of April 1971.

PROGRAM SCHEDULE



THE MARQUARDT COMPANY

V8345-22A

The analytical leakage model was discussed in detail at the Space Shuttle Auxiliary Propulsion System Conference at NASA-Lewis on October 29, 1970. In summary, the model shows that valve leakage is minimized by minimizing the height of the leakage path which is a function of surface finish, surface load and material properties and is also inversely proportional to the length of the leakage path. Seal wear is minimized by minimizing impact loads and scrubbing when sealing closure mating takes place and by choosing materials which feature a low energy of adhesion, low wear coefficient and high hardness.

- **LEAKAGE** LAMINAR + MOLECULAR FLOW

$$Q = f(A, H^3, P^2, S_S, \bar{\lambda}/H)$$

- **SURFACE ROUGHNESS**

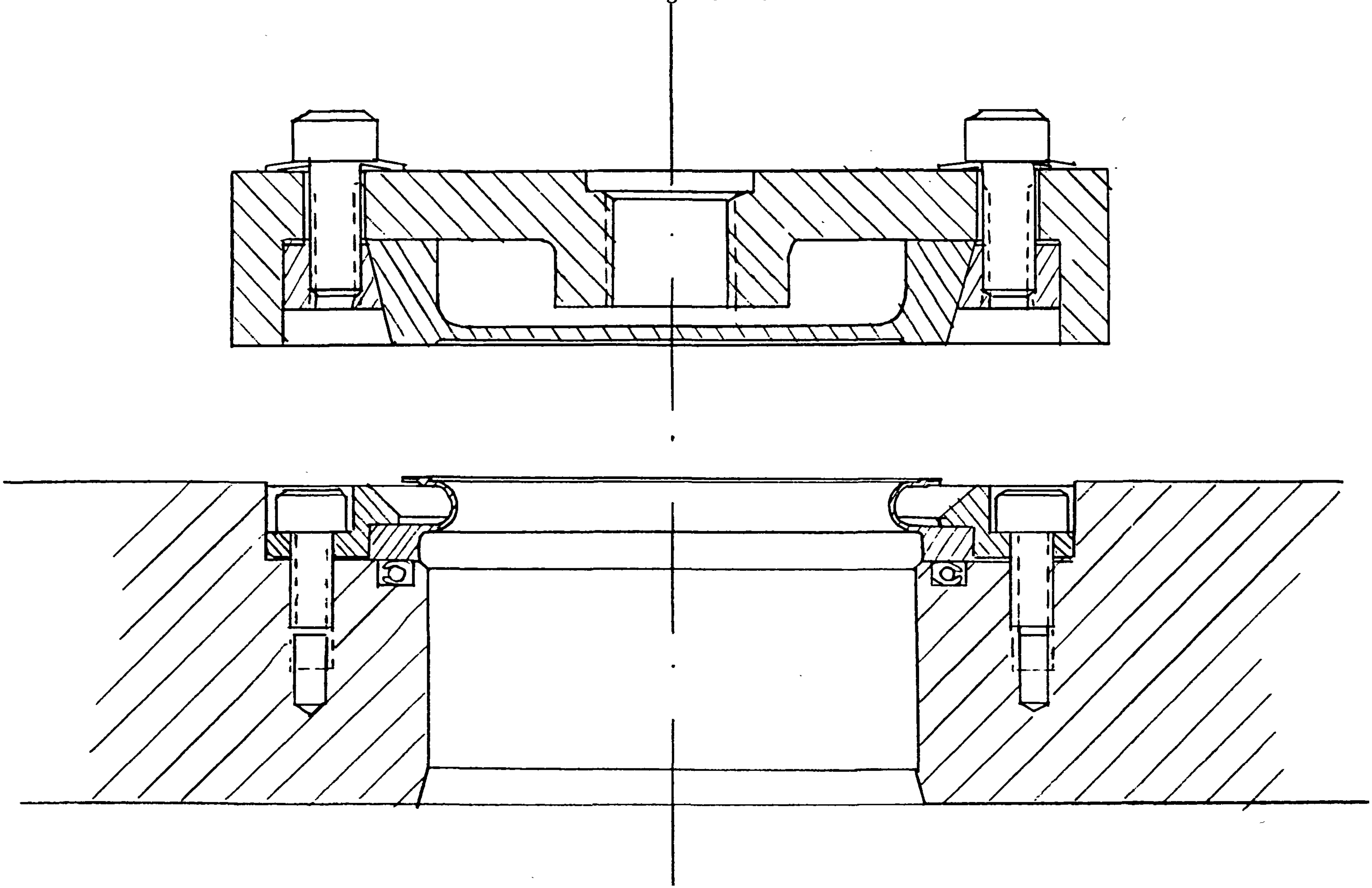
$$H = H^\circ + f(K_{AD}, N, S_i, X, P_A^{-1}, G_{AB})$$

509

This sealing closure consists of a well-lapped, flat tungsten carbide poppet mating with a teflon coated lip seat. This lip seat is allowed to deflect until the bumper surrounding the poppet stops the travel of the poppet assembly. Impact loads at the seal are minimized in this manner. The deflection of the lip assures alignment between the sealing surfaces.

FLAT TEFLON COATED LIP SEAT

Drawing No. L4683

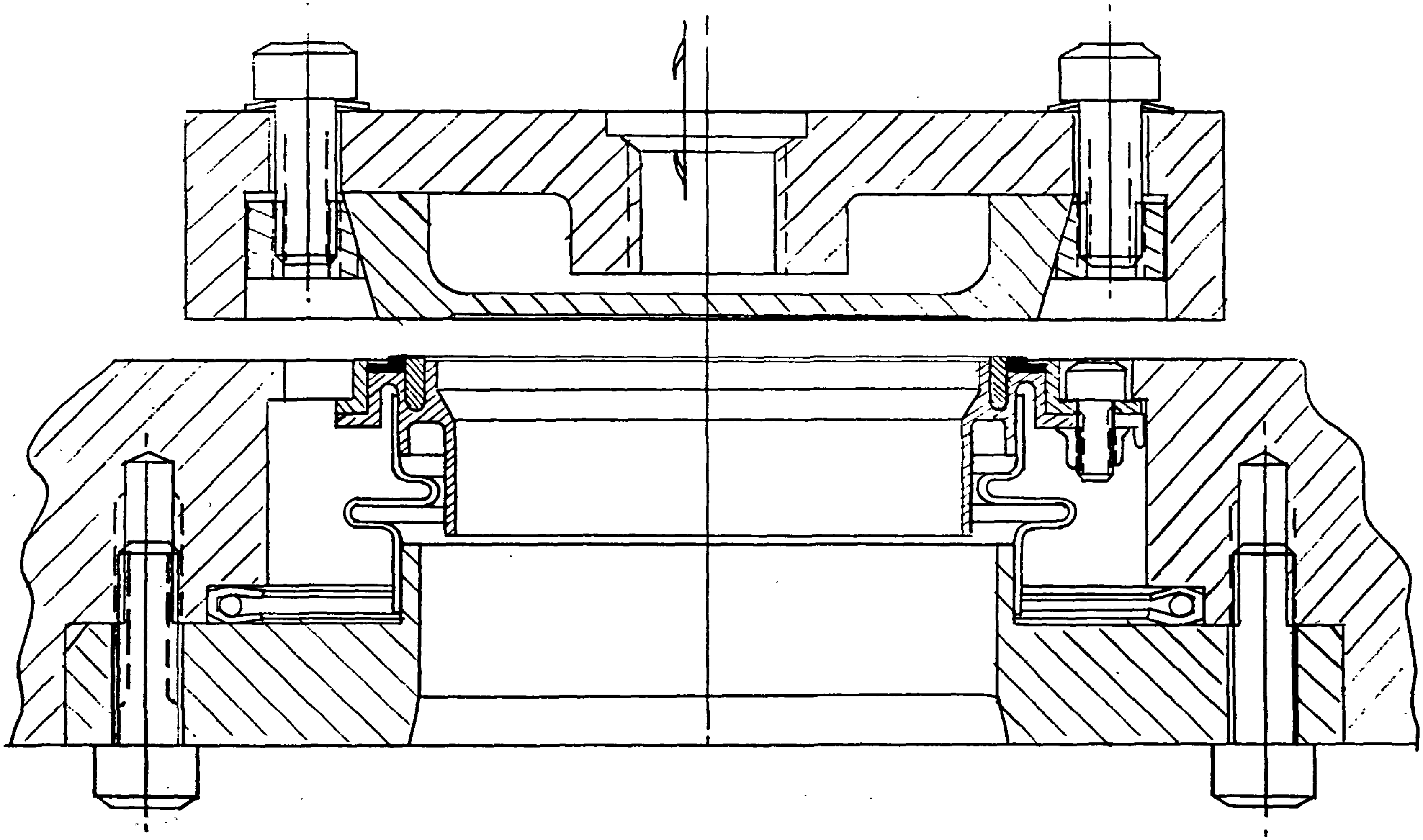


This sealing closure consists of a well-lapped, flat tungsten carbide poppet mating with a flat, combination polyimide and teflon seat. The teflon provides the primary seal and the purpose of the polyimide is to prevent cold flow of the teflon, particularly at elevated temperatures. The teflon is retained by means of a bolted retainer ring.

A bellows is utilized to establish sealing closure interface loads and to assure alignment of the seat to the poppet. The bellows also serves to minimize seal impact loads since the bumper around the poppet limits the travel to the poppet assembly. The effective diameter of the bellows is equal to that of the sealing closure for pressure balancing.

FLAT TEFLON SEAT BELLOWS LOADED

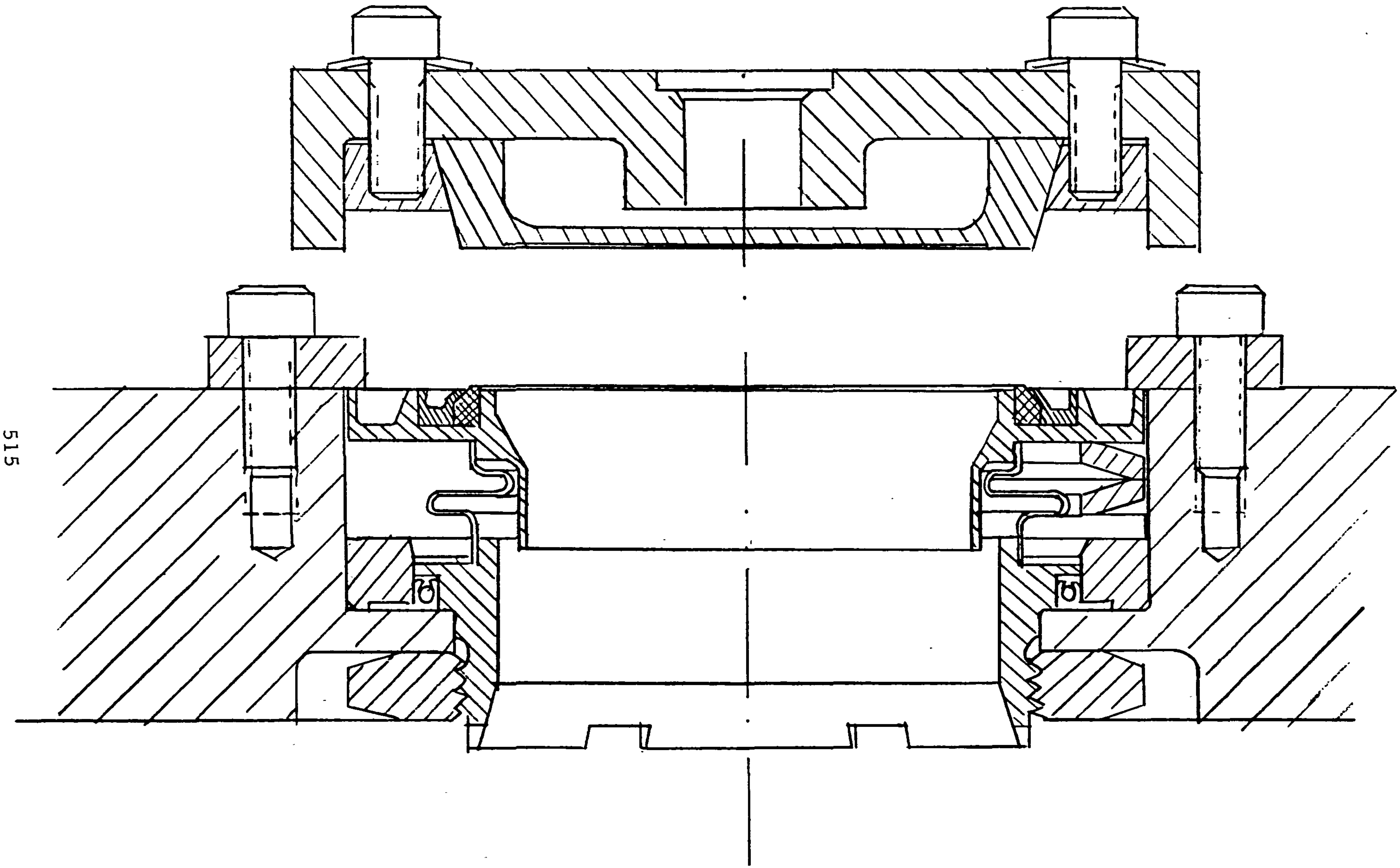
Drawing No. L4680



The P/N L4685 sealing closure is similar to the P/N L4680 sealing closure except that sealing is accomplished solely by the polyimide. This sealing closure features two Belleville springs in parallel with the bellows to permit higher sealing closure interface loads. By installing the proper size spacers seat loads up to 1500 lbs. can be preset during the sealing closure evaluation test program.

FLAT POLYIMIDE SEAT, BELLOWS FORCE LOADED

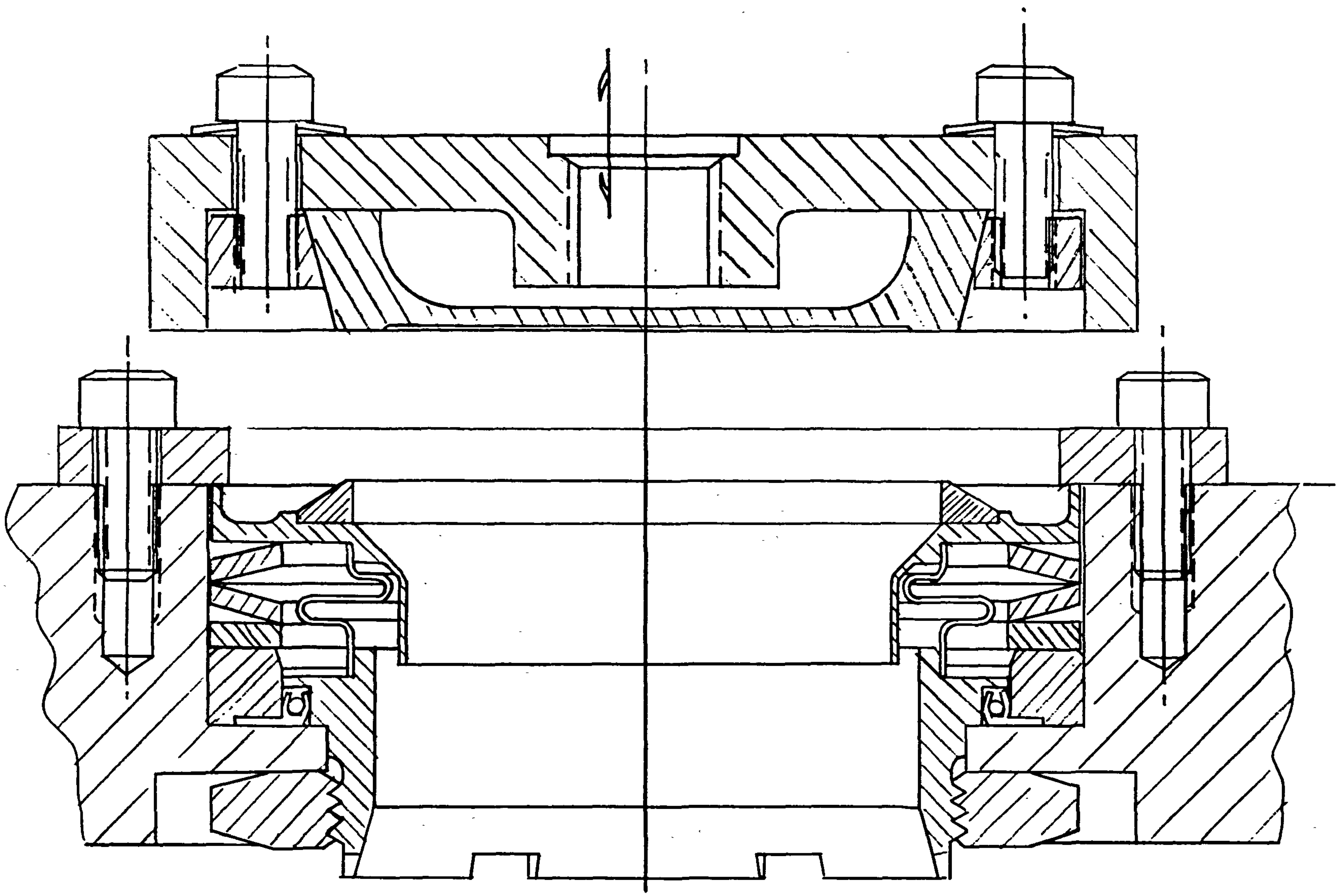
Drawing No. L4685



This sealing closure features a flat interface which utilizes well-lapped tungsten carbide in both the poppet and the seat. Again, the seat is bellows supported and also includes a Belleville spring to assure self-alignment between the sealing surfaces and to permit testing at a wide range of static sealing loads. Joining of the tungsten carbide seating to the stainless steel carrier is accomplished by brazing. Seat impact loads are limited to those required to accelerate the moving mass of the seat to the poppet closure velocity by the use of the bumper around the poppet which absorbs the kinetic energy of the poppet assembly during closure.

FLAT METAL SEAT, BELLOWS FORCE LOADED

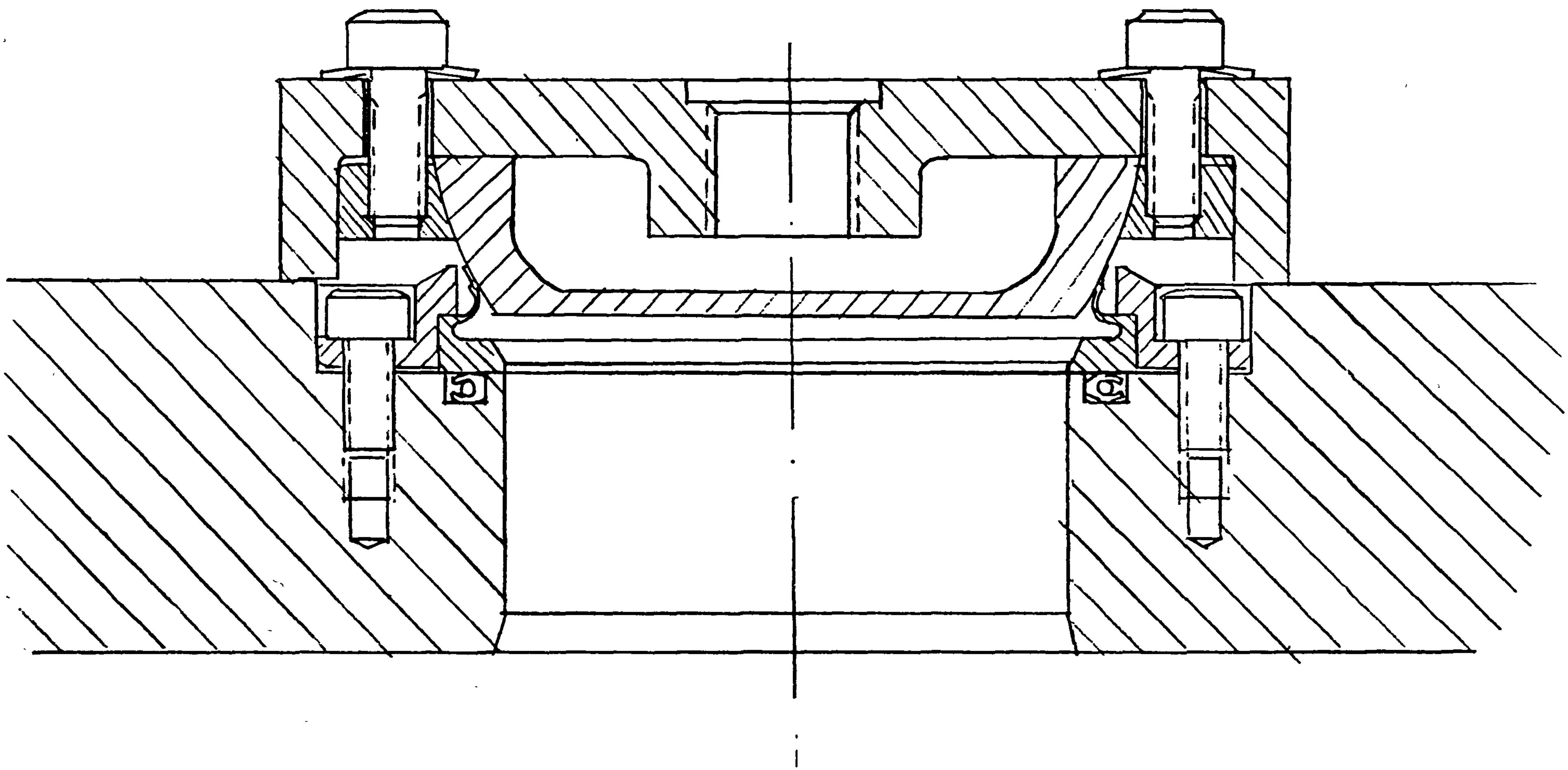
Drawing No. L4682



This sealing closure features a spherical, well-lapped tungsten carbide poppet mating with the conical, teflon coated lip seal. The mating surfaces are designed so as to always assure initial contact at the leading edge of the lip seal. The spring steel of the lip seal allows deflection of the seal during closure such that the impact forces resulting from suddenly stopping the poppet assembly during closure are nearly completely absorbed by the bumper around the poppet. Impact forces at the lip seal are very little greater than the static seal forces. Machining tolerances are such as to minimize lack of concentricity between the poppet and the seat.

SPHERICAL TEFLON COATED LIP SEAT

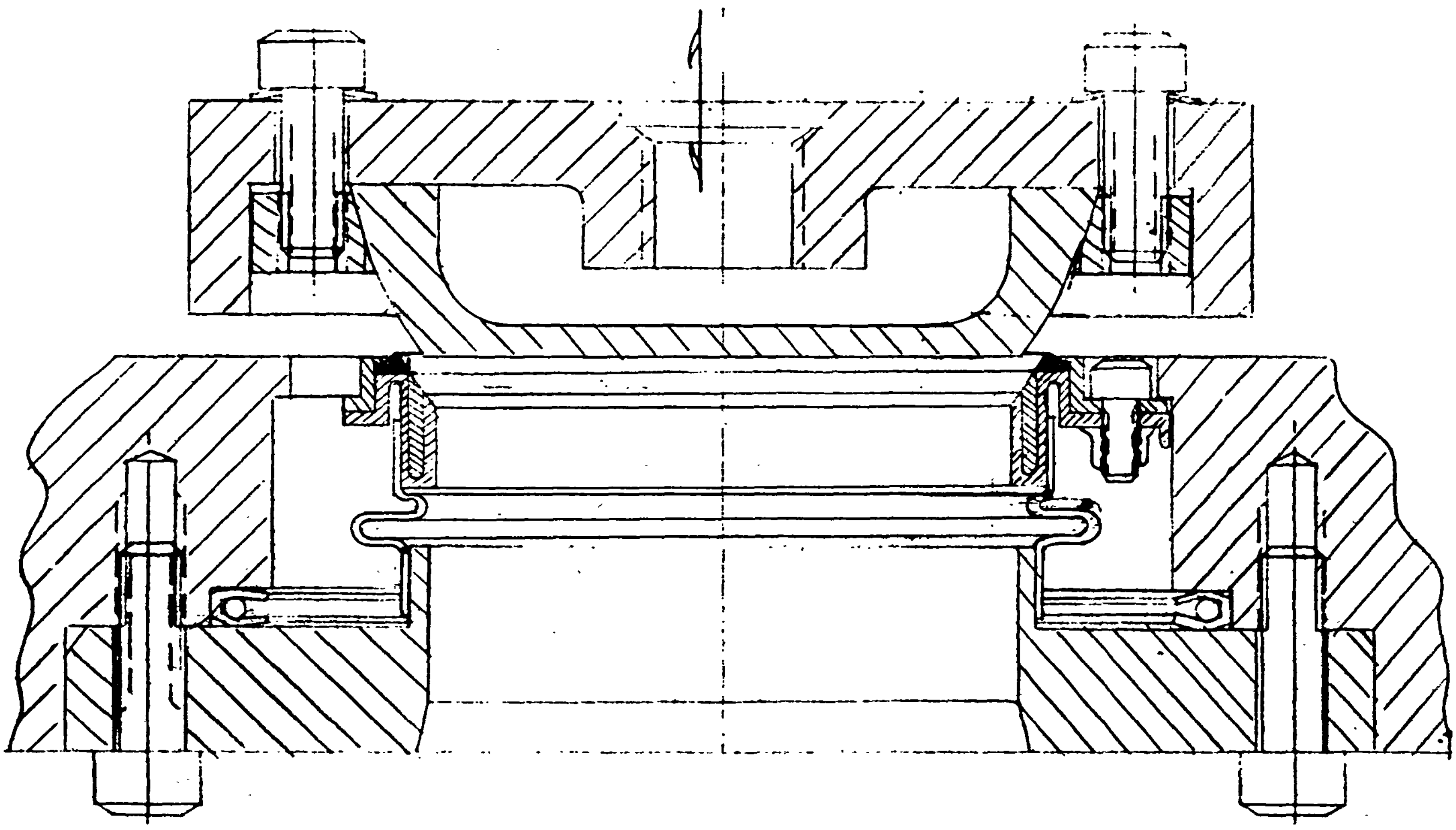
Drawing No. L4684



The P/N L4681 Sealing Closure consists of a well-lapped, spherical tungsten carbide poppet and a spherical, combination polyimide and teflon seat. The poppet diameter is very slightly larger than the seat diameter to assure sealing at the leading teflon edge. The polyimide serves to prevent cold flow of the teflon at elevated temperature and to minimize impact loads to the teflon during closure. Impact loads are further reduced to the seat as a result of the bellows and bumper construction. The bellows also serves to preset the static sealing load and to align the seat with the poppet.

SPHERICAL TEFLON SEAT BELLOWS LOADED

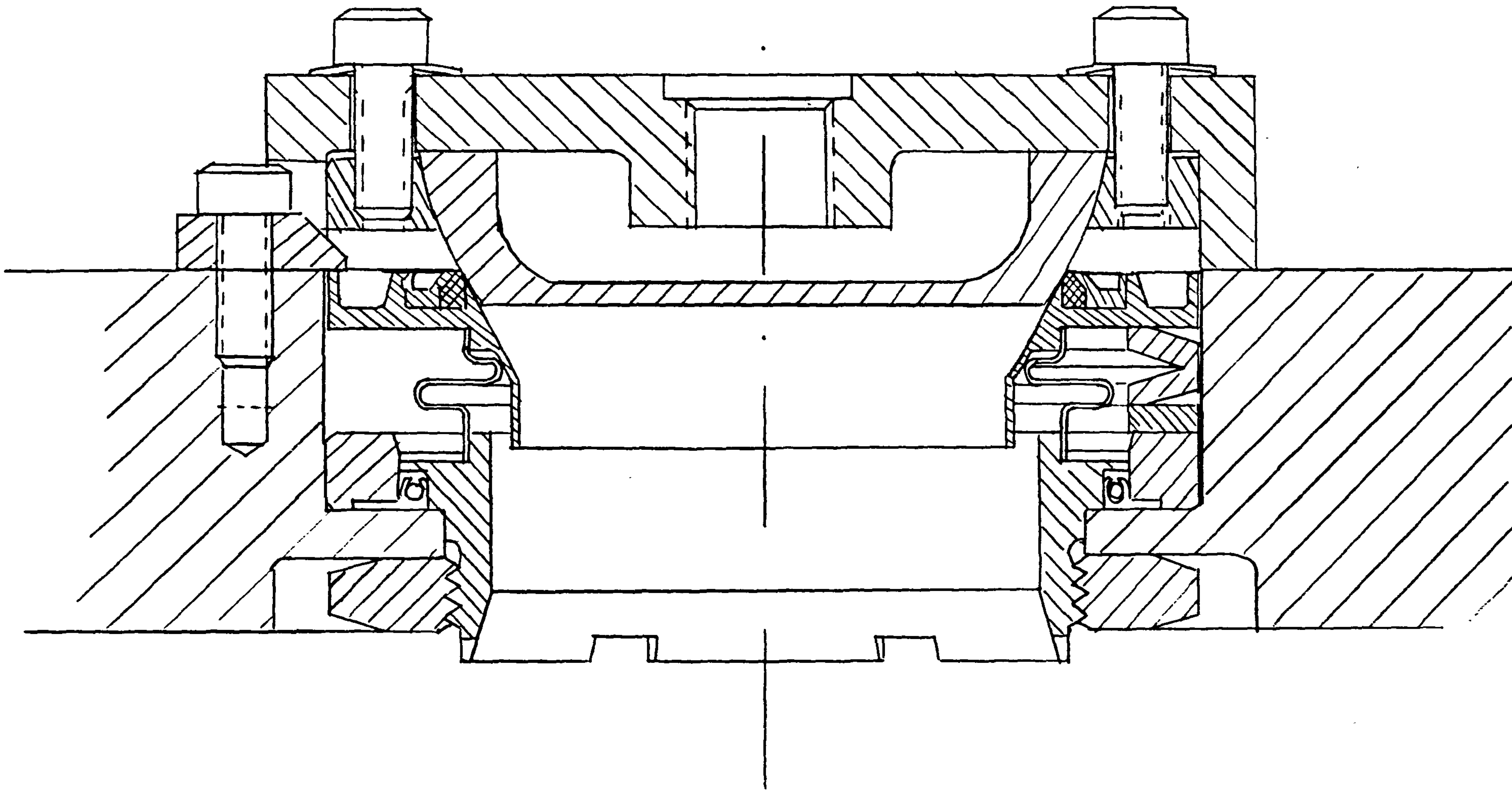
Drawing No. L4681



The spherical polyimide seat featured in sealing closure P/N L4686 is very similar to the polyimide/teflon combination seat except that sealing is now accomplished solely by the polyimide. The static sealing load of this sealing interface may be increased by the use of two Belleville springs in parallel with the bellows.

SPHERICAL POLYIMIDE SEAT, BELLOWS FORCE LOADED

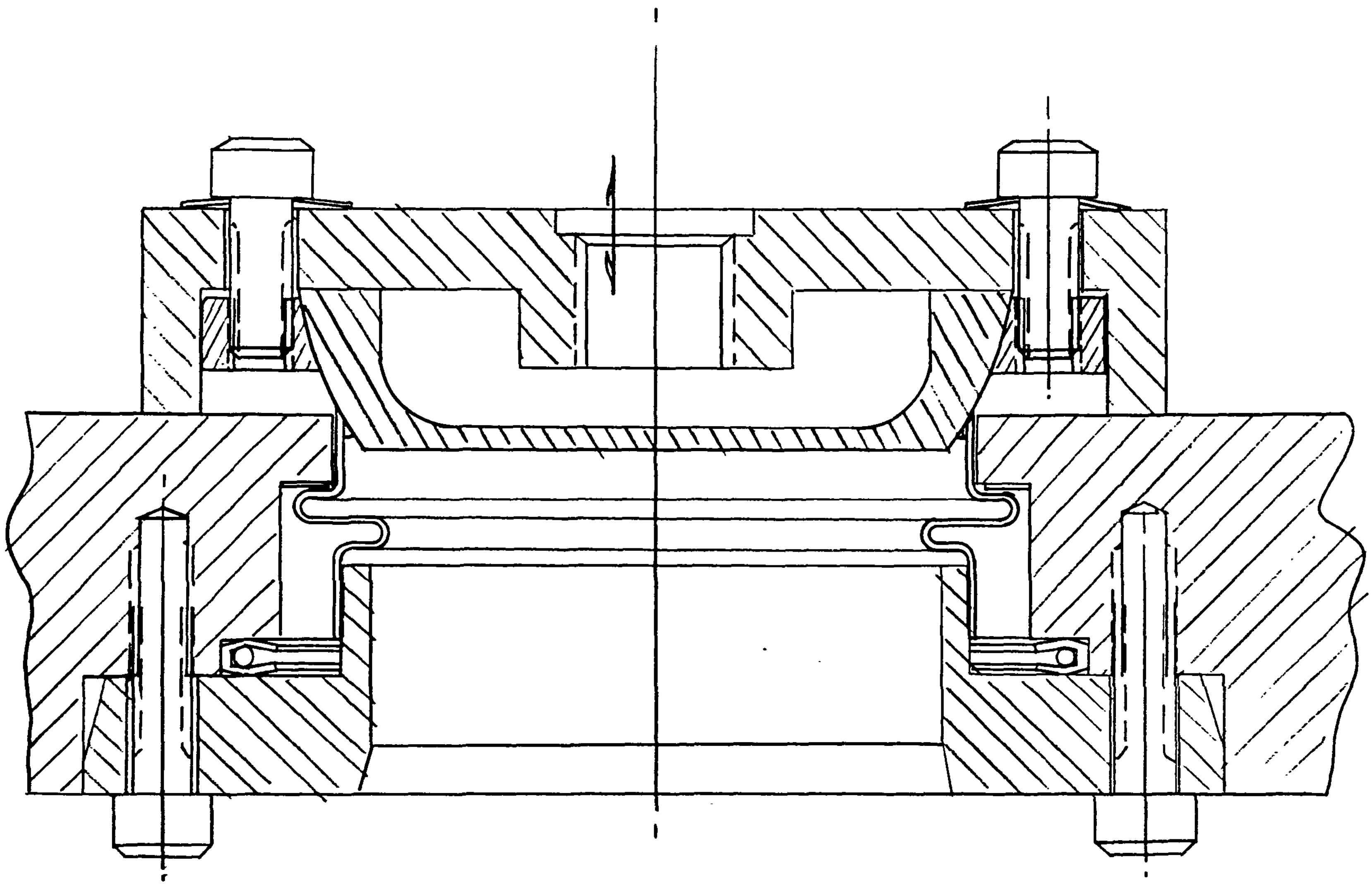
Drawing No. L4686



This sealing closure features a well-lapped, spherical tungsten carbide poppet mating with a spherical hard copper seat. The poppet diameter is very slightly larger than the seat diameter to assure sealing at the leading edge of the copper. The copper is brazed directly to the bellows. Impact energy absorption and alignment is similar to that of the other sealing closures.

SPHERICAL COPPER LIP SEAT, BELLOWS LOADED

Drawing No. L4678



The rapid screening tester developed for evaluation of the sealing closures over the required environmental ranges has several unique features. These are:

Controlled poppet guidance using radially stiff, metallic flexures.

Isolation of actuator and yoke mass from the poppet/flexure assembly by means of a spring joint.

Pressure balancing of the sealing closure.

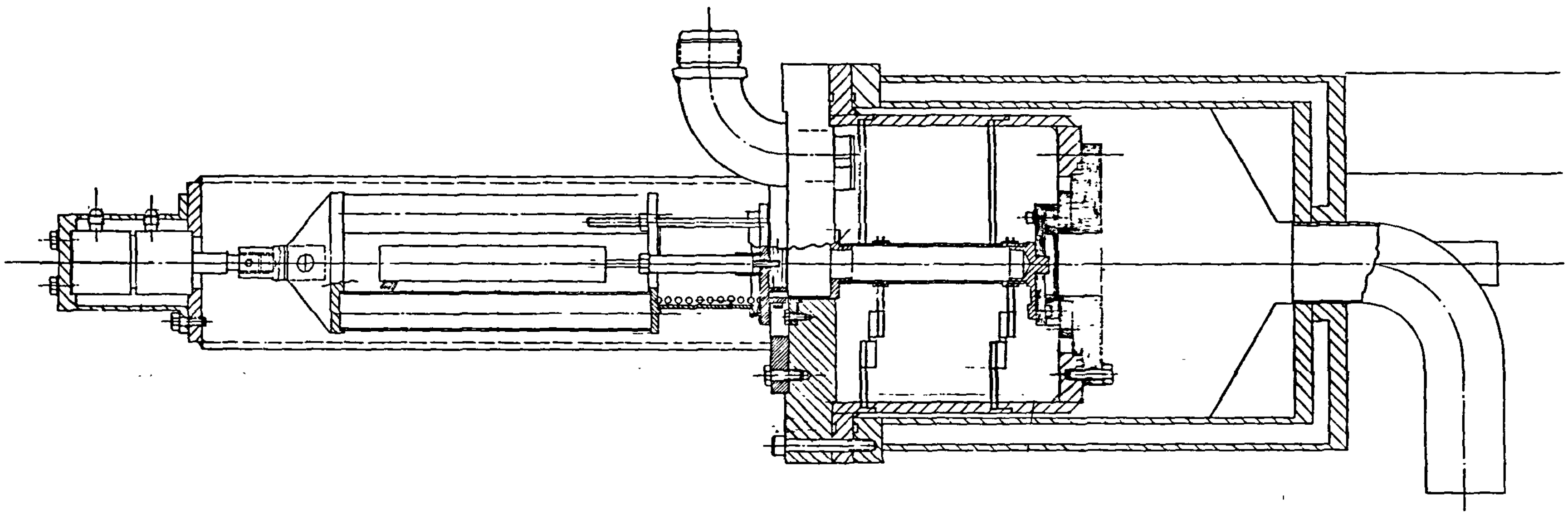
Controlled static and dynamic actuation forces through utilization of a hydraulic actuation system.

Actuator thermal isolation from the temperature conditioned sealing closure.

Variable reluctance position transducer.

Easily removable sealing closure.

RAPID SCREENING TEST FIXTURE - L4688



The picture of the sealing closure evaluation setup shows the following components:

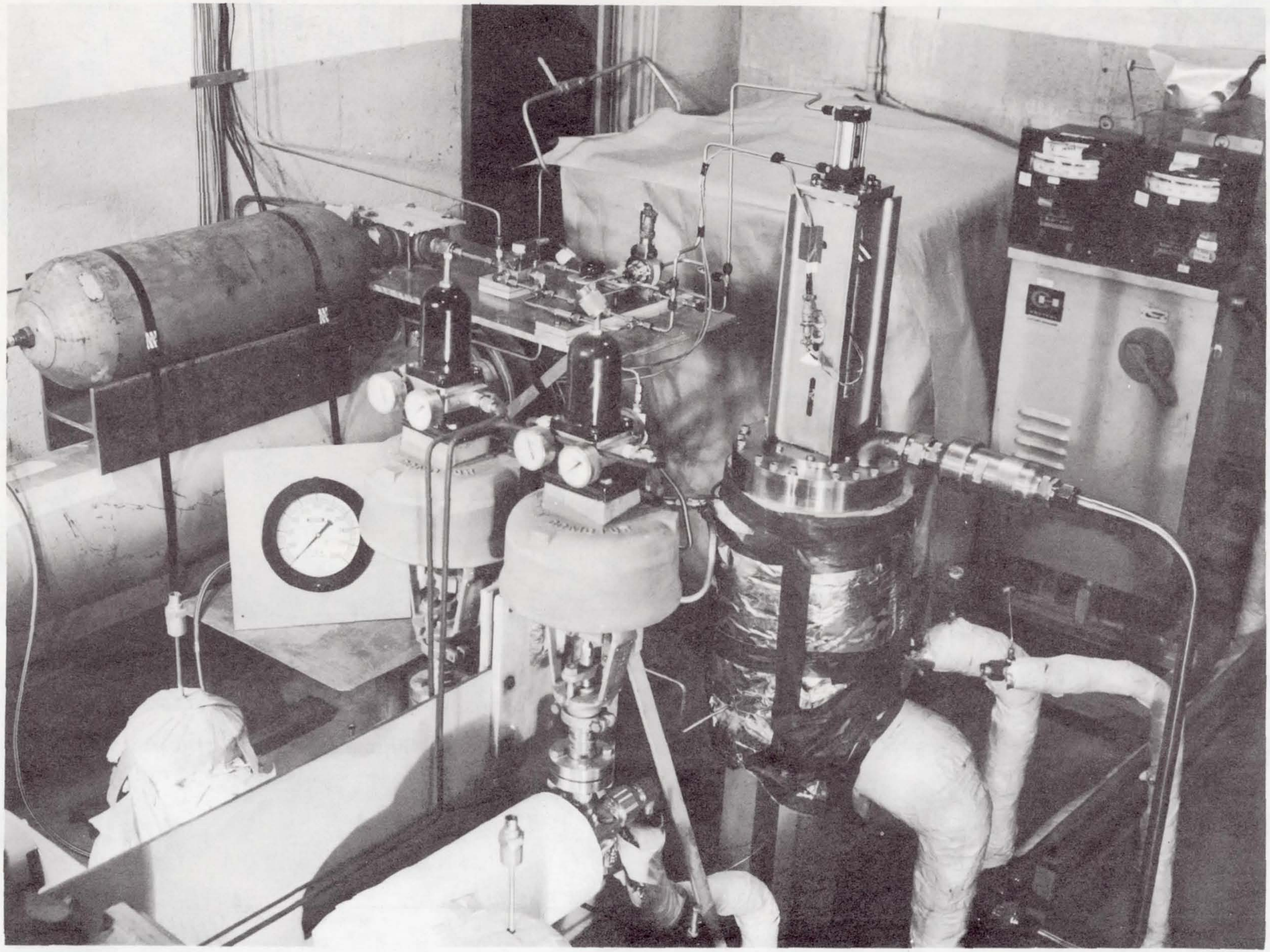
Rapid screening tester

Cold conditioning system utilizing LN_2 and GN_2

Hot conditioning system utilizing electrically heated GN_2

Hydraulic feed system including accumulator, servo valve, throttling valves, check valves and differential pressure transducer.

NEG. 71-106-1

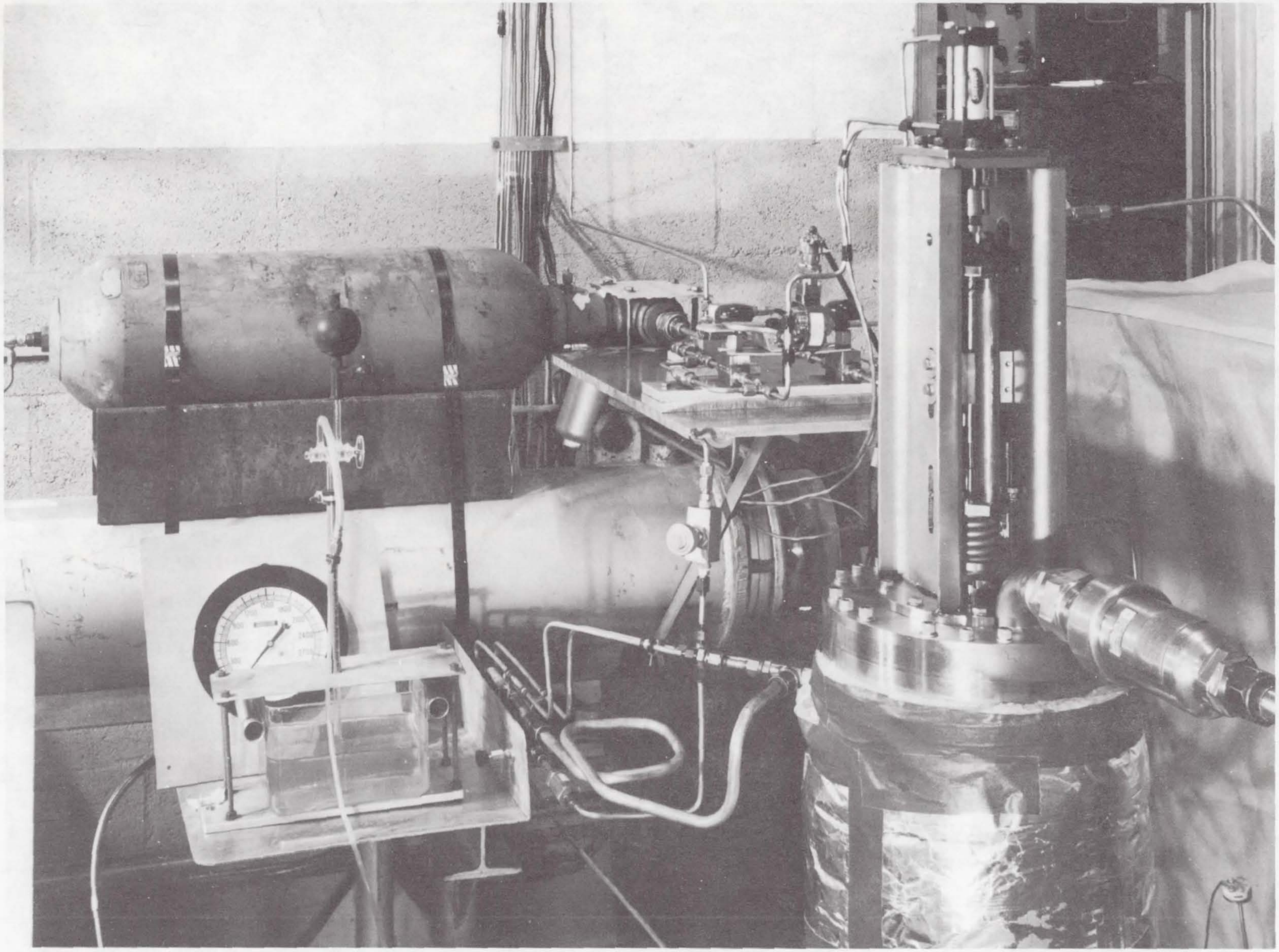


SEALING CLOSURE EVALUATION TEST SET-UP

THE Marquardt COMPANY
16555 SATICOY STREET
VAN NUYS, CALIFORNIA 91406

A closer view of the rapid screening tester also shows the GN_2 inlet filter, water displacement leakage measuring setup, and hydraulic feed system. In the background is a door leading to the next room where the instrumentation and controls are located.

NEG. 71-106-2



RAPID SCREENING & CONDITIONING UNIT

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A close up of the top of the rapid screening tester shows the

Hydraulic actuator

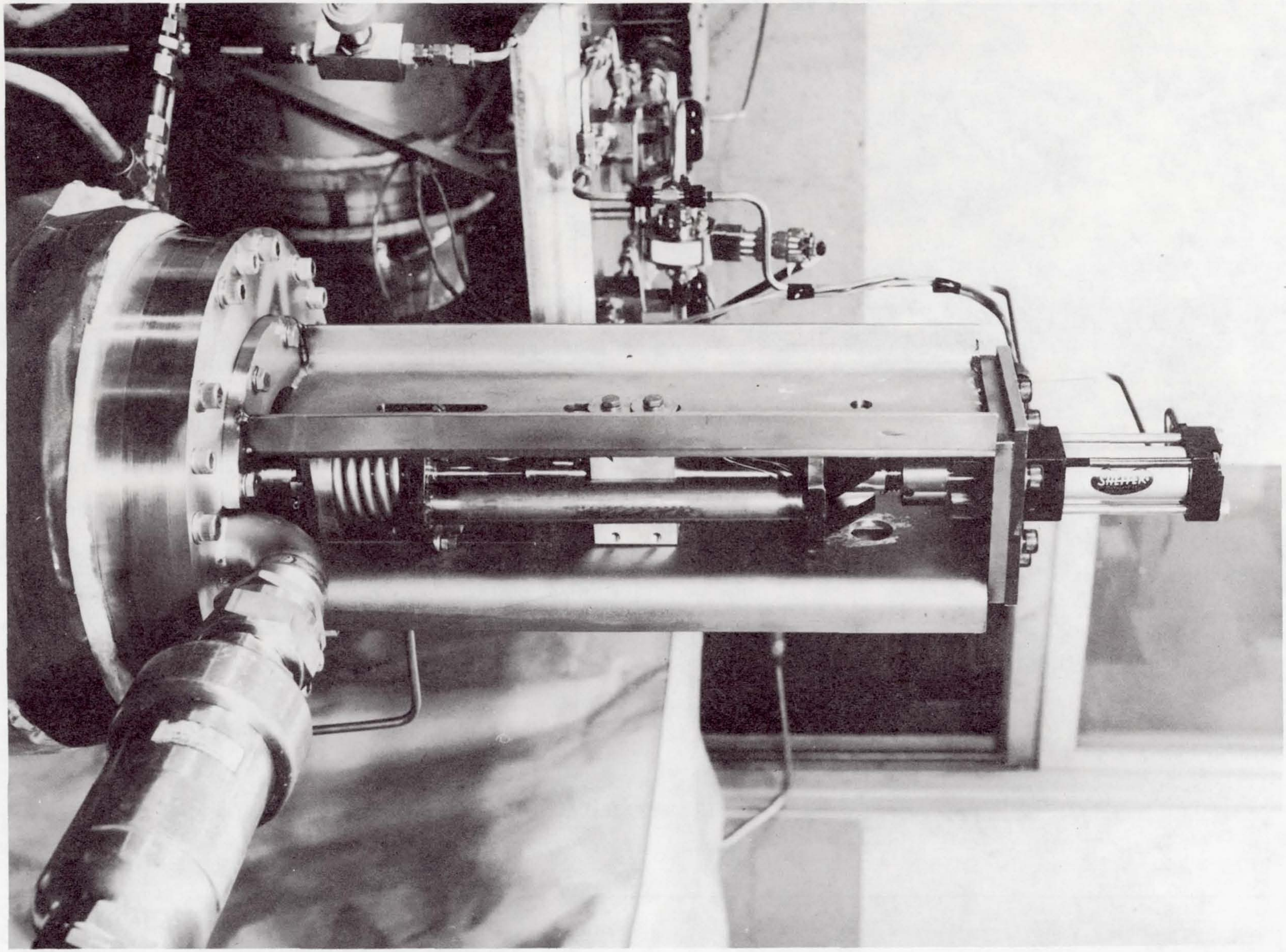
Stroke adjustment nut

Actuator yoke and variable reluctance position transducer

Isolation spring

GN_2 inlet filter.

NEG. 71–106–3



ACTUATOR MECHANISM--RAPID SCREENING TESTER

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Controls and instrumentation shown in this view include:

Frequency generator

Pulser-Driver

Counter

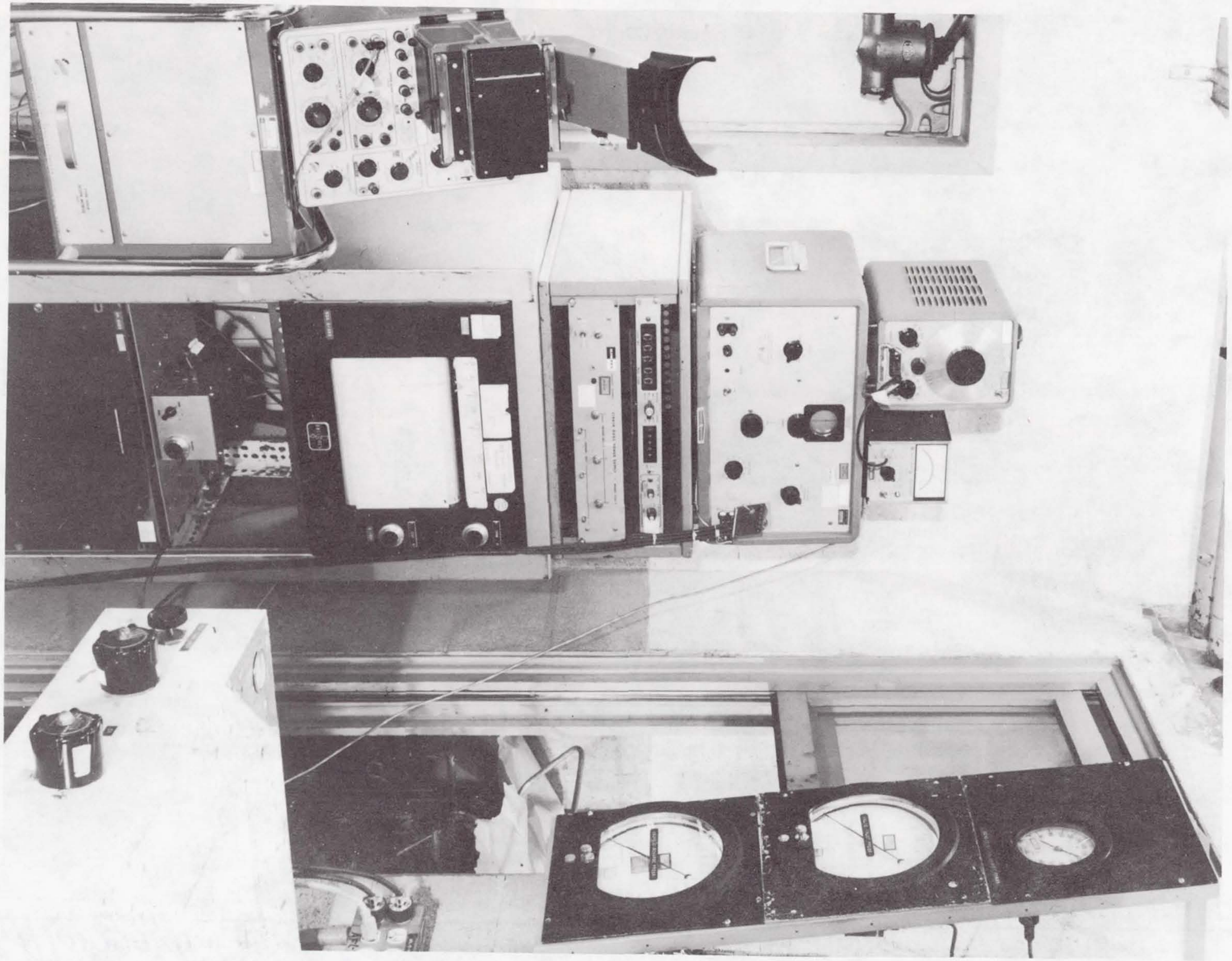
Dual beam oscilloscope

Bristol recorder for temperatures

Various pressure gages

Control regulators.

NEG. 71-106-4



INSTRUMENTATION SET-UP --SEALING CLOSURE TESTS

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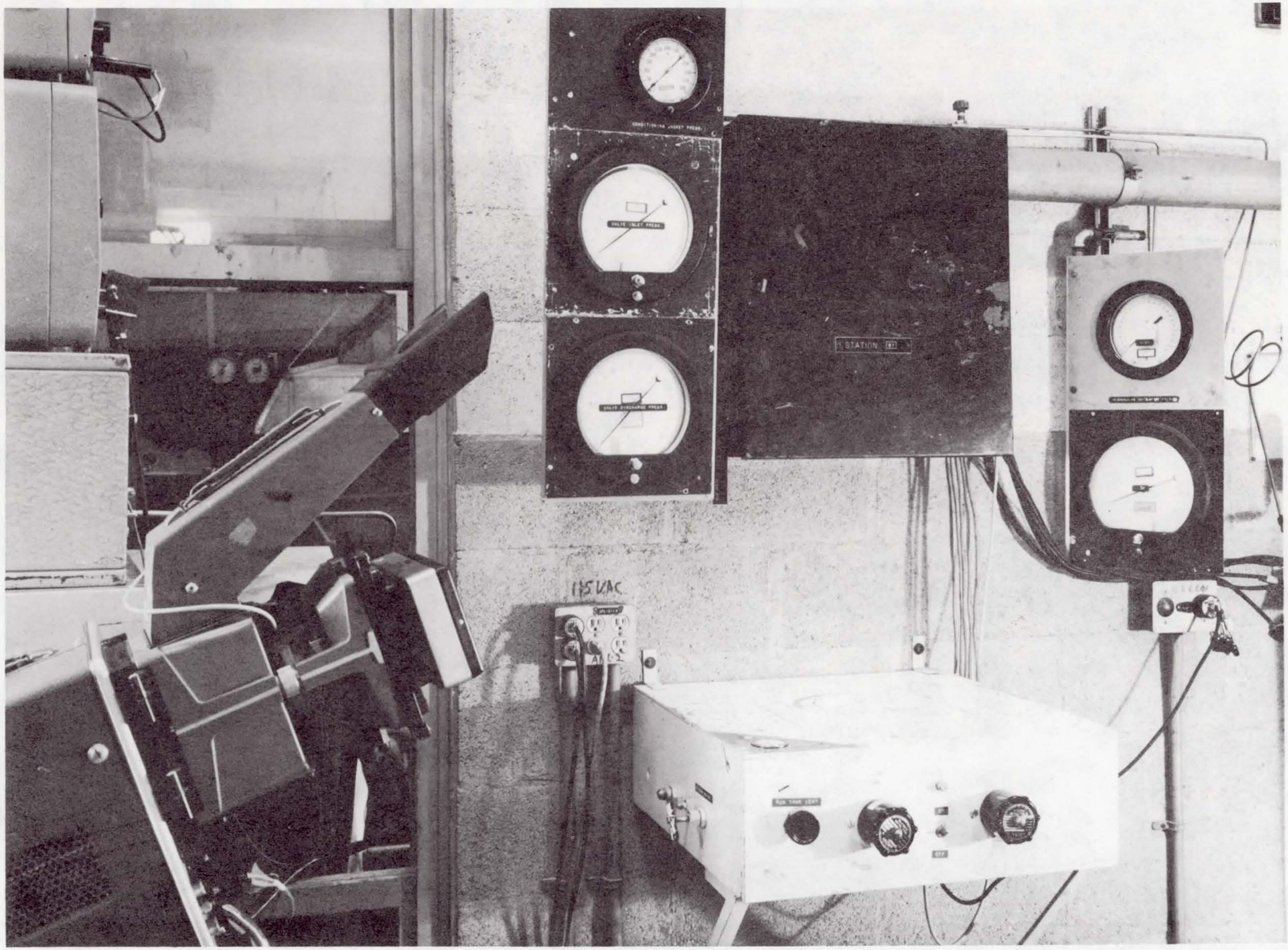
Another view of the control and instrumentation area showing additional pressure gages.

NEG. 71-106-5



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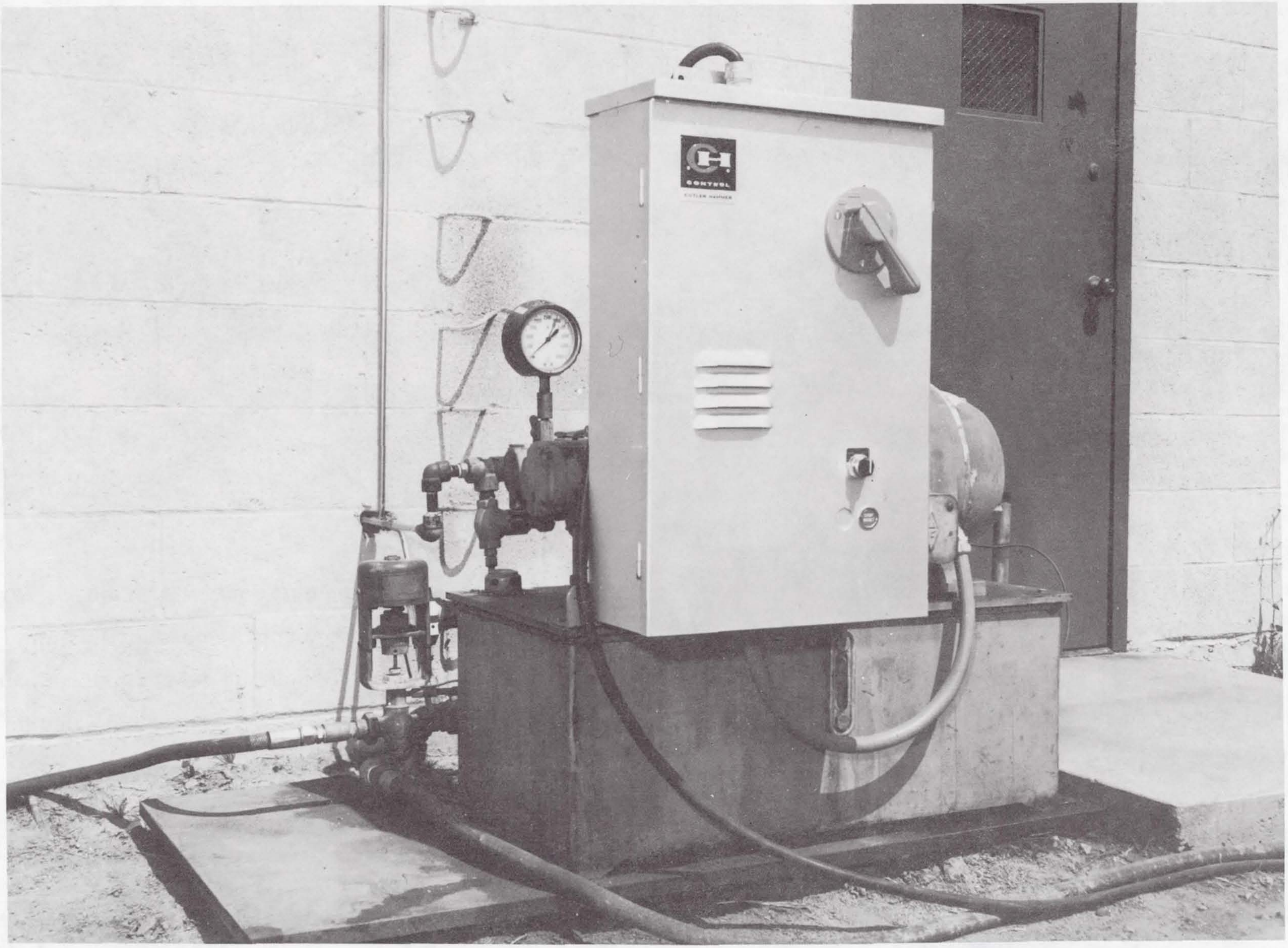
PRESSURE GAGES--SEALING CLOSURE TESTS

Hydraulic supply for the sealing closure evaluation test program.



HYDRAULIC POWER SUPPLY SYSTEM--SEALING CLOSURE TESTS

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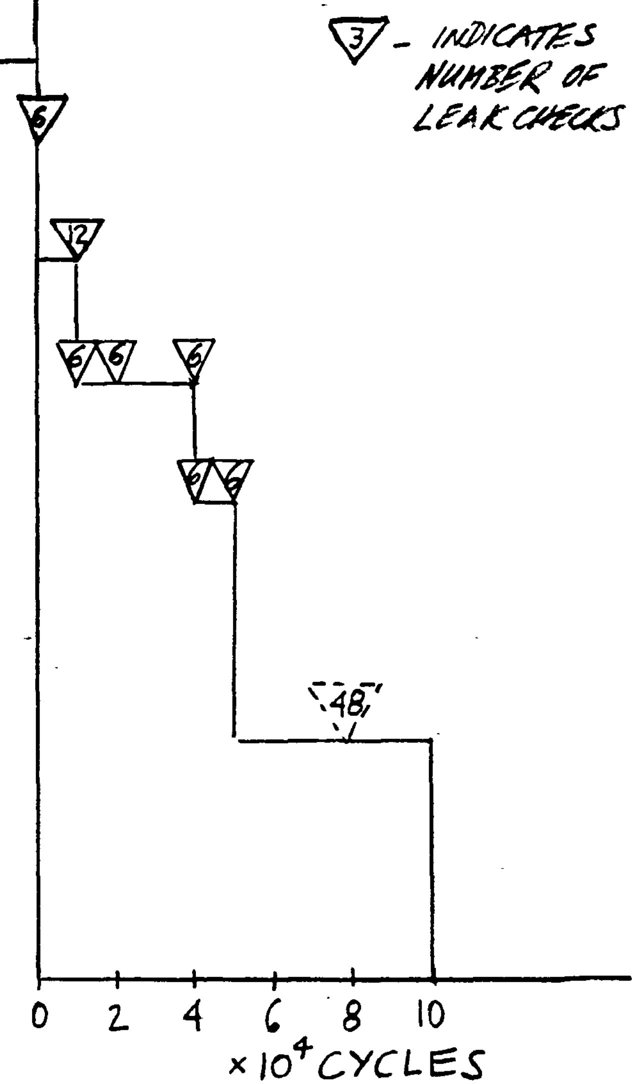
The primary purpose of the sealing closure evaluation tests is to accumulate a large number of cycles with each sealing closure at the nominal pressure settings and at the temperature extremes. Approximately 100,000 cycles will be accumulated with each sealing closure. Other variables to be explored include poppet alignment, impact kinetic energies and sealing closure surface finishes.

SEALING CLOSURE SCREENING TEST MATRIX

	TEST ENVIRONMENT
1	RESPONSE, STROKE & LEAKAGE @ 3 STATIC LOADS AND @ 20 & 400 PSIA, AMBIENT TEMP
2	CYCLING @ 400 PSIA, 850°R
3	CYCLING @ 400 PSIA, AMBIENT TEMP
4	CYCLING @ 400 PSIA, 200°R
5	REPEAT ITEMS 1-4 ABOVE AT IMPACT KINETIC ENERGY LEVELS 5 TIMES GREATER AND 20 PSIA OPERATING PRESSURE



Late last year, The Marquardt Company completed NASA Contract NAS 9-10886, the purpose of which was an analytical trade-off study of various types of injector valve concepts and their performance characteristics as applicable to gaseous oxygen/gaseous hydrogen rocket engines in the 500 to 2500 lbs. thrust range for Space Stations. This study concluded with recommendations for valve designs for the 1000 lbs. thrust level operating at 20 and 400 psia inlet pressure.

The same analysis techniques that were developed under that contract were also applied to the specific requirements of this program. The trade-off includes consideration of a variety of shut off devices and actuators.

VALVE ELEMENT - MOTION CHARACTERIZATION

MOTION TYPE	LINEAR	ROTARY
Shut Off Device	Poppet Plug Gate Sliding Spool Blade Diaphragm or Boot	Ball Butterfly Swing Flapper
Actuator	Cylinder Solenoid Piezoelectric Magnetostrictive Thermal Expansion Linear Motor	Torque Motor Electrical Motor Hydraulic or Pneumatic Motor Inertia Wheel

The major valve requirements of the "Space Shuttle Auxiliary Propellant Valves" contract which were used in the trade-off utilizing the NASA Contract NAS 9-10886 programs are presented on the opposite page.

VALVE PERFORMANCE REQUIREMENTS

- HIGH PRESSURE - 400 PSIA
  - W_{OX} - 2.76 LBS/SEC. AT 540°R
  - W_F - 0.69 LBS/SEC. AT 540°R (F = 1500 LBS, O/F = 4.0)
  - ALLOWABLE PRESSURE DROP - 5PSI
- LOW PRESSURE - 20 PSIA
  - W_{OX} - 2.86 LBS/SEC. AT 540°R
  - W_F - 1.14 LBS/SEC. AT 540°R (F = 1500 LBS, O/F = 2.5)
  - ALLOWABLE PRESSURE DROP - 1 PSI
- RESPONSE
  - SIGNAL ON TO VALVE FULLY OPEN - 30 MS MAXIMUM
  - SIGNAL OFF TO VALVE FULLY CLOSED - 30 MS MAXIMUM
  - MAXIMUM TRAVEL TIME - 15 MS HIGH PRESSURE
    - 20 MS LOW PRESSURE
- INTERNAL LEAKAGE - 100 SCC/HR H_E MAXIMUM AT ANY OPERATING PRESSURE AND TEMPERATURE
- OPERATING TEMPERATURE - 200-800°R
- LIFE (goal) - 1,000,000 CYCLES AND 10 YEARS WITH ZERO MAINTENANCE

For the low pressure oxidizer valves, the response and pressure drop requirements can be met only by poppet valves utilizing high pressure hydraulic or pneumatic cylinder actuators or by a ball valve featuring a 3000 psi hydraulic actuator if the pilot valve power requirement is limited to 28 watts. Some other combinations are also feasible at higher electrical power levels.

Line pressure operation is not feasible.

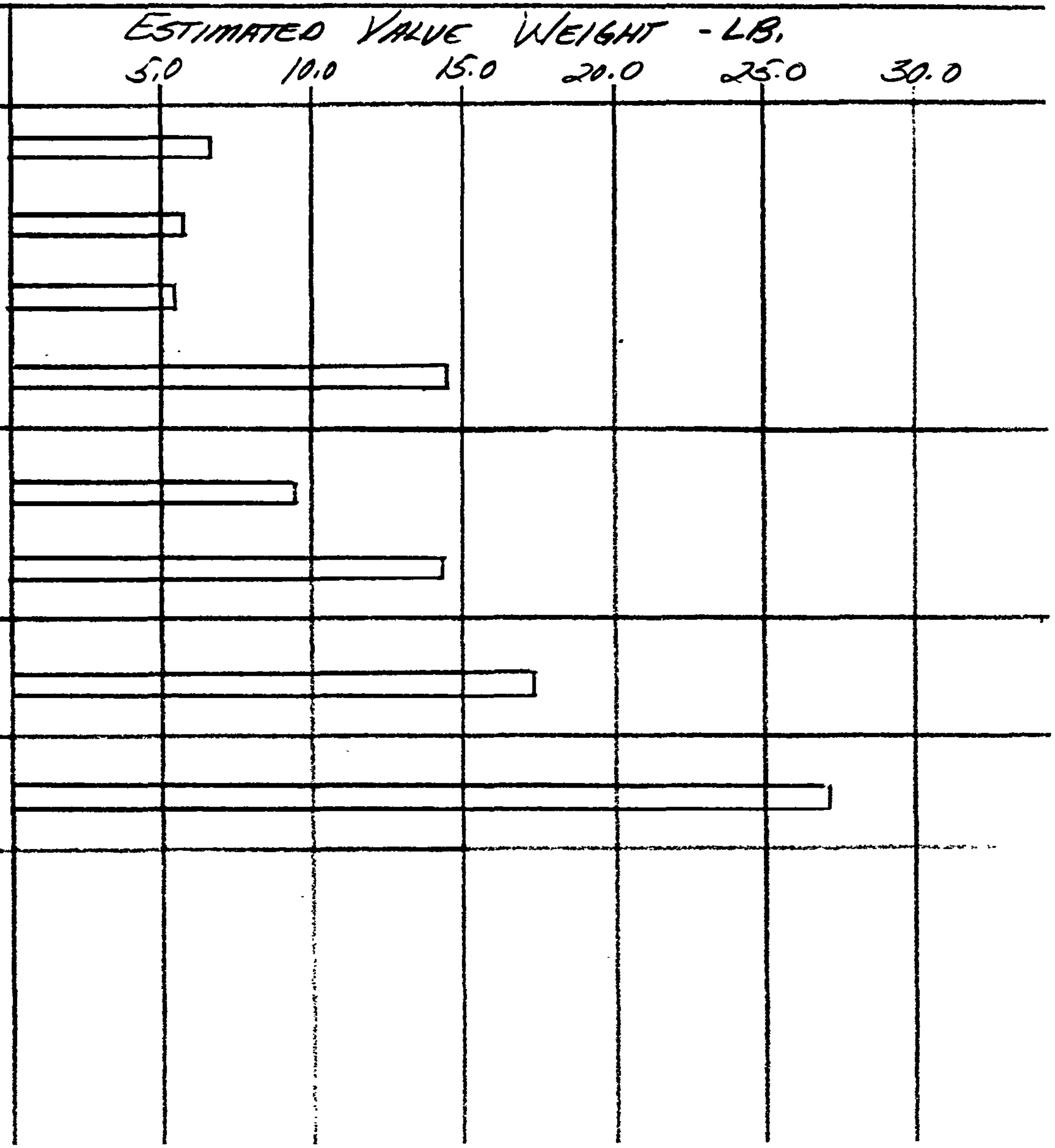
The lowest weight approaches consist of high pressure actuated poppet valves.

The weights shown are minimums and do not include inlet and outlet fittings or special components such as position indicators, bellows, etc.

Actuation pressures less than 1500 psi are propellant (oxygen in this case) pressures. Otherwise, the actuation gas is Helium.

LOW PRESSURE (20 PSIA) O_2 INJECTOR VALVE CHARACTERISTICS
RESPONSE TIME $\leq$ 30 MILLISECONDS

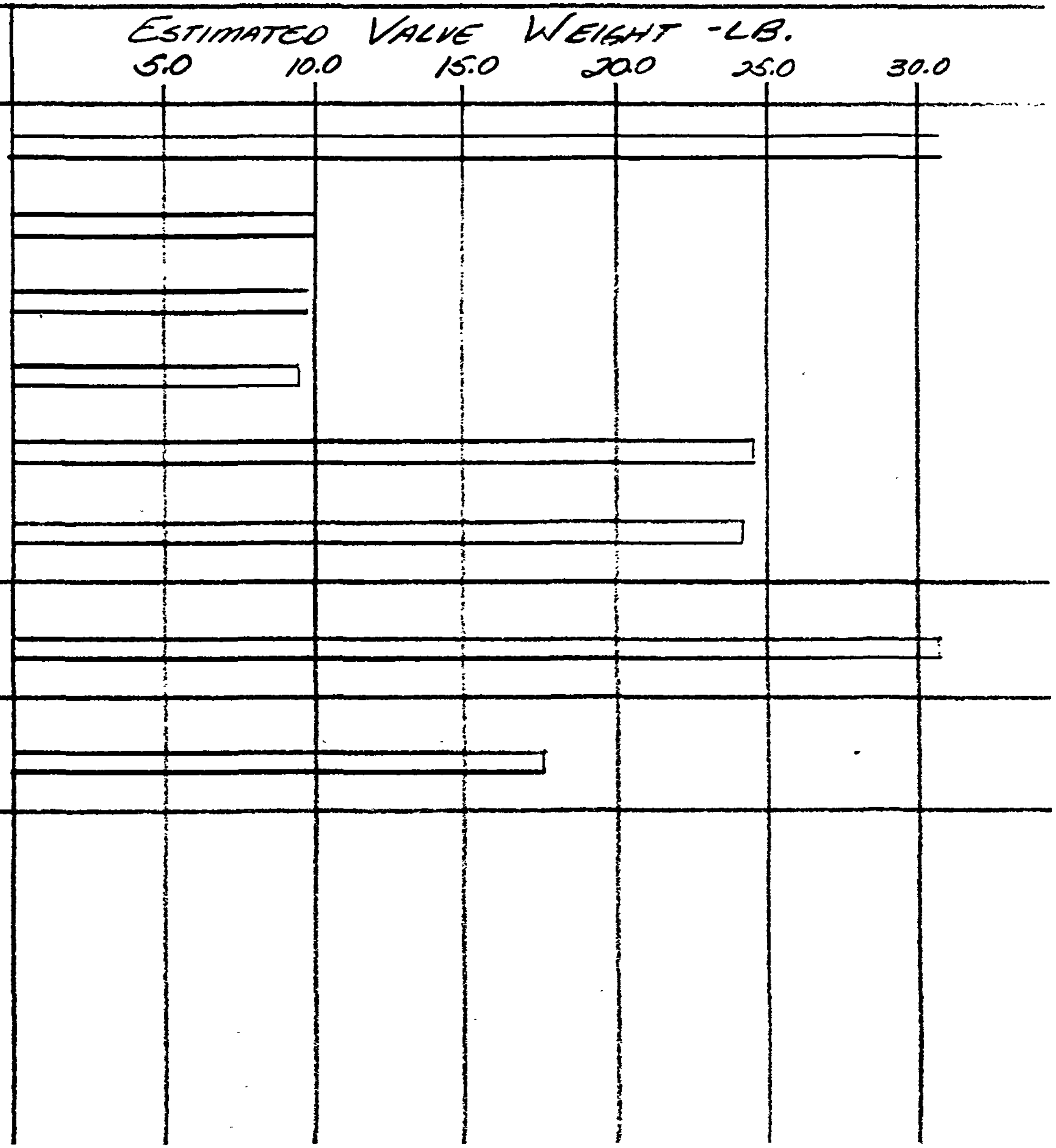
VALVE CONFIGURATION		ELECTRICAL POWER @ 70°F, 32 vdc (WATTS)
SHUT-OFF	ACTUATOR	
POPPET	1500 PSI CYL.	28 MIN
↓	3000 PSI CYL.	
	HYDRAULIC CYL.	
BALL	HYDRAULIC CYL.	
POPPET	150 PSI CYL.	56 MIN
BALL	3000 PSI CYL.	
BALL	1500 PSI CYL.	84 MIN
POPPET	60 PSI CYL.	112 MIN



Although the low pressure hydrogen valves are larger than the low pressure oxygen valves, the conclusions from the trade-off study are the same. Some of the less desirable concepts feasible for the oxidizer valve (such as the 60 psi actuation pressure, cylinder actuated poppet valve) no longer meet the response and pressure drop requirements. Minimum valve weights are approximately 10 lbs.

LOW PRESSURE (20 PSIA) H_2 INJECTOR VALVE CHARACTERISTICS
RESPONSE TIME ≦ 30 MILLISECONDS

VALVE CONFIGURATION: SHUT-OFF	VALVE CONFIGURATION: ACTUATOR	ELECTRICAL POWER @ 70°F, 32 vdc (WATTS)
POPPET	150 PSI CYL.	28 MIN.
↓	1500 PSI CYL.	
	3000 PSI CYL.	
	HYDRAULIC CYL.	
BALL	3000 PSI CYL	
↓	HYDRAULIC CYL.	
BALL	1500 PSI CYL.	56 MIN.
POPPET	60 PSI CYL.	84 MIN.



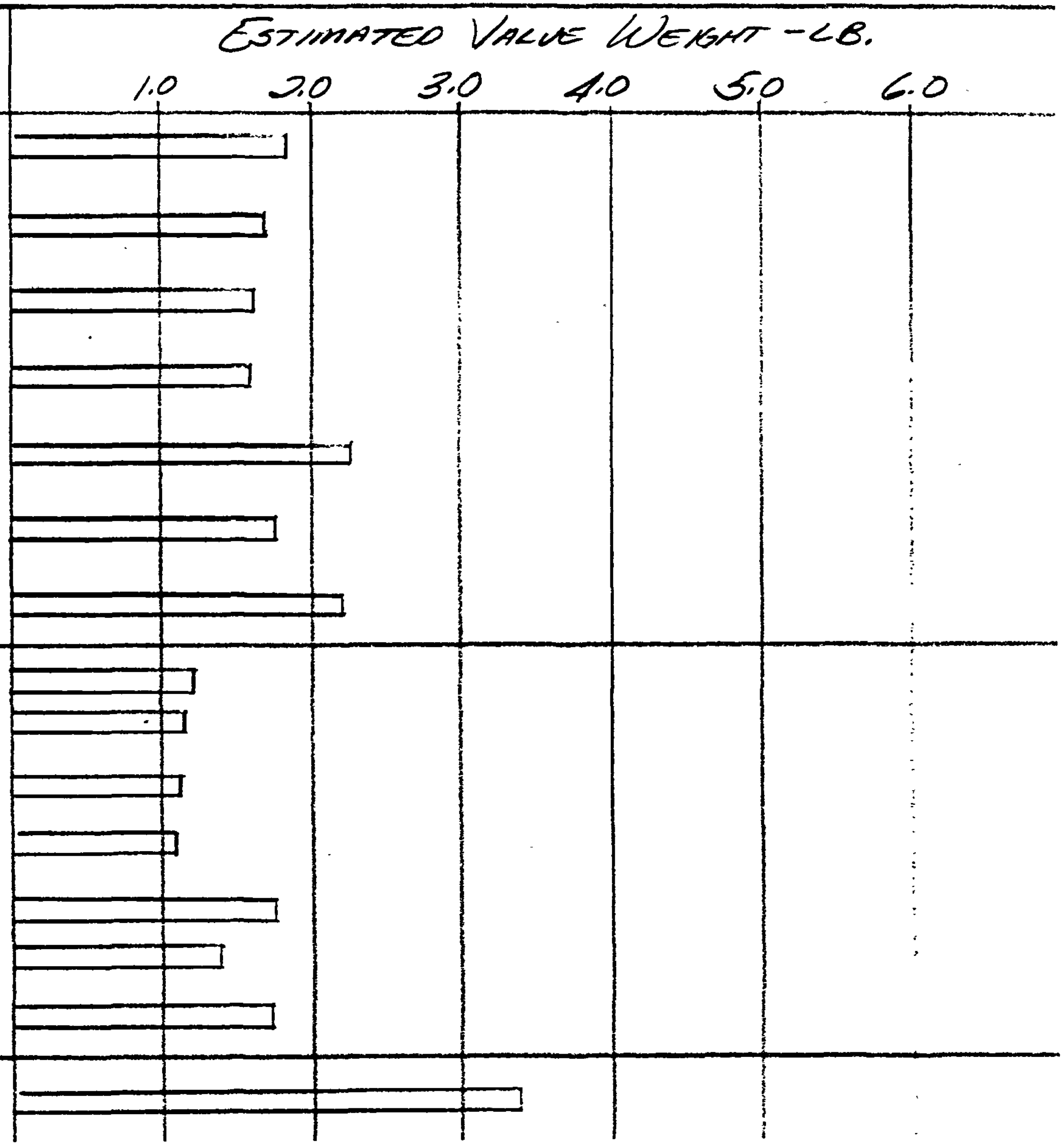
For the high pressure application, the oxidizer and fuel valve sizes are the same. In general, poppet valves tend to be lighter weight than ball valves.

Propellant line pressure operation is feasible at the 28 watt power level for the poppet valve with negligible weight penalties incurred in comparison to high pressure operation.

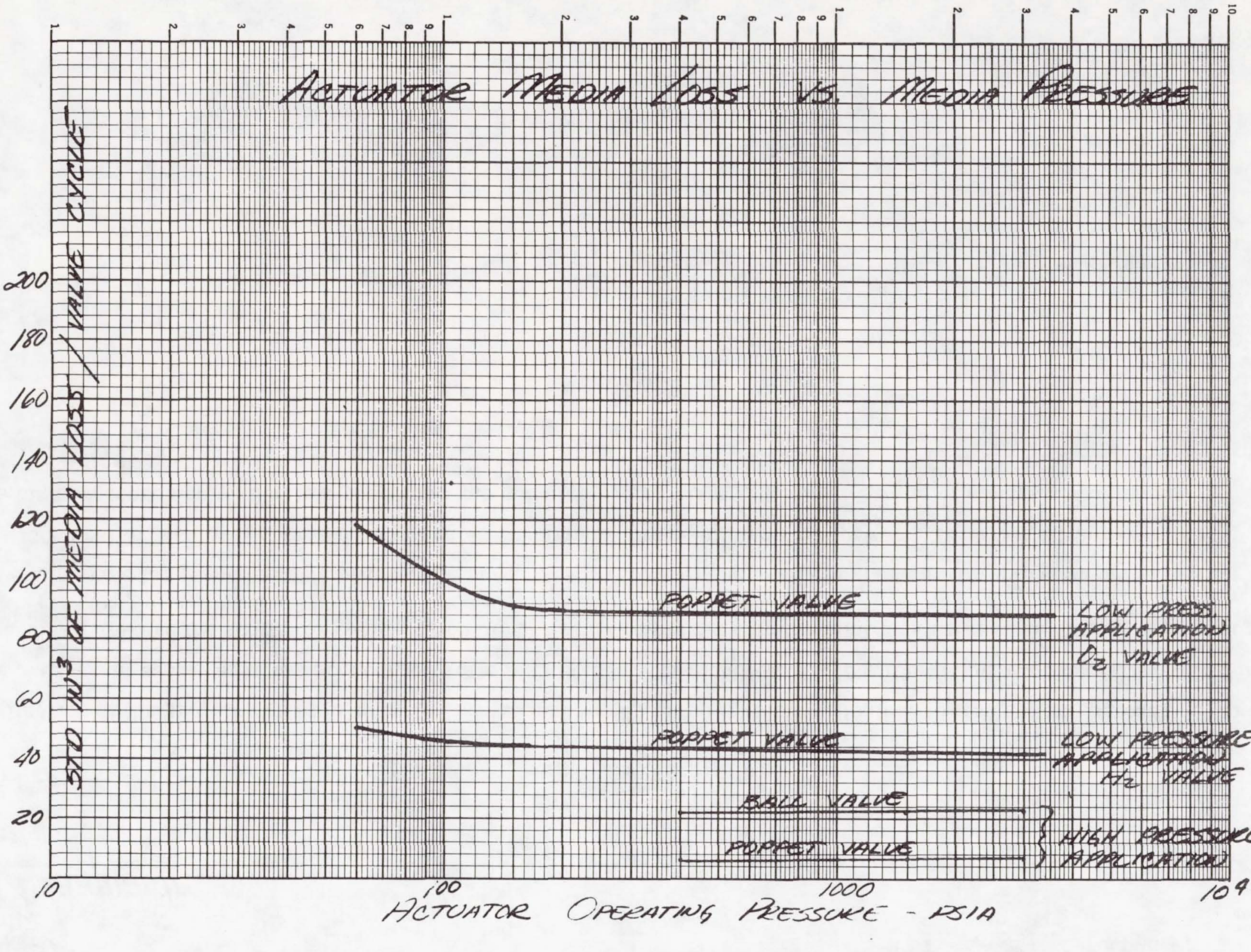
Increasing the electrical power level from 28 to 56 watts results in an approximately 25% weight savings. Further increases in power do not further reduce the weight. However, a propellant pressure operated ball valve does become feasible at the 56 watt level.

HIGH PRESSURE (400 PSIA) O_2/H_2 INJECTOR VALVE CHARACTERISTICS
RESPONSE TIME $\leq$ 30 MILLISECONDS

VALVE CONFIGURATION — SHUT-OFF	VALVE CONFIGURATION — ACTUATOR	ELECTRICAL POWER @ 70°F, 32 vdc (WATTS)
POPPET	400 PSI CYL.	
↓	1500 PSI CYL	
↓	3000 PSI CYL.	
↓	HYDRAULIC CYL.	28
BALL	1500 PSI CYL	
↓	3000 PSI CYL	
↓	HYDRAULIC CYL	
POPPET	400 PSI CYL	
↓	1500 PSI CYL	
↓	3000 PSI CYL.	
↓	HYDRAULIC CYL	56
BALL	1500 PSI CYL	
↓	3000 PSI CYL	
↓	HYDRAULIC CYL	
BALL	400 PSI CYL.	84



Examination of the volume of pressurant lost discloses that the volume is essentially the same regardless of the operating pressures except that for the low pressure valves, some increase occurs as the pressure drops below 200 psi. The high pressure poppet valves require the least volume of pressurant for operation and the actual amount is negligible as far as engine operations is concerned.



ACTUATOR MEDIA LOSS VS. MEDIA PRESSURE
STD IN3 OF MEDIA LOSS / VALVE CYCLE
200
180
160
140
120
100
80
60
40
20
10
100
1000
10 4
ACTUATOR OPERATING PRESSURE - PSIA
POPPET VALVE
LOW PRESS APPLICATION O2 VALVE
POPPET VALVE
LOW PRESSURE APPLICATION H2 VALVE
BALL VALVE
POPPET VALVE
HIGH PRESSURE APPLICATION

Utilization of the propellant as the actuation pressurant supply is the least complicated, lowest weight approach as far as the pressurant supply system is concerned since high pressure pneumatics or hydraulics require additional system components. Hydraulics feature the additional complication of not being compatible with the minimum valve temperature requirements (200°R).

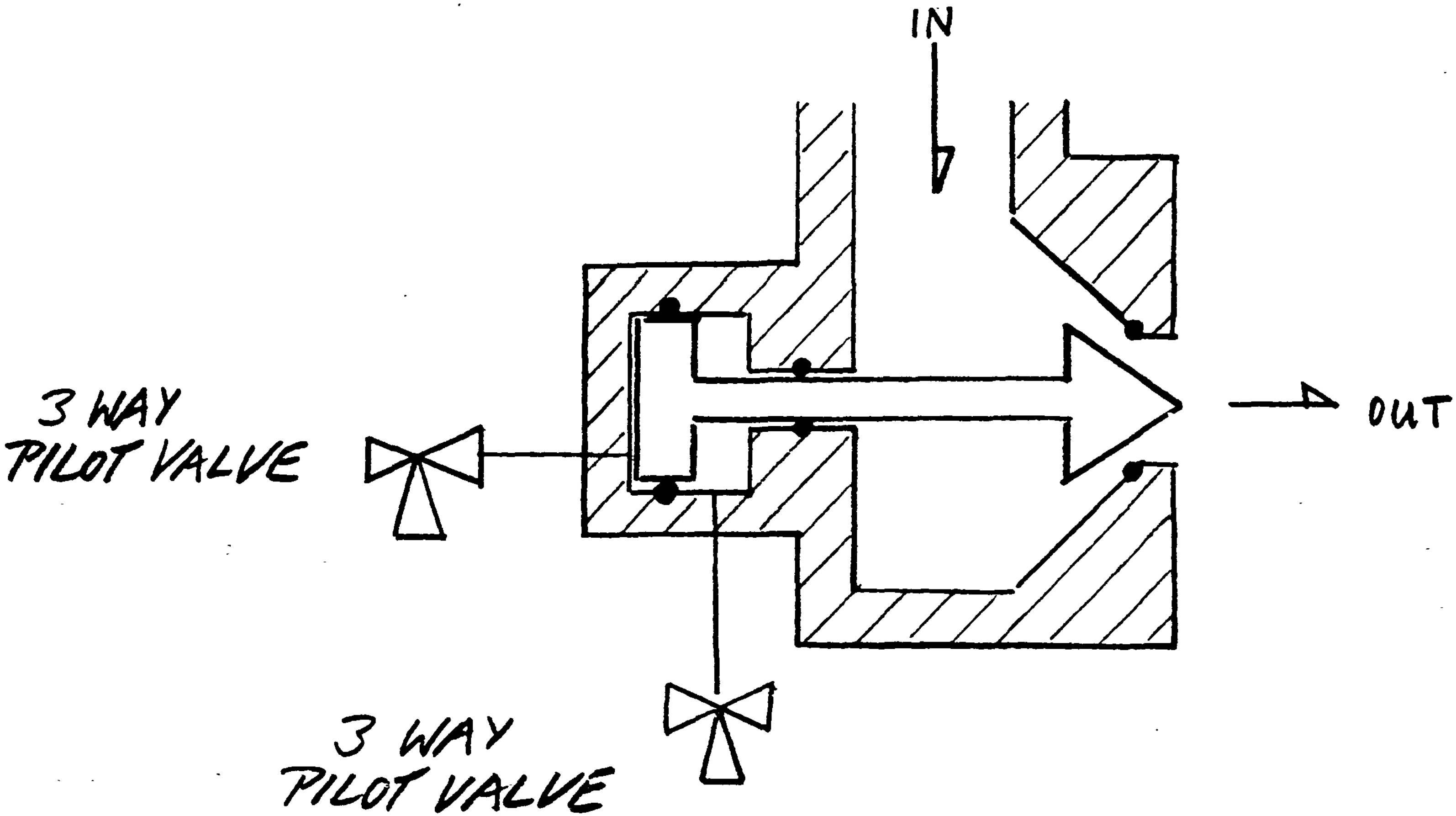
To summarize the preceding five charts, the most attractive valve type for the low pressure valve application is a high pressure gas (3000 psi or 1500 psi) operated poppet valve requiring 28 watts of electrical power. The most attractive valve type for the high pressure application is a propellant gas actuated poppet valve requiring 56 watts of electrical power.

PRESSURANT SUPPLY SYSTEM PENALTIES

PROPELLANT	HIGH PRESSURE GAS	HIGH PRESSURE HYDRAULIC
VENT LINE	VENT LINE	RETURN LINE
	PRESSURIZING LINE	PRESSURIZING LINE
	PRESSURE REGULATION	PRESSURE REGULATION
	OVER PRESSURE PROTECTION	OVER PRESSURE PROTECTION
	FILTRATION	FILTRATION
	INSTRUMENTATION	INSTRUMENTATION
	REDUNDANCY CONSIDERATION	REDUNDANCY CONSIDERATION
	TANKAGE	RESERVOIR
	BOOSTER PUMPS	PUMP
	PUMP DRIVES	PUMP DRIVE
	ENERGY SOURCES	ENERGY SOURCES

The Marquardt Company has prepared a number of analog computer programs to permit optimization of critical valve subcomponents. These programs are set up for those valve concepts which appear most attractive at this time. One of these concepts is a double pressure actuated poppet valve. Subsequent charts present the type of data being generated with the analog computer programs for this valve concept.

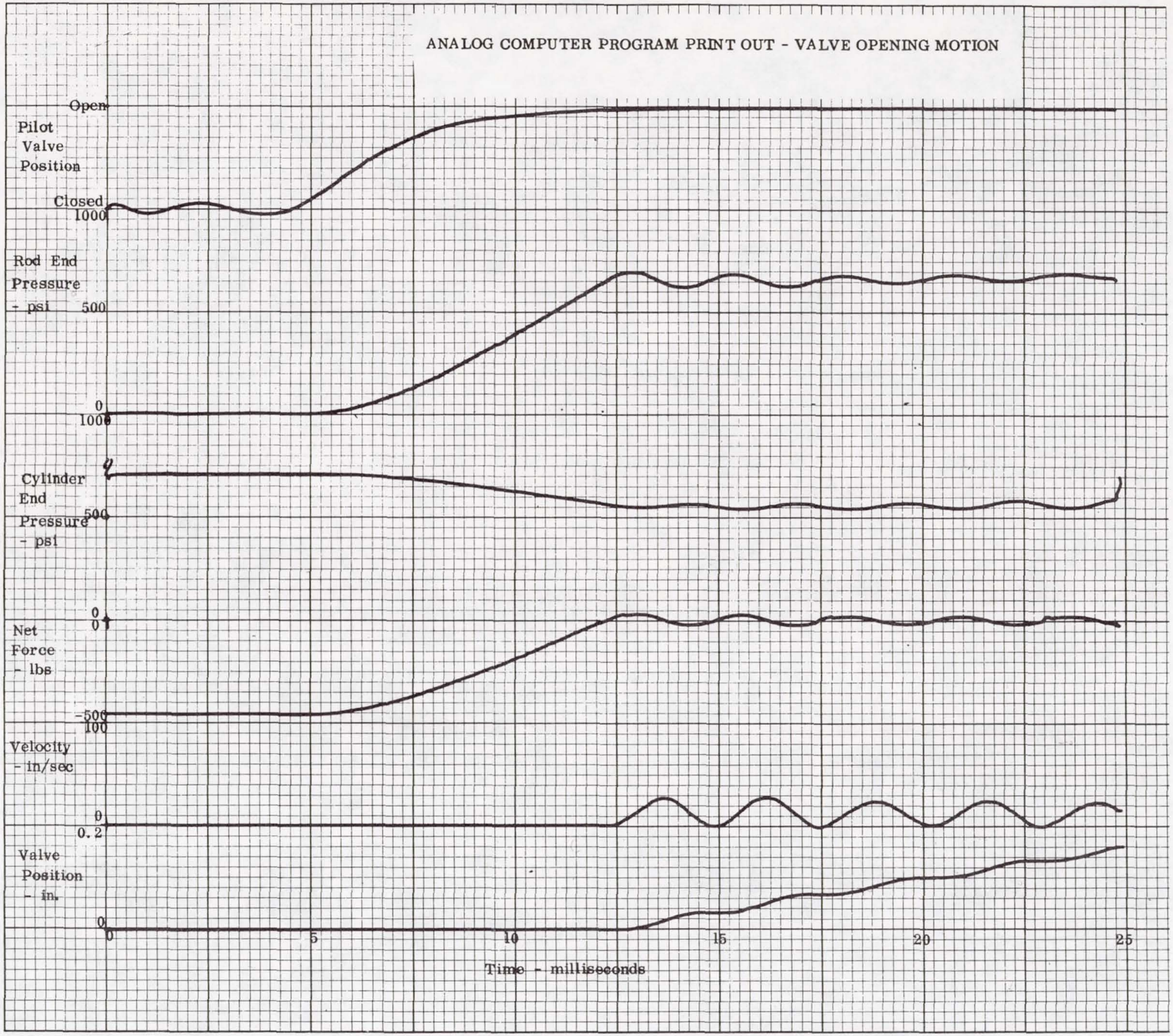
SCHEMATIC OF A PNEUMATICALLY OPERATED POPPET VALVE



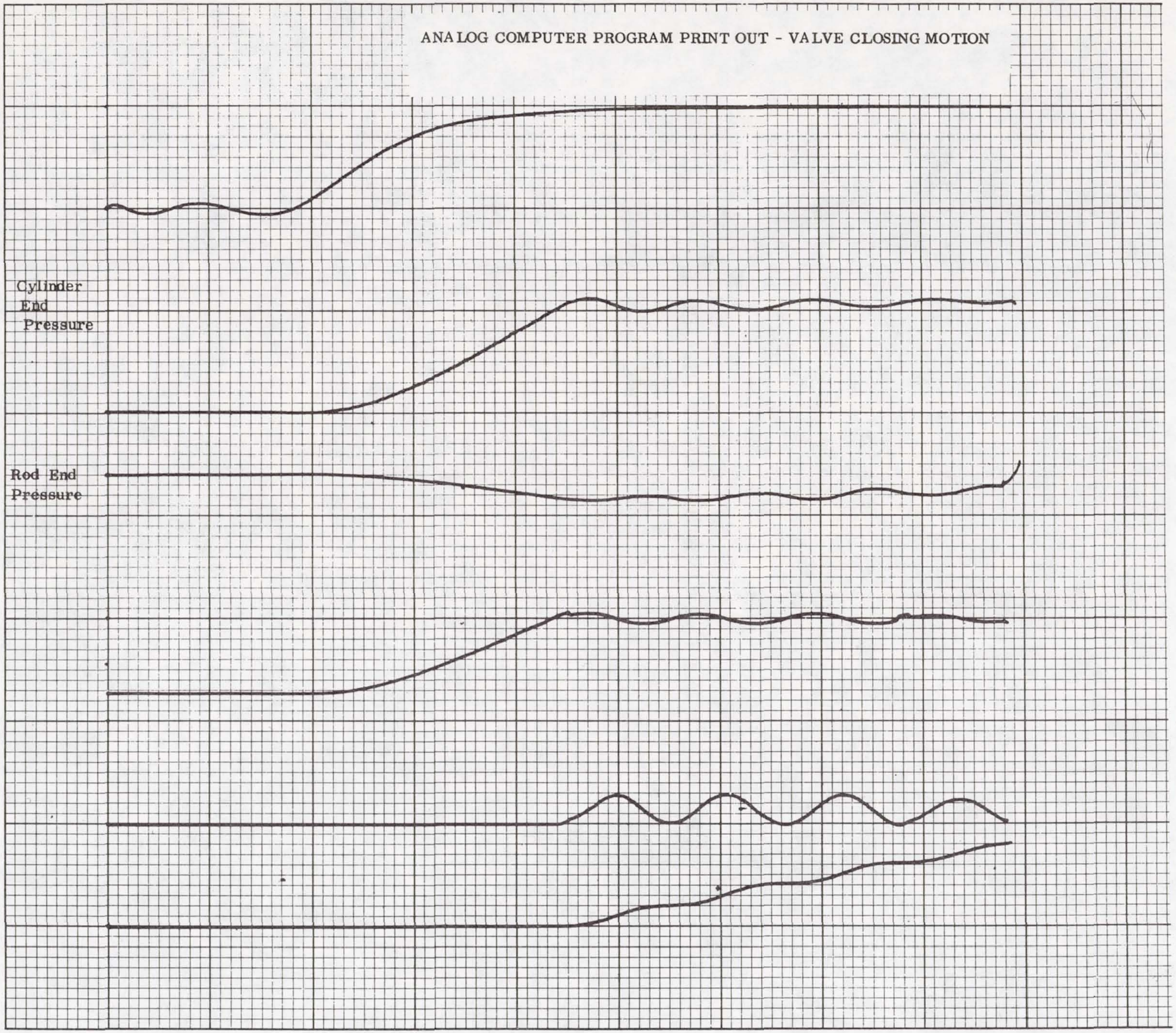
Data from the analog computer program is printed out by a Sanborn recorder. The printout is useful in determining specific sizes, such as the pilot valve orifice size, required to accomplish a certain response as well as to give a pictorial presentation of the actuation pressures and the resultant poppet velocities and poppet positions. Undesirable oscillations become very apparent.

The programs have been used extensively to perform sensitivity studies such as determining the effect of changes in moving mass on response or the effect of the piston area on response, etc.

The traces shown indicate considerable variation in poppet velocity but a fairly linear poppet position versus time characteristic during the opening motion.


ANALOG COMPUTER PROGRAM PRINT OUT - VALVE OPENING MOTION
Open
Pilot Valve Position
Closed
1000
Rod End Pressure - psi
500
0
1000
Cylinder End Pressure - psi
500
0
0
Net Force - lbs
-500
100
Velocity - in/sec
0
0.2
Valve Position - in.
0
0
5
10
15
20
25
Time - milliseconds

These traces show the same parameters as the preceding chart but during the closing motion.


ANALOG COMPUTER PROGRAM PRINT OUT - VALVE CLOSING MOTION
Cylinder End Pressure
Rod End Pressure

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"SPACE SHUTTLE ACPS SHUTOFF VALVE"

G. M. SMITH

ROCKETDYNE

TECHNICAL MANAGER

R. GREY

LEWIS RESEARCH CENTER

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TITLE: SPACE SHUTTLE AUXILIARY PROPELLANT VALVES

CONTRACT: NAS3-14350

PRESENTER: G. M. SMITH, PRINCIPAL ENGINEER

COMPANY: ROCKETDYNE, DIVISION OF NORTH AMERICAN ROCKWELL

NASA PROJECT MANAGER: R. GREY

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SPACE SHUTTLE AUXILIARY PROPULSION SYSTEM VALVES

NAS3-14350

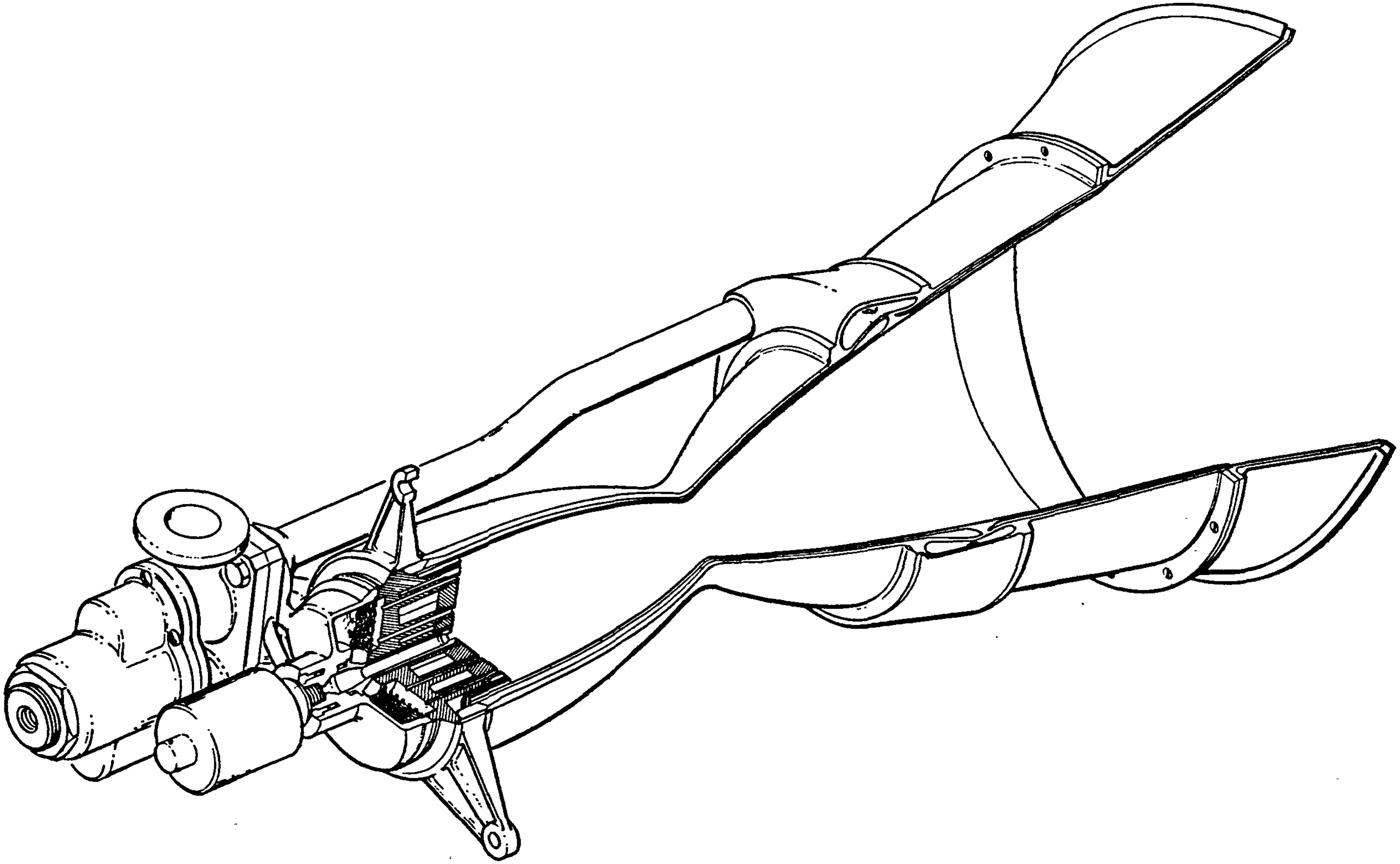
OBJECTIVE: TO DEVELOP AND DEMONSTRATE DESIGN CRITERIA FOR FLIGHT-TYPE GASEOUS HYDROGEN - GASEOUS OXYGEN PROPELLANT SHUTOFF VALVES TO BE USED ON THE THRUSTERS IN THE SPACE SHUTTLE AUXILIARY PROPULSION SYSTEM

The subject of this briefing is the program being conducted to develop technology and design criteria necessary to successfully design and fabricate an APS Propellant Valve, illustrated in this chart.

T275-829

SPACE SHUTTLE AUXILIARY PROPULSION SYSTEM

275-829
3-71



Rocketdyne
North American Rockwell

From the contract design requirements, these were picked as being of most significant impact upon a potential valve design. The operating life goal of 10^6 cycles coupled with the extremely low leakage allowed, represent a stringent requirement for valves of the size needed for the APS system.

DESIGN REQUIREMENTS

- OPERATING LIFE (GOAL) ---------------- 1,000,000 CYCLES
- INTERNAL LEAKAGE ---------------- 100 SCC/HR
- MAINTENANCE ---------------- ZERO
- RESPONSE ---------------- 30 MS MAXIMUM
- OPERATING TEMPERATURE ---------------- 200 R TO 850 R
- VALVE INLET PRESSURE ---------------- 20 ±5 PSIA LOW PRESSURE SYSTEM
  400 ±50 PSIA HIGH PRESSURE SYSTEM

The technical effort to be accomplished during the contract is conducted in the four tasks shown, along with a continuing reporting task. The closure concepts developed in Task I are tested in Task IIIA; these test data are then used to modify the original concepts, so that valve preliminary design (Task II) can proceed.

These preliminary valve designs will be used to design and fabricate valve simulating fixtures (laboratory-type valves) to perform the Task IIIB evaluations, thus leading to the final flightweight valve designs of Task IV.

PROGRAM LOGIC

TASK I

VALVE SUBCOMPONENT
ANALYSIS & CONCEPTUAL DESIGN

TASK II

VALVE PRELIMINARY DESIGN

TASK IIIA

SEALING CLOSURE SCREENING

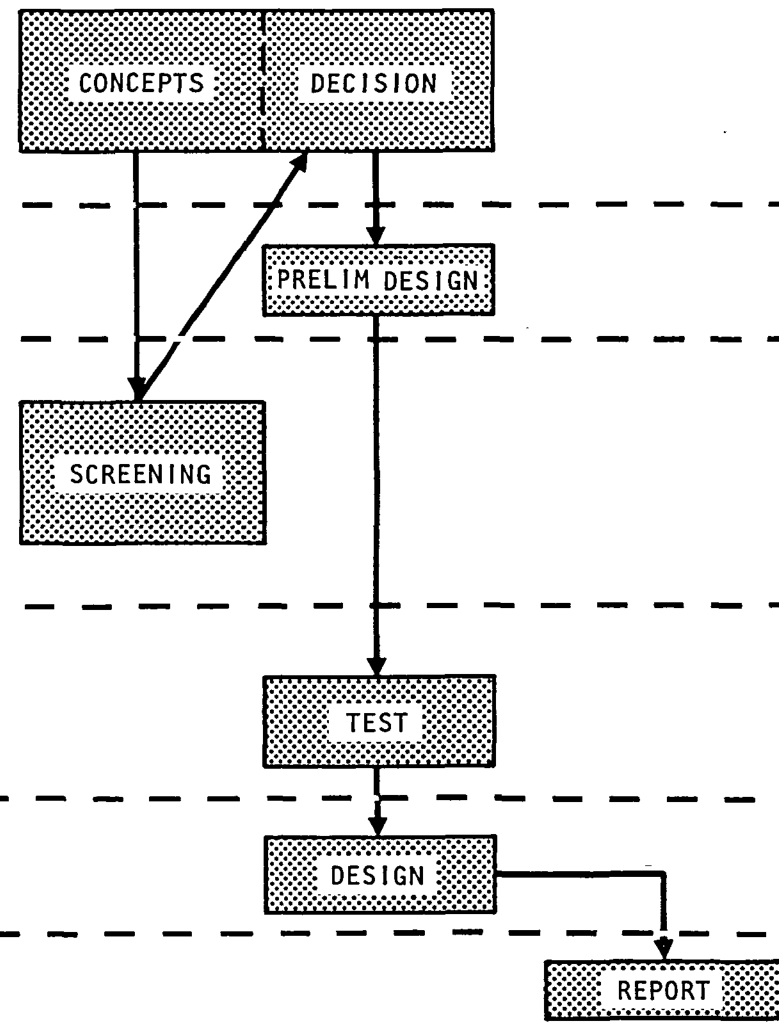
TASK IIIB

VALVE SUBCOMPONENT EVALUATION

TASK IV

VALVE DESIGN

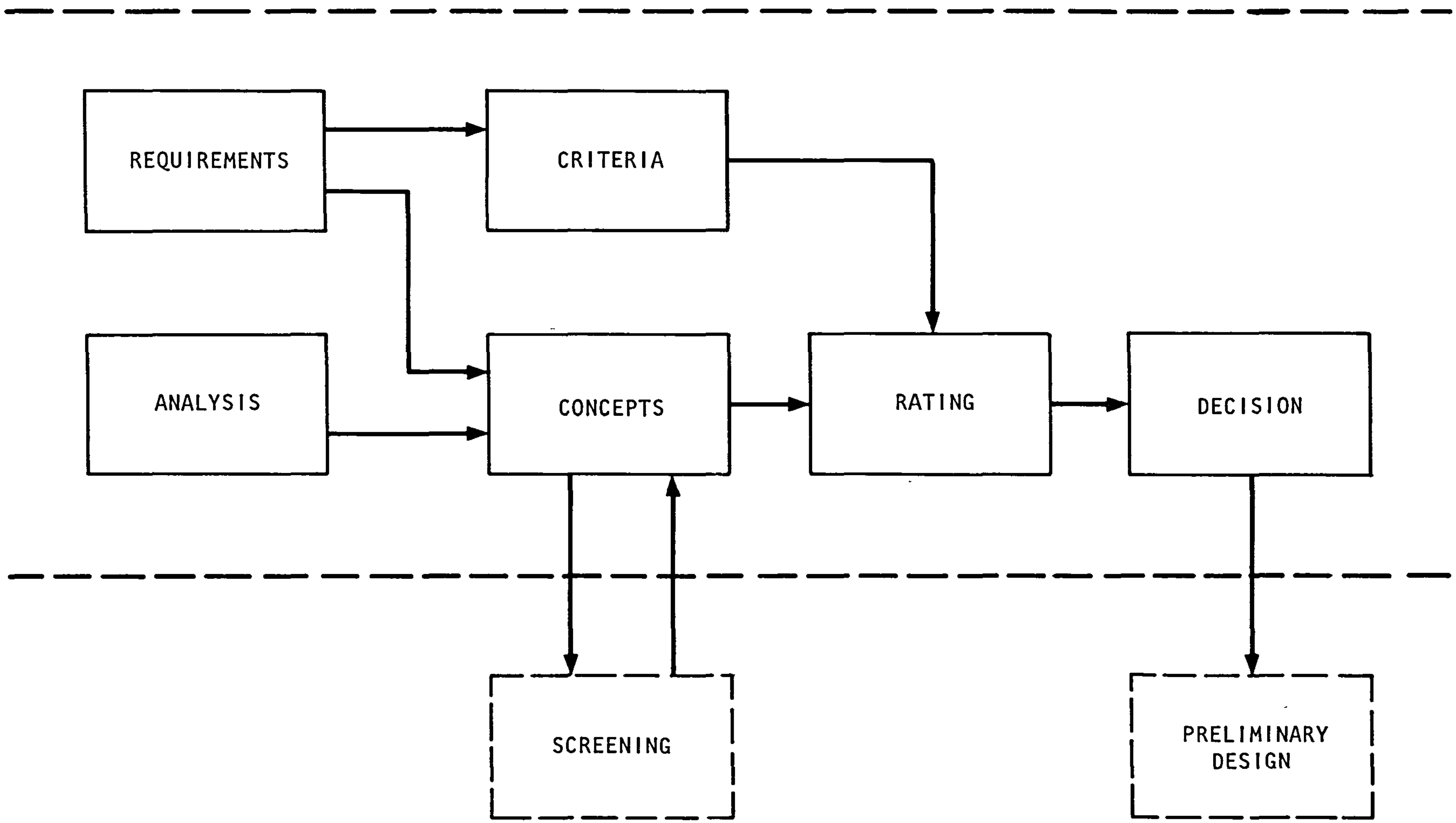
TASK V
REPORTING



Rocketdyne
North American Rockwell

Task I effort will be conducted as shown. Initial analysis during a trade study resulted in preliminary conceptual sketches of sealing closure designs to be tested in Task IIIA. The concepts have been approved by NASA-LeRC. Data from Task IIIA will be used to modify the sealing concepts as required for a final decision on sealing closure and actuator concepts for Task II valve preliminary design.

TASK I
VALVE SUBCOMPONENT ANALYSIS AND CONCEPTUAL DESIGN

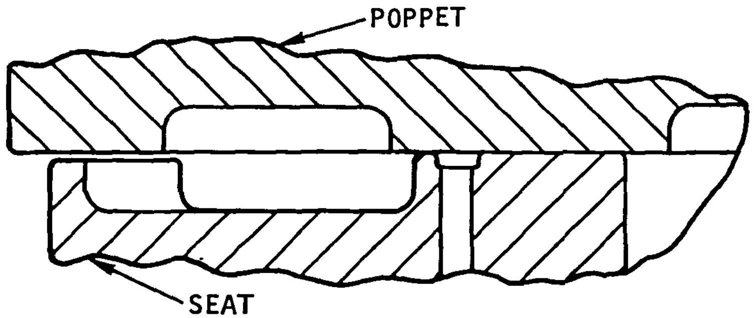


The flat hard poppet on flat hard seat is the most basic type to be evaluated and is one for which considerable data have been accumulated in past programs. It will provide an ideal frame of reference for additional investigations of closure abnormalities.

The flat hard poppet on flat soft seat is also a basic, previously evaluated type, and will provide a logical set of data from which to extend further investigations.

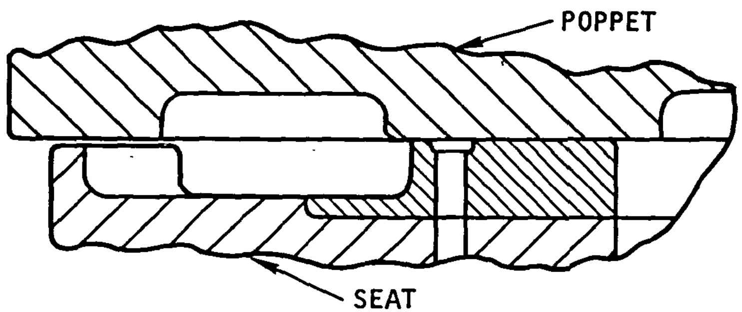
The flat hard poppet on grooved soft seat is a concept which extends the contaminant tolerance capabilities of the soft seating concept. Severe damage to one or even several of the lands may not seriously impair sealing capability.

CLOSURE CONCEPTS



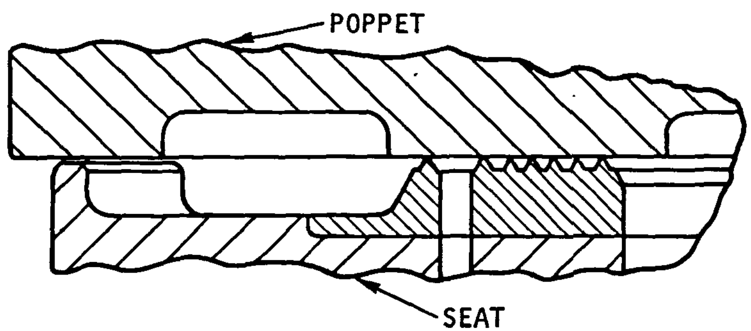
CONCEPT 1 FLAT HARD POPPET ON FLAT HARD SEAT

- EVALUATED IN AFRPL CONTRACTS AF04(611)-8392 & AF04(611)-9712
- CONTAMINANT SENSITIVE



CONCEPT 2 FLAT HARD POPPET ON FLAT SOFT SEAT

- EVALUATED IN AFRPL CONTRACT F04611-67-C-0085
- SOMEWHAT CONTAMINANT TOLERANT



CONCEPT 3 FLAT HARD POPPET ON GROOVED SOFT SEAT

- EVALUATED IN AFRPL CONTRACT F04611-67-0085
- VERY CONTAMINANT TOLERANT

AA = Arithmetic Average, microinches, which supersedes RMS (Root Mean Square) terminology. A 0.7 AA surface (0.7 microinches) represents a roughness of approximately 2.1 microinches peak-to-valley average surface imperfection dimension.

275-845
3-71

0.7AA
UNIDIRECTIONAL
DIAMOND LAPPED
SURFACE
500X

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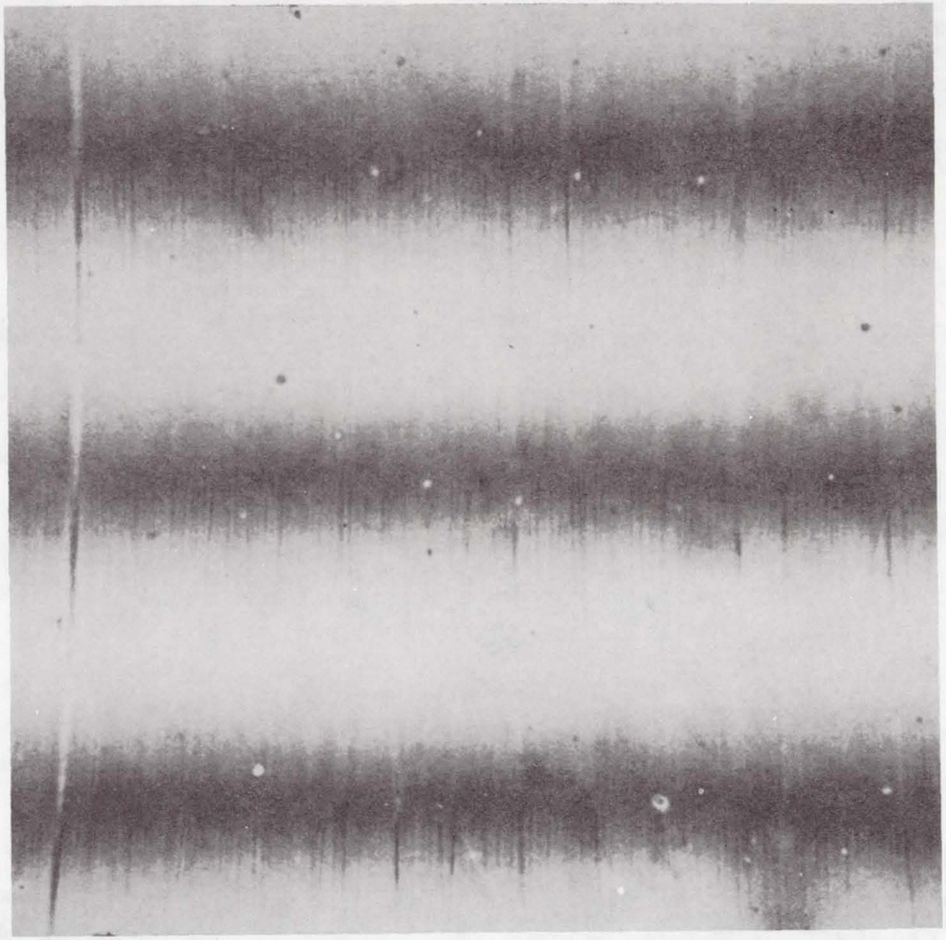
275-846
3-71



0.3AA
UNIDIRECTIONAL
DIAMOND LAPPED
SURFACE
500X

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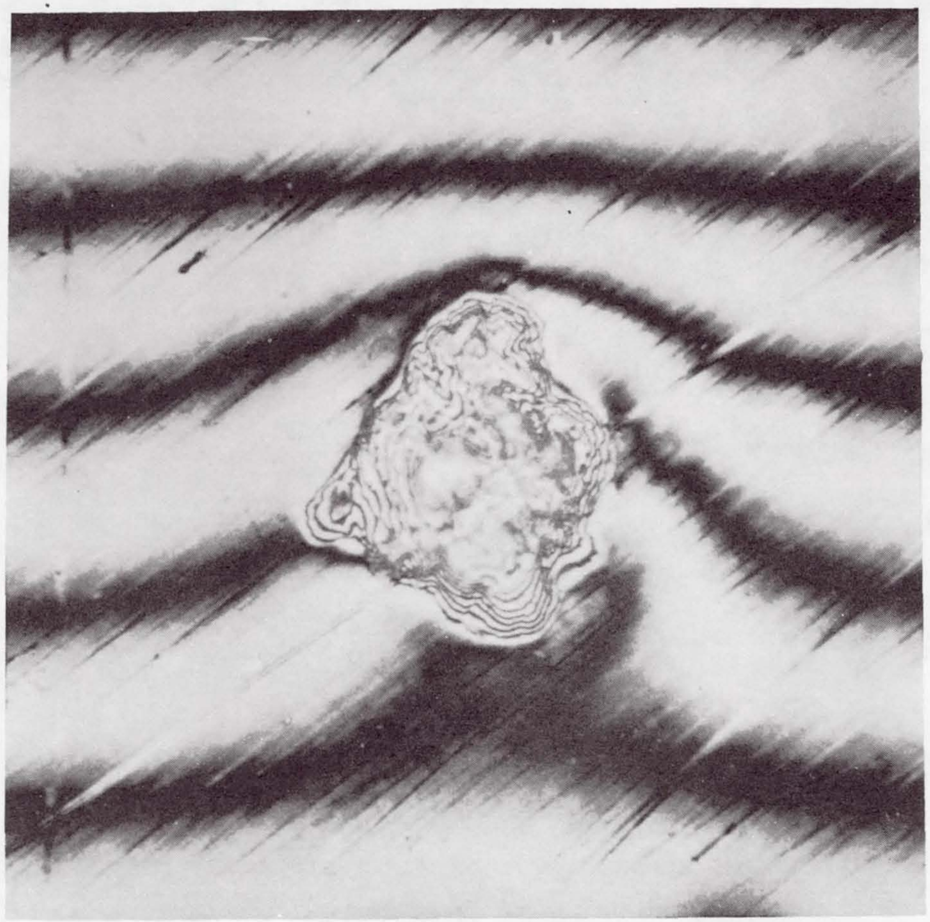
North American Rockwell

Seat material adjacent to the 30 micron (approximately 0.0012 inch diameter) particle impact area has been displaced above the normal seating plane. To maintain sealing integrity in a valve closure with such damage, sufficient force margin must be available to elastically depress the raised material and bring the surfaces into intimate contact.



DEPRESSION

FROM

30 MICRON

HARD PARTICLE

IN

FLAT 440C SEAT

500X

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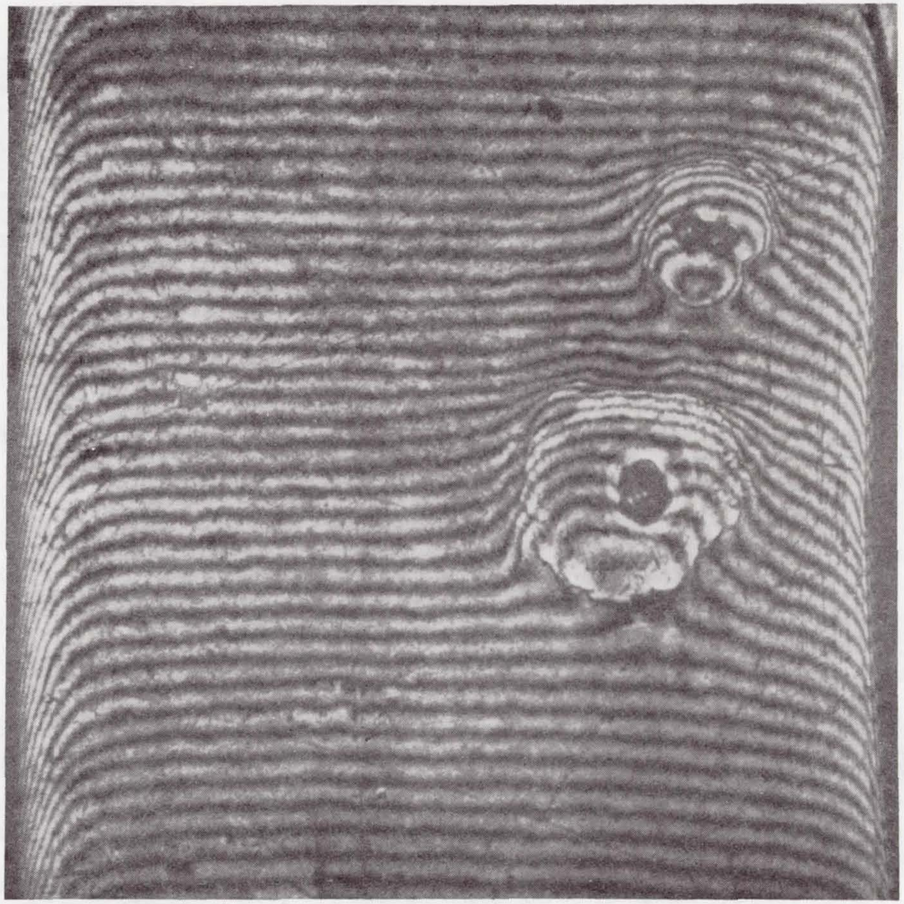
The displaced seat material is more pronounced in a contaminated soft seat since the particles have remained intact and embedded.

275-848
3-71



EMBEDDED HARD 30 MICRON
AND
SMALLER PARTICLES
IN
FLAT COPPER SEAT
100X

585

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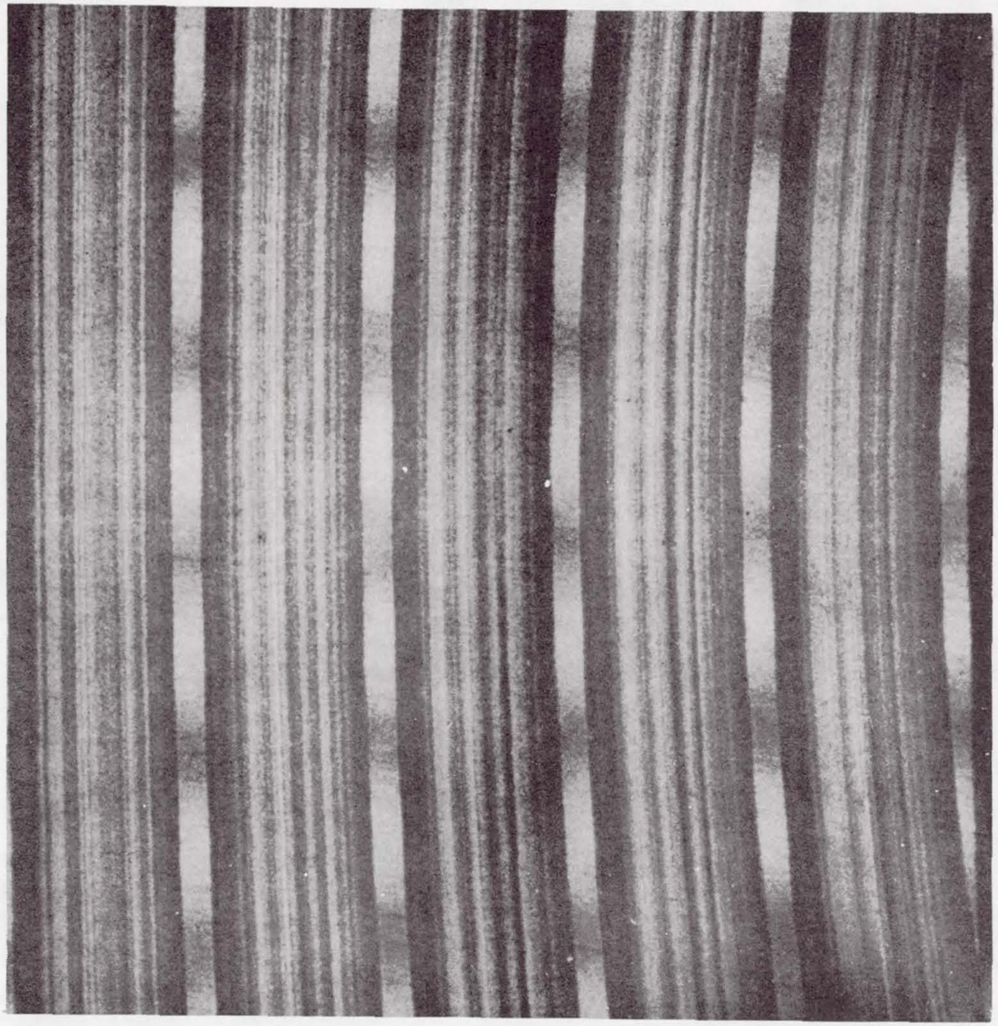
Grooved seat land design provides multiple sealing lands and voids for displaced material in the event of contaminant encounters. Land and groove widths are typically 0.001 inch on a 0.006 inch pitch with a groove depth of 0.001 inch. As machined dimensions held to ±50 micro-inches tolerance; sealing lands are lapped to final finish.

275-849
3-71



GROOVED COPPER SEAT

LANDS

100X

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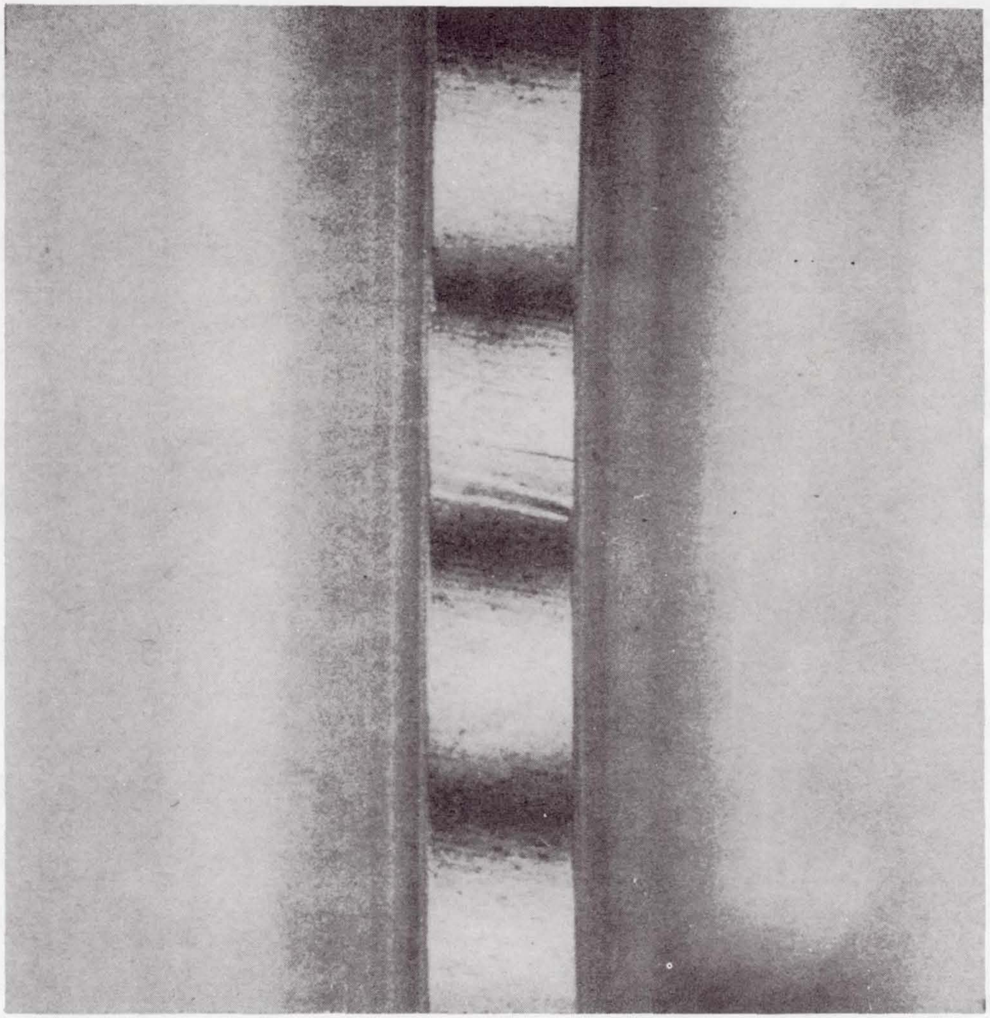
Individual land detail indicating geometrical precision and surface finish on the order of 0.2 microinch AA.

T275-850

275-850
3-71



GROOVED COPPER SEAT

LAND

500X

589

Particle and land material displaced laterally leaving no raised metal. Damage bridges this land sealing area; however, adjacent lands maintain integrity for leakage control.

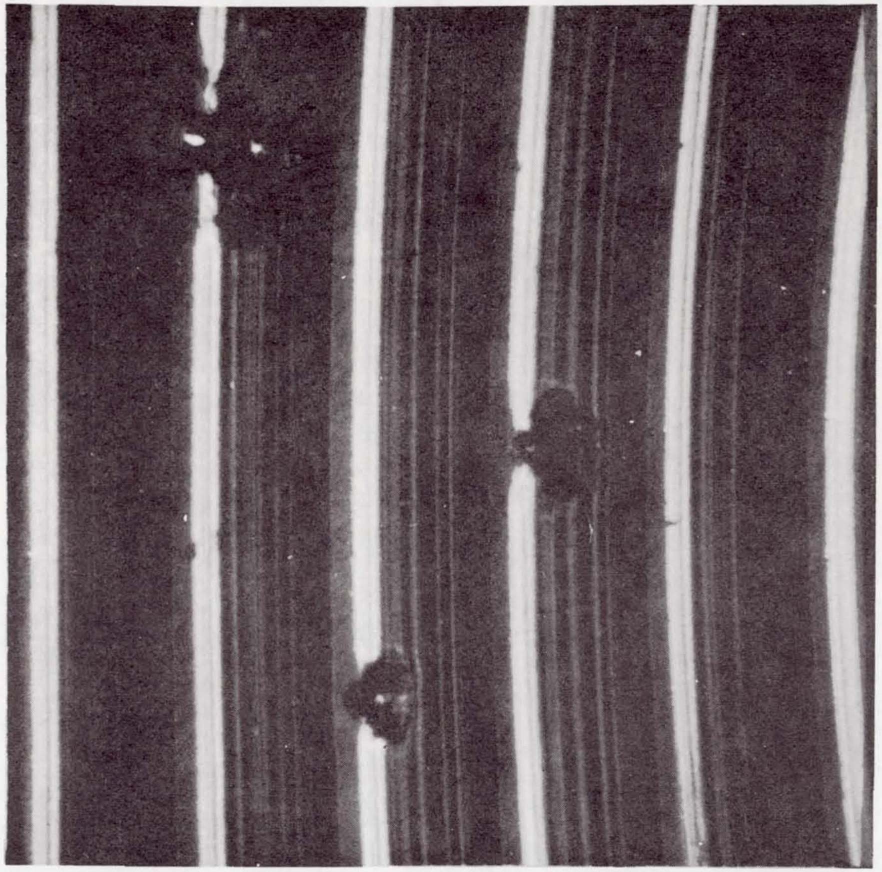
T275-851

EMBEDDED 30 MICRON SPHERICAL PARTICLE IN GROOVED COPPER SEAT LAND, NO RAISED METAL 500X

591

Typical view of particle entrapments resulting from cycling with a controlled particulate population in the fluid stream. The six lands accumulated 21 particle hits, none of which bridged the sealing surfaces.

T275-864



GROOVED COPPER SEAT WITH
EMBEDDED 30 AND 60 MICRON
SPHERICAL PARTICLES
100X

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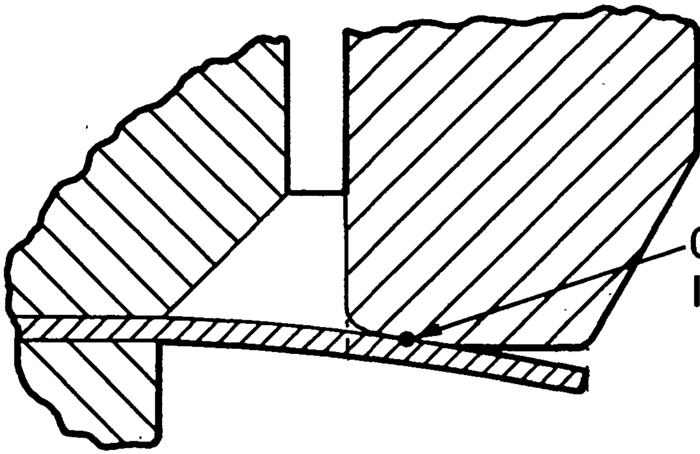
This configuration represents a concept which has been reduced to flight engine practice but without benefit of specific detail closure parametric investigations. It currently is used on J-2 engine pneumatic packages in a 4-way solenoid valve operating at cryogenic temperatures and 400 psig (276 newtons/cm^2). The current materials have exhibited a tendency toward scrubbing wear which may be alleviated by the utilization of disc or seat plating-coatings.

CLOSURE CONCEPTS



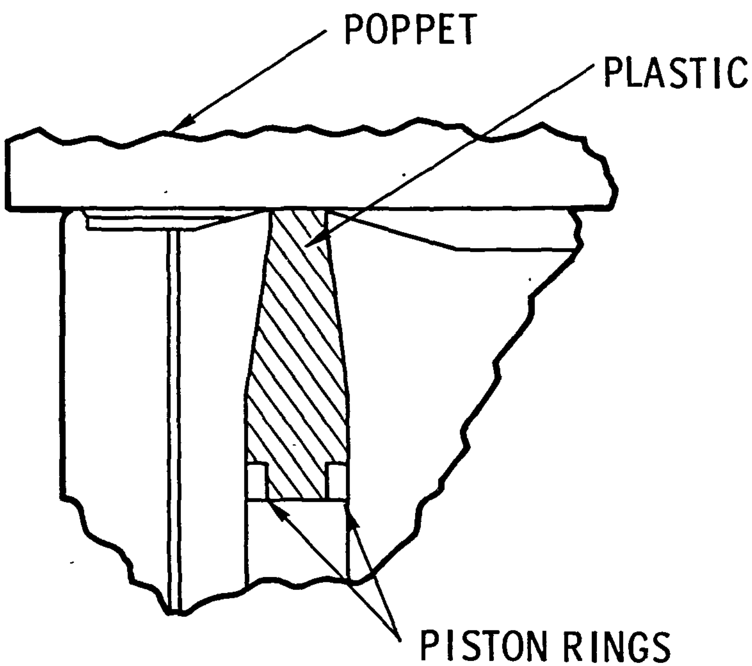
- USED ON J-2 ENGINE
- REDUCES INTERFACIAL ALIGNMENT REQUIREMENTS

CONCEPT 4 FLEXIBLE DISC SEAL

595

This closure concept utilizes entrapped fluoroplastic as the sealing element which is loaded by valve closing action such that it plastically flows to mate with opposing sealing surface imperfections. The design requires approximately 1.7 times more seating load than the previously discussed metal-to-metal closures for a given seat bearing stress. This feature, however, may be offset by potentially improved sealing capabilities occasioned by plastic seating surface conformation. In this case, seat load could be reduced. This concept has been reduced to practice on facility valves but currently lacks comprehensive analytical and experimental investigation information. Performance data, however, are encouraging. Upper operating temperature limit is undetermined.

CLOSURE CONCEPTS



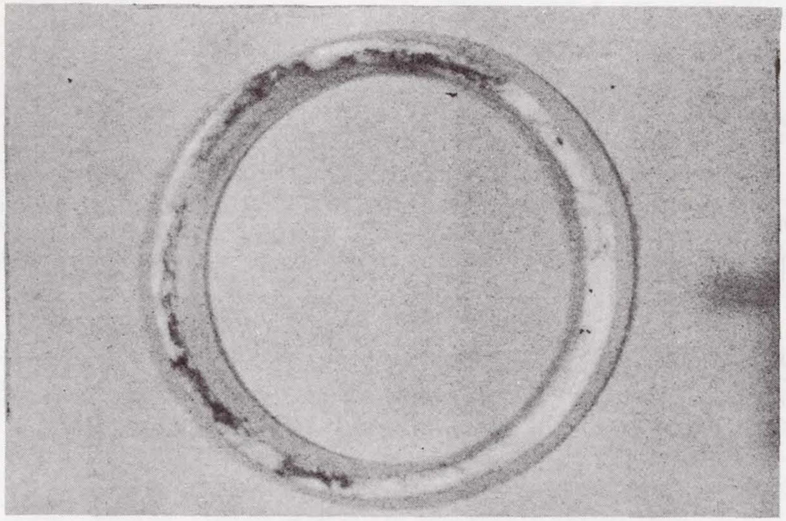
CONCEPT 5 CAPTIVE PLASTIC SEAL

- USED IN ROCKETDYNE FACILITY VALVES
- VERY CONTAMINANT TOLERANT

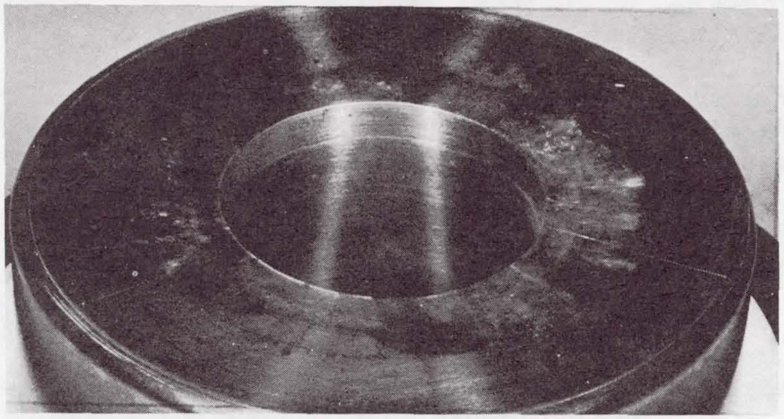
These photos show the captive plastic insert and mating seat from a 4" diameter facility valve. The seat was purposely damaged, and the valve then cycled with gross contaminant particles placed on the seat. Following cycling, the valve was flowed to flush the seat area, then leak tested. Using a water displacement buret, no leakage was evident with 3600 psig GN_2 after several minutes of observation.

T275-865

CLOSURE COMPONENTS



CAPTIVE PLASTIC INSERT-CYCLED WITH GROSS CONTAMINANTS



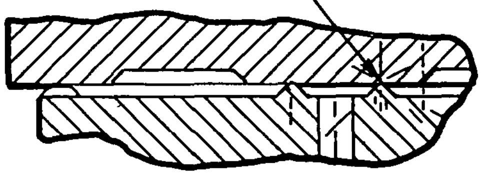
MATING SEAT-PURPOSELY DAMAGED BY SCRIBING, CYCLED WITH GROSS CONTAMINANTS

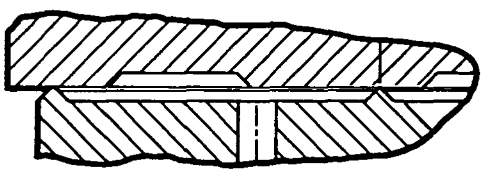
NO LEAKAGE DETECTED WITH WATER DISPLACEMENT BURET { 3600 PSIG GN_2 (AMB.)

In this configuration the poppet will be fabricated from hard carbide (R_A 90 to 94) while the seat will be "soft" carbide (R_A 84 to 86). In both cases, carbide inserts will be brazed to CRES poppet and seat blanks. The seat land will be ground and lapped to a sharp top, then plastically formed with a hard carbide platen to form a seating land crown with less than .0005 inch (0.00127 cm) radius, and flat within 5 microinches (0.0000127 cm) total. This concept represents a contaminant "resistant" approach as opposed to the contaminant "tolerant" intent of the soft-seated configurations. The extremely small radius of the seat land crown greatly reduces the probability of a particle being trapped on its top; however, in that event, the small radius and hard nature of the closure poppet and seat would cause contaminant shearing or fracture and permit seating in a normal manner. This configuration has not been previously evaluated. It offers, however, the combined potential advantages of contamination and scrubbing wear resistance.

CLOSURE CONCEPTS

CLOSURE INTERFACE





ALTERNATE

CONCEPT 6 HARD-SHARP CARBIDE SEAL

- UNIQUE CONCEPT
- CONTAMINANT TOLERANT
- CONTAMINANT DESTROYER

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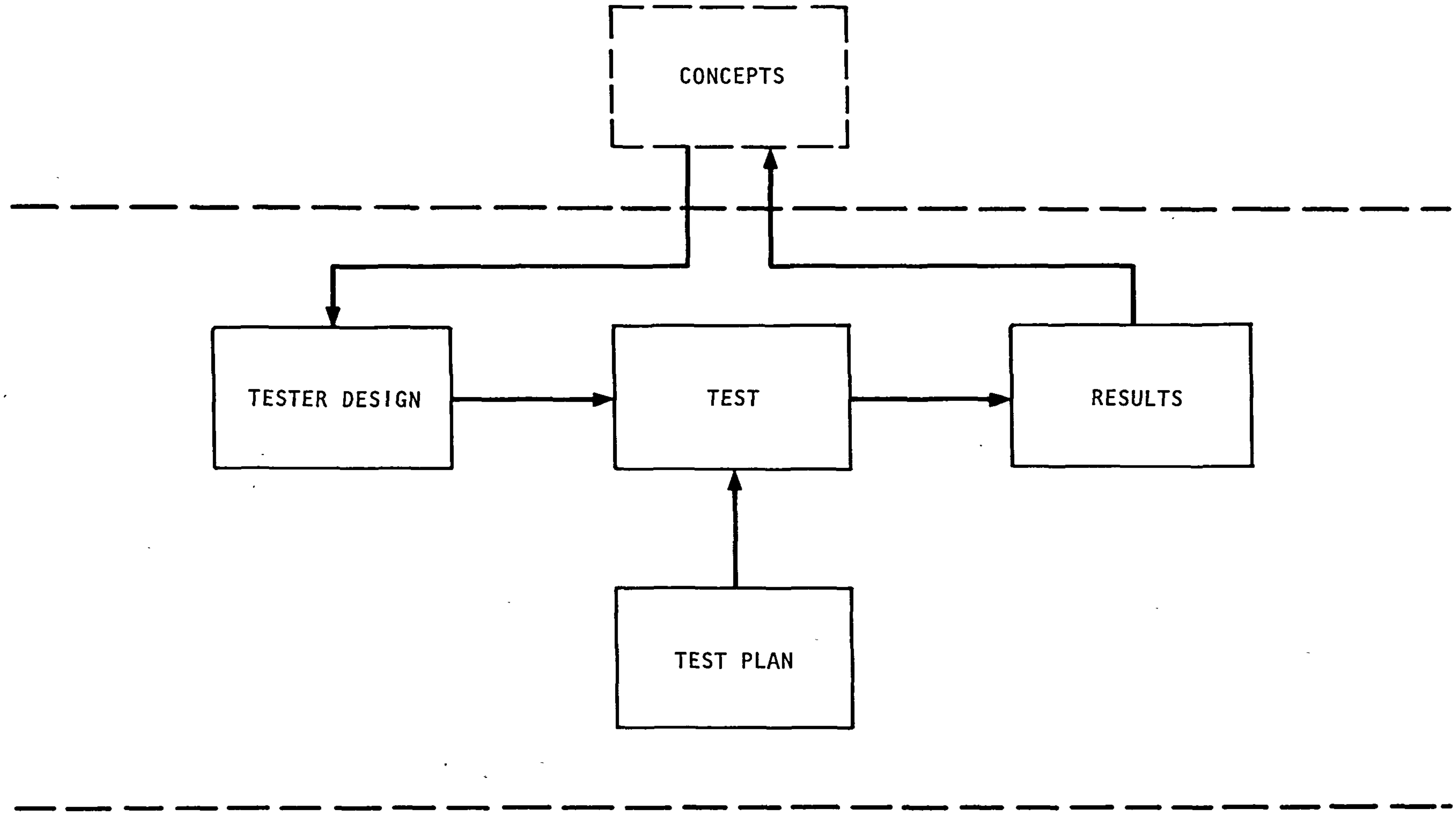
Task IIIA effort will be conducted as shown. A closure screening tester will be used to test the closure concepts developed in Task I. Data from these tests will be used to modify the concepts for Task I decisions on Valve Preliminary Designs.

TASK III A
SEALING CLOSURE SCREENING

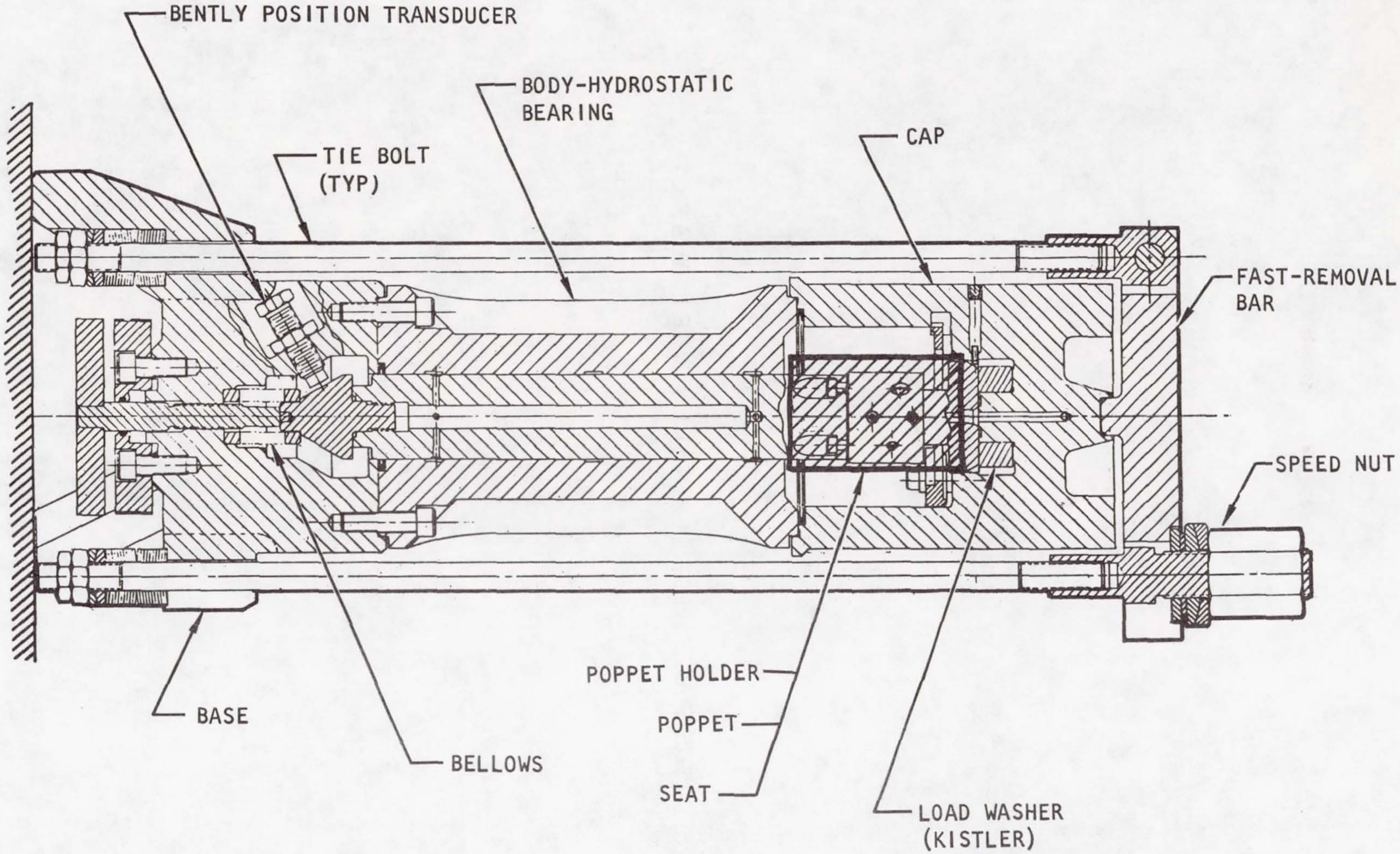
275-837
3-71



603

The closure tester is a multiple feature fixture providing for closure mounting, loading, and leakage collection together with integral instrumentation for monitoring fundamental closure parameters. The basic structure consists of a pneumatically pressurized hydrostatic bearing (body and piston) which permits near frictionless loading of closure test surfaces via pressure application to one end of the piston. The test closure poppet is retained at the opposite end of the piston. The mating test closure seat is located in a removable cap through which closure inlet pressure is directed and resulting leakage is captured. The tester is designed to operate in both static and dynamic modes. In the static application, tester function is to provide a means of positioning a closure poppet and seat in intimate contact with supply pressure applied while incrementally varying closure interface load and monitoring leakage variation as a function of that load. In the dynamic mode, impact loads are applied to closure interfaces to simulate cyclic operating conditions found in actual valves.

CLOSURE SCREENING TESTER



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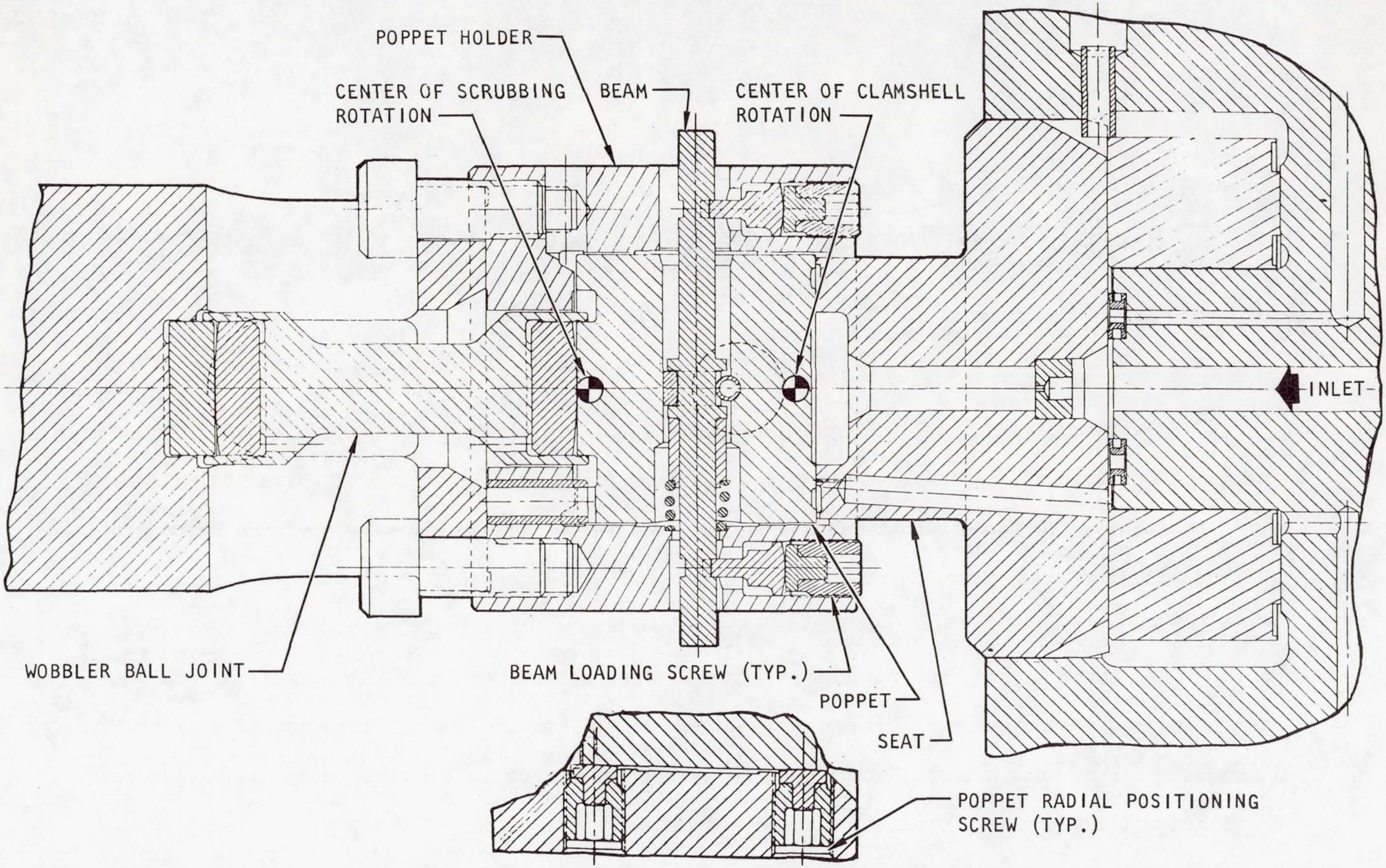
TESTER FEATURES

- PRECISION LOAD CONTROL (STATIC AND DYNAMIC)
- PRECISION POSITIONING CAPABILITY
- FRICTIONLESS ACTUATION
- SEAT-POPPET ACCESS IN LESS THAN ONE MINUTE
- UNLIMITED CYCLE LIFE
- 140 R TO 850 R OPERATION
- MEASURE INTERFACIAL ENERGY

This device is the heart of the tester. The holder positions the poppet on the wobbler ball joint located within the piston. Poppet radial position is controlled with 8 carbide-button tipped screws located near each end of the poppet at 45 degrees from the beam axis. A beam loads the poppet onto the wobbler ball joint via two end bearings and one center bearing. The end bearings are axially adjustable to vary beam deflection, and thereby load. The articulation mechanism provides for three modes of closure. Clamped (parallel) closure will be effected by removal of the wobbler ball joint. The other modes are clamshell or scrubbing depending upon the settings of the radial positioning screws. These modes of closure, in combination with the ability to precisely measure interfacial energy, will enable the development of design criteria to successfully design and fabricate the APS Propellant Valves.

POPPET ARTICULATION DETAIL



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POPPET ARTICULATION FEATURES

- THREE CLOSURE MODES
  - PARALLEL (CLAMPED)
  - CLAMSHELL
  - SCRUBBING

  } CONTROL INTERFACIAL MOVEMENT
- ULTRA-CLEAN INLET DESIGN
- UP TO 3000 PSIG INLET CAPABILITY
- LOW LEAKAGE MEASUREMENT

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"IGNITION DEVICES FOR ACPS"

S. D. ROSENBERG

AEROJET

TECHNICAL MANAGER

E. A. EDELMAN

LEWIS RESEARCH CENTER

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CONTRACT NAS 3-14348

IGNITION SYSTEM FOR SPACE SHUTTLE AUXILIARY PROPULSION SYSTEM

For

NASA-LEWIS RESEARCH CENTER

ALRC Project Manager: S. D. Rosenberg (Speaker)
ALRC Project Engineer: A. J. Aitken
NASA Project Manager: E. A. Edelman

AEROJET LIQUID ROCKET COMPANY
SACRAMENTO, CALIFORNIA • A DIVISION OF AEROJET-GENERAL

The objectives of this program are to obtain basic ignition data on spark and plasma torch ignition methods at operating conditions typical of the Space Shuttle Auxiliary Propulsion System and to complete a preliminary design of an optimum ignition system based upon the test results.

The work was comprised of: an analytical and experimental program of design, fabrication and testing of gaseous hydrogen-gaseous oxygen ignition system with data obtained for wide ranges of operating conditions; and the completion of a preliminary design of an ignition system.

PROGRAM OBJECTIVES

NAS3-14348

(1) OBTAIN BASIC IGNITION DATA ON SPARK AND PLASMA TORCH IGNITION METHODS AT SPACE SHUTTLE APS OPERATING CONDITIONS.

(2) COMPLETE PRELIMINARY DESIGN OF TWO IGNITION SYSTEMS BASED ON THE BASIC IGNITION DATA.

AEROJET LIQUID ROCKET COMPANY
SACRAMENTO, CALIFORNIA ● A DIVISION OF AEROJET-GENERAL

AGCS 0100-71(REV 5/70)

The test variables which were investigated in the Igniter-Only operational mode included spark (plasma) energy and gap, O_2/H_2 core and total mixture ratio, igniter torch chamber pressure, propellant and igniter temperature at ignition, and hot fire duration at steady-state and pulse conditions.

The test variables which were investigated in the Igniter-Complete Thruster operational mode included spark (plasma) energy and gap, igniter thrust level (25- and 50-lbF nominal), propellant temperature at ignition, main propellant lead/lag, and thruster chamber pressure at ignition and steady state.

In addition, the electrode durability of the spark and plasma plugs was determined and was found to be acceptable over a range of approximately 10^5 simulated ignitions.

TEST PROGRAM VARIABLES

NAS3-14348

TEST VARIABLES	SPARK IGNITER	PLASMA IGNITER
GAP, in.	0.025 TO 0.100	0.025 TO 0.040
ENERGY AT GAP, mj	1 TO 10	0.15 TO 100
CORE MR	35 TO 80	40 TO 75
TOTAL MR	6 TO 12	6 TO 11
P_c, psia	10-20, 100-500	10-20, 100-500
OXIDIZER TEMPERATURE, °R	200 TO 500	190 TO 500
FUEL TEMPERATURE, °R	150 TO 500	135 TO 500
IGNITER BODY TEMPERATURE, °R	175 TO 500	140 TO 500
STEADY-STATE DURATION, sec	0.1 TO 151	0.1 TO 10
PULSE OPERATION	1 TO 1000	1 TO 876
ELECTRODE DURABILITY	10^6 SPARKS	10^6 PULSES
FUEL LEAD/LAG, msec	+30 TO -30	+30 TO -30
BACK PRESSURE AT FS_1	2×10^{-4} mm TO 14.7 psia	0.1 TO 14.7 psia

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The test results of primary interest which were obtained as a function of the test variables were igniter ignition delay and response time, thruster ignition delay (time to 90% of P_c), thruster ignition pressure and the chamber overpressure resulting from ignition, igniter durability in Igniter-Only and Igniter-Complete Thruster operational modes and igniter compatibility with the thruster in the Igniter-Only operational mode. Igniter durability and compatibility with the thruster was found to be excellent under all conditions evaluated.

KEY TEST PROGRAM MERIT FACTORS

NAS3-14348

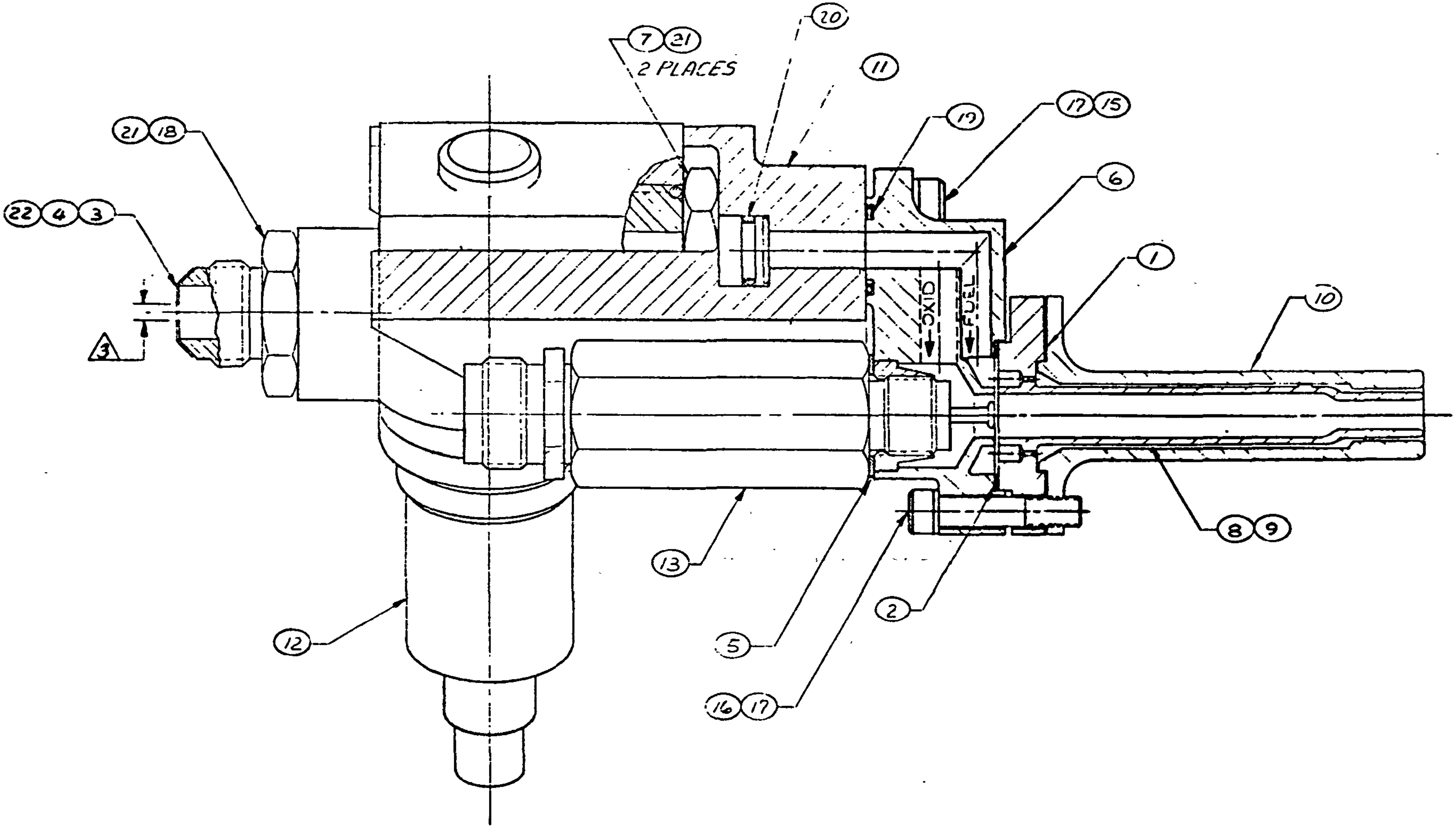
(1) IGNITER AND THRUSTER IGNITION DELAYS

(2) THRUSTER IGNITION PRESSURE AND OVERPRESSURE

(3) IGNITER DURABILITY AND COMPATIBILITY WITH THRUSTER

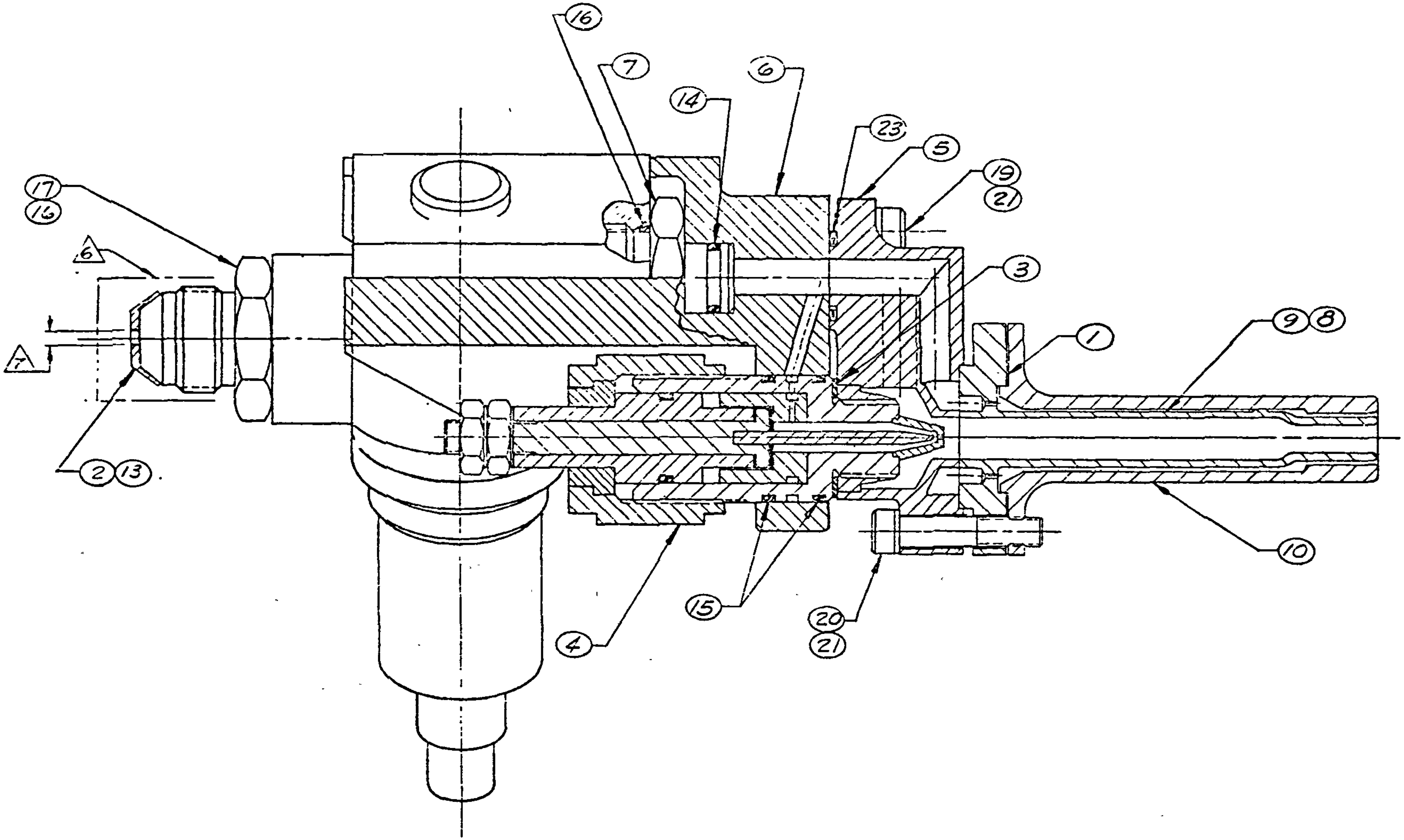
All of the O_2 flows through the spark gap where it is activated. The core H_2 is impinged (45°) on the activated O_2 just downstream of the spark plug (core MR $\sim$ 45). Most of the H_2 ($\sim$85%) is used to cool the torch chamber wall (total MR $\sim$ 6.5). The cooling H_2 reacts with the core exhaust at the exit of the igniter, providing a very hot torch for thruster ignition.

SPARK IGNITER FOR THE HIGH CHAMBER PRESSURE DESIGN POINT



The core H_2 flows through the plasma gap where it is activated. All of the O_2 is mixed (coaxially) with the activated H_2 just downstream of the plasma plug (core MR $\sim$ 45). Most of the H_2 ($\sim$85%) is used to cool the torch chamber wall (total MR $\sim$ 6.5). The cooling H_2 reacts with the core exhaust at the exit of the igniter, providing a very hot torch for thruster ignition.

PLASMA PULSE IGNITER FOR THE HIGH CHAMBER PRESSURE DESIGN POINT



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No significant change in igniter ignition delay was caused by varying the torch chamber pressure at fire switch 1.

IGNITER-ONLY TESTS - HIGH P_C SPARK: AMBIENT PRESSURE EFFECTS

NAS3-14348

AMBIENT PRESSURE AT FS1 PSIA	MM Hg	IGNITER IGNITION DELAY MSEC
14.7		10
0.3		10
0.1	5.17	10
	2.74×10^{-1}	6
	1.30×10^{-2}	5
	1.26×10^{-3}	9
	2.10×10^{-4}	11

NOTE: IGNITION DELAY IS DEFINED AS THE TIME FROM PRESSURE RESPONSE IN THE LAGGING PROPELLANT MANIFOLD TO 90% OF FULL P_C

No significant change in igniter ignition delay was noted over the range of spark gap and energy and P_c tested. Prompt ignition was obtained, i.e., 5 msec, under worst conditions tested, i.e., gap, 0.100 in.; energy, 1 mj; P_c, 505 psia. Ignition delay varied between 4 and 11 millisec under the conditions tested, demonstrating excellent response.

IGNITER-ONLY TESTS - HIGH P_C SPARK: AMBIENT TEMPERATURE PROPELLANTS

NAS3-14348

	TEST CONDITIONS			TEST RESULTS			
					MIXTURE RATIO		IGNITION DELAY
TEST	P_C PSIA	GAP IN.	ENERGY MJ	P_C PSIA	CORE	TOTAL	MSEC
110	300	0.050	10	298	38.8	5.6	8
123	300	0.050	10	302	45.0	6.5	9
126	300	0.025	10	295	45.0	6.5	7
129	300	0.100	10	297	45.0	6.5	11
132	100	0.100	10	97	40.0	6.0	8
135	500	0.100	10	476	40.5	6.7	9
138	300	0.100	5	298	45.0	6.5	9
141	100	0.100	5	97	40.0	6.0	11
144	500	0.100	5	481	40.7	6.5	9
147	300	0.025	5	297	45.0	6.5	8
150	300	0.025	1	297	45.0	6.5	8
153	300	0.050	1	299	45.0	6.5	4
156	300	0.100	1	300	45.0	6.5	6
159	100	0.100	1	98	40.0	6.0	4
185	500	0.100	1	505	40.7	6.6	5

NOTES

TEST DURATION, 150 MSEC

THRUST, 25 lbF (NOM)

SPARK RATE, 500 SPS

P_C AT FS_1, 0.3 ± 0.2 PSIA

The spark igniter demonstrated excellent durability under all conditions tested.

IGNITER-ONLY TESTS - HIGH P_C SPARK: PULSE AND DURABILITY

NAS3-14348

	TEST CONDITIONS			TEST RESULTS				
					MIXTURE RATIO			
TEST	P_C PSIA	GAP IN.	ENERGY MJ	P_C PSIA	CORE	TOTAL	IGNITION DELAY MSEC	NOTES
194	300	0.050	10	~303	45	6.5	~6	1000 CONSECUTIVE PULSE TEST 200 MSEC ON/300 MSEC OFF TOTAL ON TIME, 200 SEC
199	300	0.050	10	301	45	6.5	4	TEST DURATION, 10 SEC
409	300	0.050	1	300	45	6.5	-	TEST DURATION, 151 SEC

Igniter response remained the same as the temperature of the propellants was reduced from 500°R to below 300°R, i.e., 7 to 9 millisec.

IGNITER-ONLY TESTS - HIGH P_C SPARK: LOW TEMPERATURE PROPELLANTS

NAS3-14348

	TEST CONDITIONS			TEST RESULTS					
	P_C	GAP	ENERGY	P_C	MIXTURE RATIO		IGNITION DELAY	TEMP °R	
TEST	PSIA	IN.	MJ	PSIA	CORE	TOTAL	MSEC	OX	FU
202	500	0.050	10	530	39.2	5.8	9	374	283
205	500	0.050	5	533	39.7	5.9	8	390	290
208	500	0.050	1	531	38.8	5.7	8	390	276
212	500	0.100	1	533	38.8	5.9	8	390	277
220	300	0.100	1	319	39.6	5.9	7	307	255
224	300	0.100	1	310	40.4	6.1	8	297	258
227	100	0.100	1	111	35.6	5.3	9	299	260

Note that the fast response igniter valves had to be removed from the test apparatus to obtain the very low propellant temperature. Igniter response remained the same as the temperature was reduced from 500°R to below 200°R. Note that ignition was achieved at very high core MR, i.e., 81.2, demonstrating the insensitivity of the igniter to varying inlet conditions.

IGNITER-ONLY TESTS - HIGH P_C SPARK: VERY LOW TEMPERATURE PROPELLANTS

NAS 3-14348

	TEST CONDITIONS			TEST RESULTS					
					MIXTURE RATIO		IGNITION DELAY	TEMP °R	
TEST	P_C PSIA	GAP IN.	ENERGY MJ	P_C PSIA	CORE	TOTAL	MSEC	OX	FU
403	300	0.100	1	300	45.0	6.5	37	505	503
412	300	0.100	1	199	40.0	6.0	43	280	150
414	~200	0.100	1	159	56.0	8.1	37	280	175
415	~175	0.100	1	148	81.2	12.2	40	280	175
421	100	0.100	1	91	45.0	6.8	38	200	173

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No significant change in igniter ignition delay was noted over the conditions tested. Prompt ignition was obtained, i.e., 8 millisec.

IGNITER-ONLY TESTS - HIGH P_C PLASMA PULSE: AMBIENT TEMPERATURE PROPELLANTS

NAS3-14348

	TEST CONDITIONS			TEST RESULTS				
TEST	P_C PSIA	GAP IN.	ENERGY MJ	P_C PSIA	MIXTURE RATIO CORE	MIXTURE RATIO TOTAL	IGNITION DELAY MSEC	NOTES
308	300	0.030	~0.2	302	41.6	6.6	11	TEST DURATION, 150 MSEC
311	100	0.040	~0.2	101	41.5	5.8	10	THRUST, 25 lbF (NOM)
316	300	0.040	~0.2	297	41.6	6.6	8	PULSE RATE
319	500	0.040	~0.2	488	41.2	6.7	8	P_C AT FS_1, 0.4 ± 0.1 PSIA
								ENERGY, 0.15 TO 0.5 MJ

The plasma pulse igniter demonstrated
excellent durability under all conditions tested.

IGNITER-ONLY TESTS - HIGH P_C PLASMA PULSE: PULSE AND DURABILITY

NAS3-14348

	TEST CONDITIONS			TEST RESULTS				
TEST	P_C PSIA	GAP IN.	ENERGY MJ	P_C PSIA	MIXTURE RATIO CORE	MIXTURE RATIO TOTAL	IGNITION DELAY MSEC	NOTES
322	300	0.040	~0.2	~297	41.6	6.6	~12	876 CONSECUTIVE PULSE TEST 100 MSEC ON/200 MSEC OFF TOTAL ON TIME, 87.6 SEC
320	300	0.040	~0.2	297	41.6	6.6	11	TEST DURATION, 10 SEC

Igniter response remained the same as the temperature of the propellants was reduced from 500°R to below 300°R, i.e., 9 to 11 millisec. However, the plasma gap had to be reduced from 0.040 to 0.030 in. to obtain reliable ignition as the temperature was reduced.

IGNITER-ONLY TESTS - HIGH P_C PLASMA PULSE: LOW TEMPERATURE PROPELLANTS

NAS 3-14348

	TEST CONDITIONS			TEST RESULTS					
					MIXTURE RATIO		IGNITION DELAY	TEMP °R	
TEST	P_C PSIA	GAP IN.	ENERGY MJ	P_C PSIA	CORE	TOTAL	MSEC	OX	FU
324	500	0.040	~0.2	485	44	6.6	9	418	327
325	500	0.040	~0.2	NO IGNITION			-		
326	500	0.040	~0.2	485	44	6.6	10	416	322
327	500	0.040	~0.2	NO IGNITION			-		
334	500	0.030	~0.2	499	44.7	6.5	11	377	258
342	300	0.030	~0.2	299	43.9	6.6	9	313	259
345	100	0.030	~0.2	95	37.2	5.2	9	301	259

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AGCS 0100-71 REV 5/70)

Note that the fast response igniter valves had to be removed from the test apparatus to obtain the very low propellant temperature. Igniter response remained the same as the temperature was reduced from 500°R to below 200°R. Note that ignition was achieved at very high core MR, i.e., 76.0, demonstrating the insensitivity of the igniter to varying inlet conditions.

IGNITER-ONLY TESTS - HIGH P_C PLASMA PULSE: VERY LOW TEMPERATURE PROPELLANTS

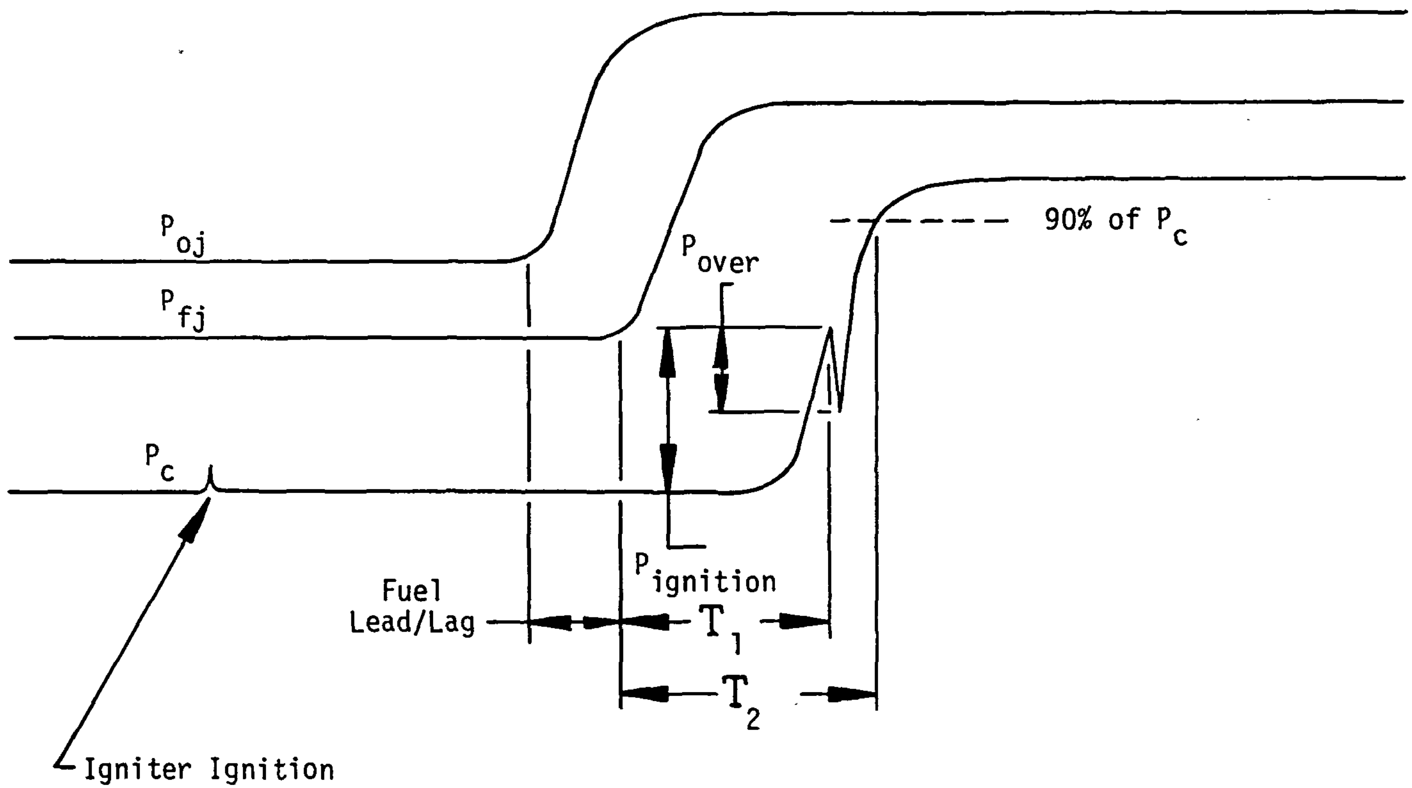
NAS 3-14348

	TEST CONDITIONS			TEST RESULTS					
					MIXTURE RATIO		IGNITION	TEMP °R	
TEST	P_C PSIA	GAP IN.	ENERGY MJ	P_C PSIA	CORE	TOTAL	DELAY MSEC	OX	FU
404	300	0.030	~0.2	300	45.0	6.5	36	503	504
425	300	0.030	~0.2	179	39.0	5.7	46	194	155
426	~200	0.030	~0.2	163	49.3	7.4	42	193	134
430	~125	0.030	~0.2	72	76.0	10.9	31	194	136
432	100	0.030	~0.2	66	45.0	6.8	37	193	145

The principal objective of these tests was to determine the effect of main propellant sequencing on thruster ignition delay, thruster ignition pressure, and ignition overpressure.

IGNITER-COMPLETE THRUSTER TESTS - DEFINITION OF START TRANSIENT TERMS

NAS3-14348



1. T_1 is defined as the Ignition Delay.
2. T_2 is the time required to reach 90% of full P_c.
3. $P_{ignition}$ is the absolute pressure obtained at ignition.
4. P_{over} is the pressure generated by ignition that is observed to be over and above the slope of P_c rise.
5. Effects of cold flow pressures are not shown and no time frame is intended.

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Simultaneous propellant sequencing
results in the optimum ignition conditions.

IGNITER-COMPLETE THRUSTER TESTS - HIGH P_C SPARK: AMBIENT TEMPERATURE PROPELLANTS

NAS3-14348

	THRUSTER TEST CONDITIONS			THRUSTER TEST RESULTS							FUEL LEAD/LAG MSEC	
TEST	P_C PSIA	MR	FUEL SEQUENCE MSEC	P_C PSIA	MR	THRUST lbF	τ_1 MSEC	τ_2 MSEC	P_{IGN} PSIA	P_{OVER} PSIA	LEAD	LAG
7-101-C	300	4.0	0	300	3.93	1196	4	11	50	0	3	
7-101-D	300	4.0	+20	300	3.93	1184	4	13	212	0	18	
7-101-E	300	4.0	+10	310	3.93	1189	5	16	100	0	11	
7-104-A	500	4.0	+10	450	4.03	1854	8	13	75	0	0	0
7-104-B	500	4.0	+10	450	4.03	1871	11	13	185	0	10	
7-104-D	500	4.0	-10	450	4.03	1853	9	12	500	290		14
8-101	300	4.0	-10	310	4.07	1220	3	15	270	120		10
8-102	300	3.0	-10	308	3.84	1151	3	11	260	40		12
8-103	500	5.0	-10	467	3.82	1837	7	16	435	275		14
8-105	500	3.0	0	489	2.93	1918	6	15	100	0	2	
8-110	100	4.0	0	99	4.04	292	2	7	150	20	3	
8-111	100	3.0	0	102	3.00	305	2	6	165	15		2
8-112	100	5.0	0	100	5.21	299	2	7	115	12		2

NOTES: SPARK ENERGY, GAP, RATE; 10 MJ, 0.050 IN., 500 SPS
IGNITER THRUST AND MR; 25 lbF, CORE MR-40, TOTAL MR-6.0
DURATION; IGNITER - 100 MSEC, THRUSTER - 0.5 TO 2.0 SEC

Simultaneous propellant sequencing
results in the optimum ignition conditions.

IGNITER-COMPLETE THRUSTER TESTS - HIGH P_C PLASMA PULSE: AMBIENT TEMPERATURE PROPELLANTS

NAS3-14348

	THRUSTER TEST CONDITIONS			THRUST TEST RESULTS							FUEL LEAD/LAG MSEC	
TEST	P_C PSIA	MR	FUEL SEQUENCE MSEC	P_C PSIA	MR	THRUST lbF	τ_1 MSEC	τ_2 MSEC	P_{IGN} PSIA	P_{OVER} PSIA	LEAD	LAG
7-108-D	300	4.0	-10	310	3.91	1174	4	9	320	85		12
7-108-F	300	4.0	+10	295	3.71	1182	4	16	265	60	16	
7-108-G	300	4.0	+20	300	3.77	1171	4	16	105	22	22	
7-108-H	300	4.0	-20	310	3.94	1161	3	8	300	75		17
8-155	300	4.0	0	316	3.98	1238	4	7	270	20		5
8-156	300	5.0	0	306	5.24	1262	5	8	275	18		3
8-157	300	5.0	0	308	5.00	1292	4	6	280	15		2
8-158	300	5.0	0	312	5.11	1194	4	9	262	10		4
8-152	100	5.0	0	105	4.21	307	2	6	58	0		3
8-153	100	5.0	0	106	5.55	313	3	5	30	15		3
8-154	100	5.0	0	113	6.89	294	3	1	42	25		4

NOTES: PLASMA PULSE ENERGY, GAP, RATE; 0.2 MJ, 0.030 IN., 220 PPS
IGNITER THRUST AND MR; 25 lbF, CORE MR - 41.5, TOTAL MR - 5.8
DURATION; IGNITER - 100 MSEC, THRUSTER - 0.4 TO 1.0 SEC

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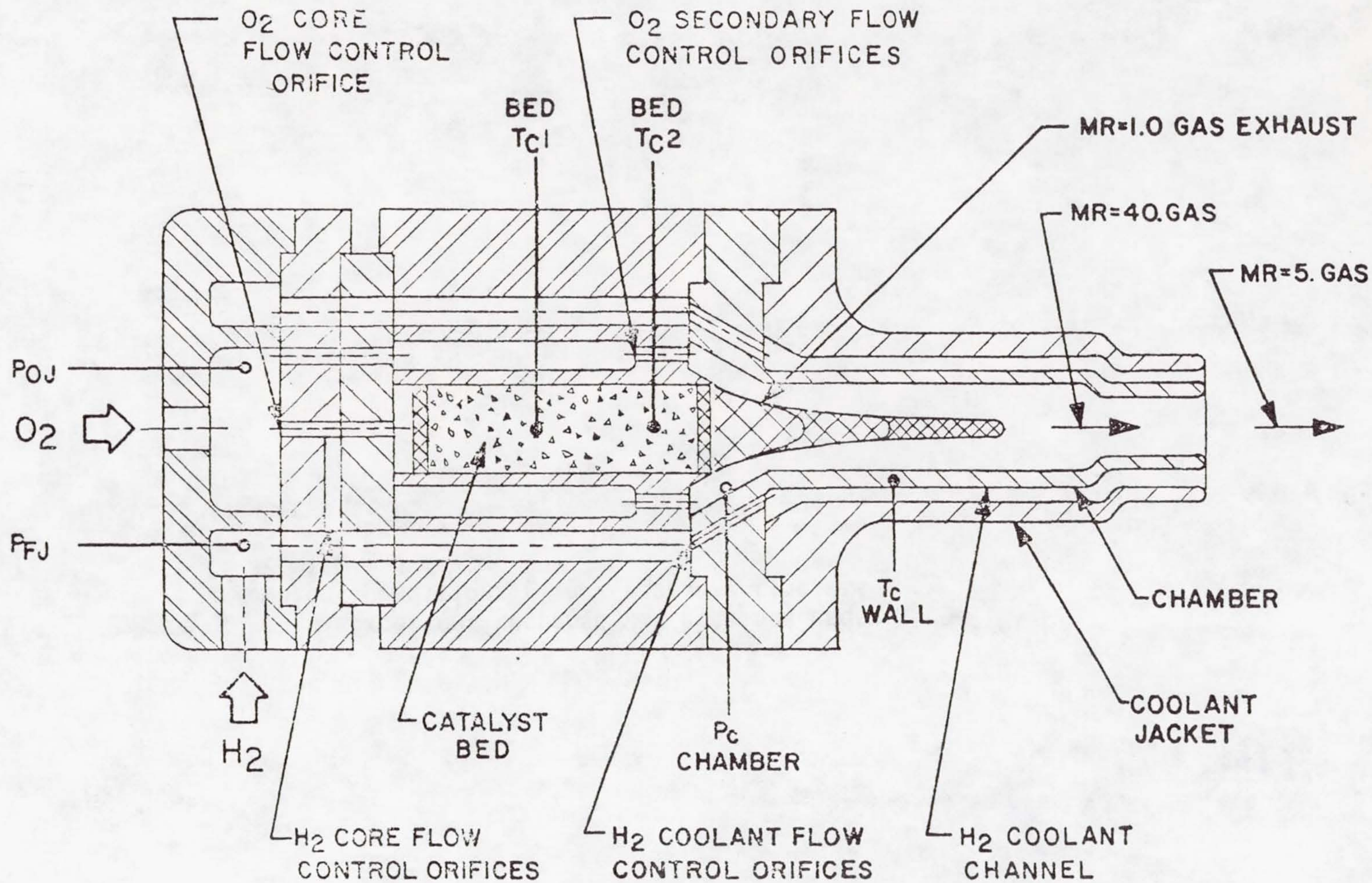
The O_2 flow is split with part being mixed with H_2 (MR = 1) just before entering the Shell 405 catalyst bed and most (~97%) being used to cool the catalyst bed wall prior to being injected into the torch chamber just downstream of the catalyst bed (MR = 40). The H_2 flow is split with part being mixed with O_2 (MR = 1) just before entering the Shell 405 catalyst bed and most (~85%) being used to cool the torch chamber wall. The cooling H_2 reacts with the core exhaust (MR = 5) at the exit to the igniter, providing a very hot torch for thruster ignition.

HIGH-RESPONSE CATALYTIC IGNITER



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Excellent response was obtained under ambient conditions, i.e., response time was 32 msec $\pm$ 3 msec.

IGNITER-ONLY TESTS - HIGH P_C CATALYTIC: AMBIENT TEMPERATURE PROPELLANTS

NAS3-14354

TEST	P_C PSIA	RESPONSE TIME MSEC	TOTAL MR	BED MR	BED TEMP. °F	CHAMBER TEMP. °F	DURATION MSEC
106	215	44	4.84	~1.0	947	-	70
107	253	28	4.84	~1.0	1057	900	65
108	265	35	4.92	~1.0	1726	968	117
109	303	34	4.80	~1.2	2188	1329	116
110	295	28	4.80	~1.2	2247	1201	207
111	104	35	4.80	~1.0	1750	1450	195
112	465	29	4.24	~1.0	-	1057	126

NOTES: RESPONSE TIME; FIRST PRESSURE RESPONSE IN H_2 MANIFOLD TO 100% P_C

DURATION; TIME AT 100% P_C

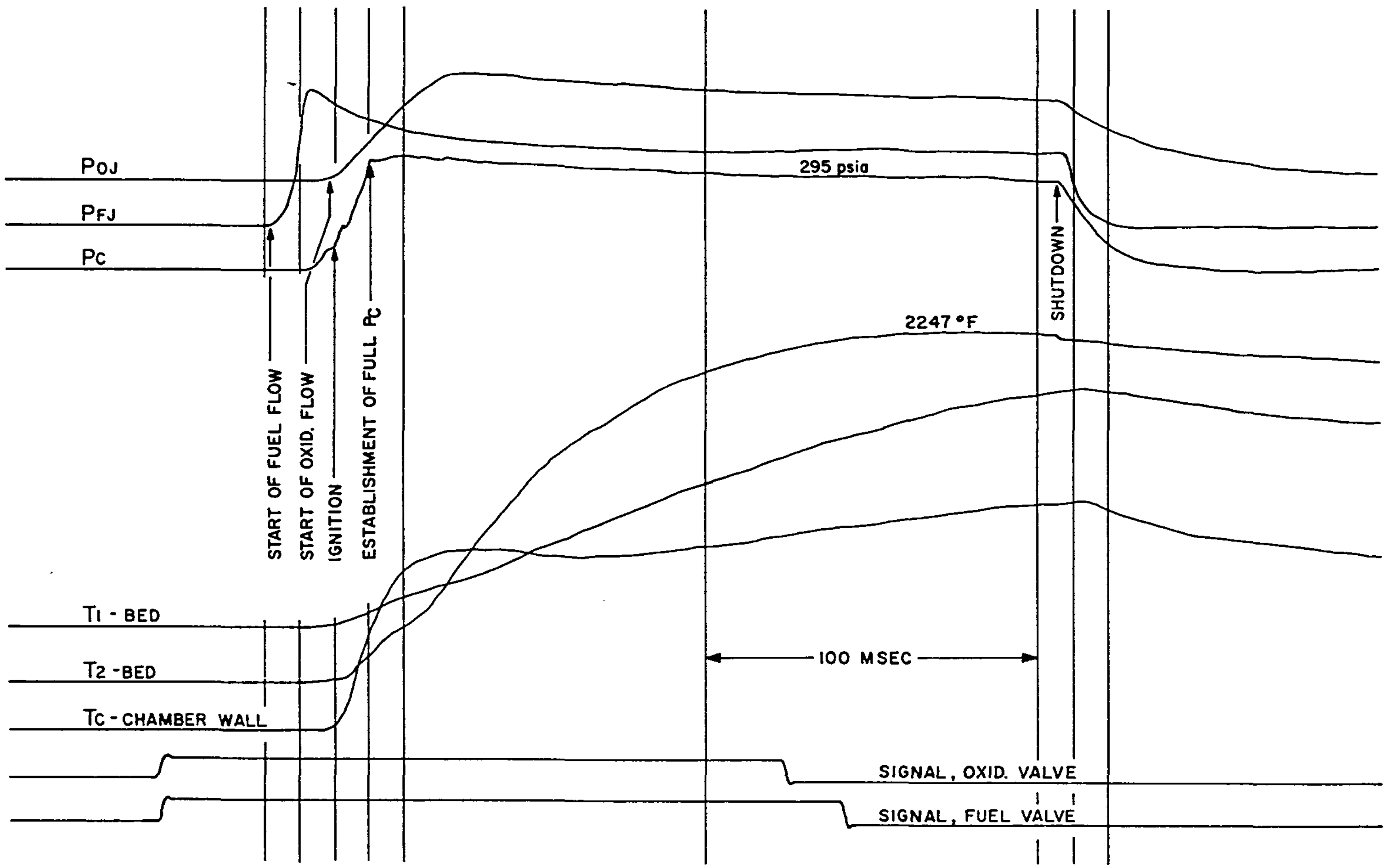
CATALYST; SHELL 405

P_C AT FS_1; 0.35 PSIA

A representation of a typical oscillograph trace for a steady-state test is shown. Note that the pressure rise rates are faster than the temperature rise rates. Reaction of the secondary O_2 is virtually instantaneous.

IGNITER-ONLY TESTS - HIGH P_C CATALYTIC: OSCILLOGRAPH TRACE/TEST 110

NAS3-14354



The catalytic igniter performed quite well under pulse conditions. The response time was 28 millisec.

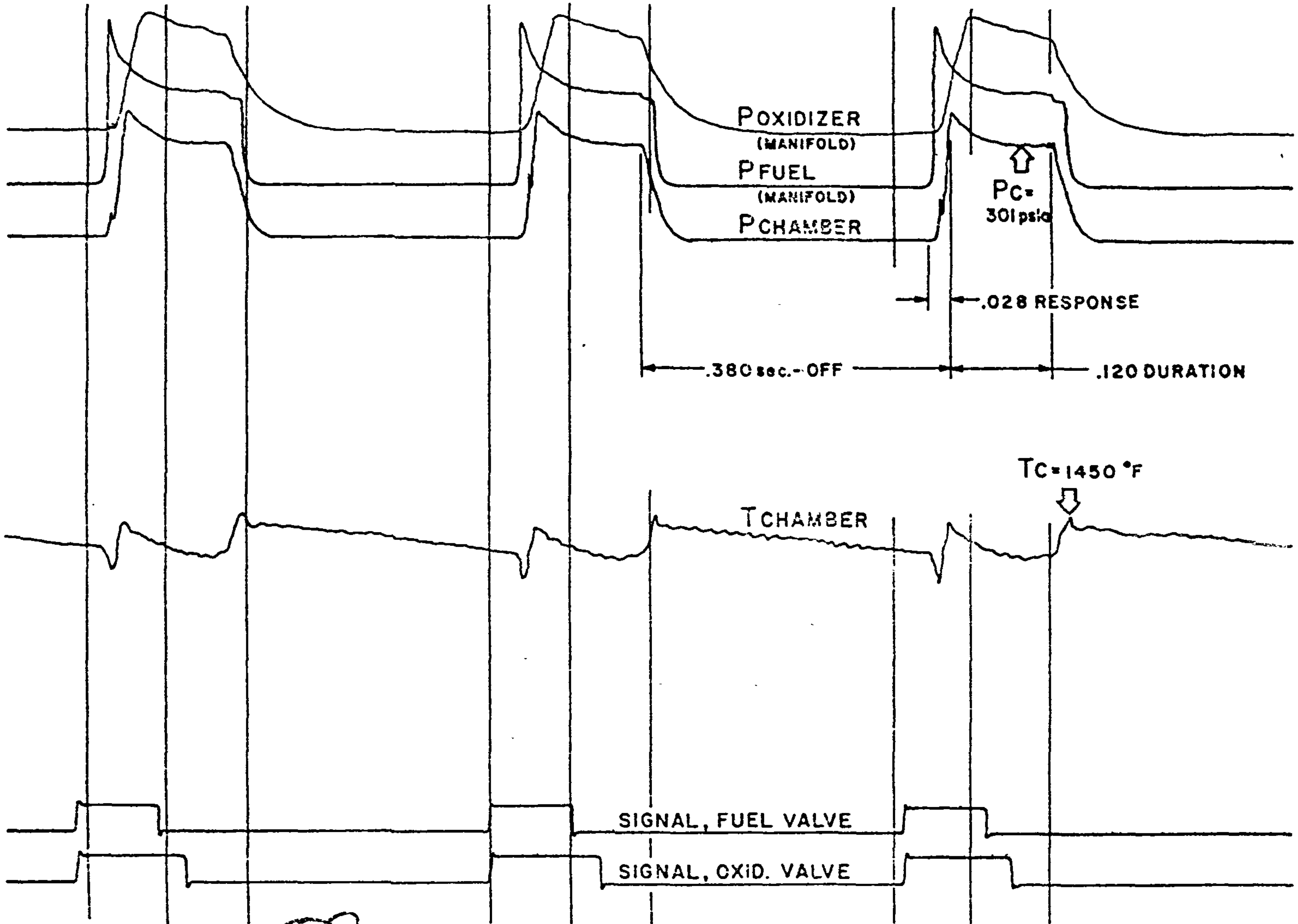
IGNITER-ONLY TESTS - HIGH P_C CATALYTIC: PULSE MODE

NAS3-14354

TEST	P_C PSIA	RESPONSE TIME MSEC	TOTAL MR	BED MR	CHAMBER BED °F	NOTES
114	294	29	4.92	~1.0	1310	6 CONSECUTIVE PULSE TEST, 120 MSEC ON/380 MSEC OFF TOTAL ON TIME, 720 MSEC
115	294	28	4.92	~1.0	1450	100 CONSECUTIVE PULSE TEST 120 MSEC ON/380 MSEC OFF TOTAL ON TIME, 12 SEC

A representation of the oscillograph trace for the second pulse test is shown, i.e., pulses 98, 99, and 100. The repeatability from pulse to pulse was excellent.

CATALYTIC IGNITER PULSE TEST OPERATION



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The response time (32 msec) of the catalytic igniter remained the same as the temperature was lowered from 500°R to approximately 350°R. At approximately 300 to 325°R, a sharp increase in the response time was noted, i.e., from 2 millisec to 106 millisec.

IGNITER-ONLY TESTS - HIGH P_C CATALYTIC: LOW TEMPERATURE PROPELLANTS

NAS3-14354

TEST	P_C PSIA	TEMP. °R OX	TEMP. °R FU	RESPONSE TIME MSEC	TOTAL MR	DURATION MSEC	CAT BED TEMP, FS_1 °R
122	250	501	500	29	4.8	220	506
123	250	501	500	32	4.8	217	505
124	250	460	460	33	4.8	199	~472
125	258	425	410	33	4.5	189	~425
126	261	418	360	33	4.9	202	~427
127	262	391	330	32	4.0	208	~382
128	250	392	261	39	3.1	209	~350
130	265	322	261	106	3.9	172	~337

NOTES: RESPONSE TIME; FIRST PRESSURE RESPONSE IN H_2 MANIFOLD TO 100% P_C

DURATION; TIME AT 100% P_C

CATALYST; SHELL 405

P_C AT FS_1; 0.35 PSIA

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The catalytic torch igniter was used to ignite the thruster. Excellent results were obtained. Note that by proper sequencing it may be possible to obtain a thruster response time of 20 to 25 millisec.

IGNITER-COMPLETE THRUSTER TESTS - HIGH P_C CATALYTIC: AMBIENT TEMPERATURE PROPELLANTS

NAS3-14354

TEST	P_C PSIA	MIXTURE RATIO IGNITER	MIXTURE RATIO THRUSTER	THRUST lbF	IGNITION DELAY MSEC IGNITER	IGNITION DELAY MSEC THRUSTER	RESPONSE TIME MSEC IGNITER	RESPONSE TIME MSEC THRUSTER	DURATION SEC
120	302	4.99	4.00	1258	3	6	37	80	1.2
136	319	4.88	5.08	1314	2	2	40	63	1.1

NOTE: BY PROPER SEQUENCING OF IGNITER AND THRUSTER, IT MAY BE POSSIBLE TO REDUCE THRUSTER RESPONSE TO AN OVERALL VALUE OF 20 TO 25 MSEC.

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"SPARK AND AUTO IGNITION DEVICES FOR ACPS"

J. R. LAUFFER

ROCKETDYNE

TECHNICAL MANAGER

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TITLE:	SPARK AND AUTO IGNITION FOR APS
CONTRACT:	NAS3-14351
PRESENTER:	J. R. LAUFFER, PRINCIPAL ENGINEER
COMPANY:	ROCKETDYNE, DIV. OF NORTH AMERICAN ROCKWELL
NASA PROJ. MANAGER:	E. EDELMAN

Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

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ABSTRACT

The use of oxygen and hydrogen propellants in the Space Shuttle Auxiliary Propulsion System (SS/APS) has established unique ignition system requirements. As many as 50 thrusters must be ignited reliably up to 1×10^6 times each with a minimum of interference with other systems. Ignition systems for SS/APS have been investigated analytically and experimentally under NASA and Rocketdyne funding. An improved augmented spark igniter (ASI) and a new auto-ignition device have been evaluated in detail.

The systematic engineering investigation conducted for both ignition concepts is described The preliminary design concept, laboratory tests, igniter design(s), "igniter only" testing, "ignition/thruster ignition" testing, and the overall ignition system design are discussed. An improved augmented spark igniter design with very low power requirements and a uniquely packaged integrated spark plug/exciter unit which minimizes weight and eliminates R.F.I. is presented. An auto-ignition device employing a resonance heating phenomenon and requiring no external electrical power or catalytic material is also presented.

On 23 July 1970, Rocketdyne received Contract NAS3-14351 to evaluate the spark and auto-igniter systems. The objective of this program was to investigate experimentally candidate ignition systems for the Space Shuttle APS. The nine-month program was comprised of three basic tasks: Task I - Igniter Design and Fabrication, Task II - Igniter Testing, and Task III - Preliminary System Design. During Task I, laboratory-type testing was conducted on both ignition techniques to evaluate basic design variables prior to final igniter design. In Task II, "igniter only" testing was conducted to obtain data on igniter operational limits. High and low pressure thruster testing was also conducted to determine the effects of igniter energy (mixture ratio and flowrate) on thruster ignition delay. A complete ignition system design was conducted during Task III for the spark and auto-igniter for the high pressure applications.

This technology effort will provide the groundwork input for APS trade-off studies. It will ensure that specifications established for the APS are realistic and achievable and will provide the basis for an expeditious future engine development program.

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SPACE SHUTTLE AUXILIARY PROPULSION SYSTEM

IGNITION SYSTEM

NAS3-14351

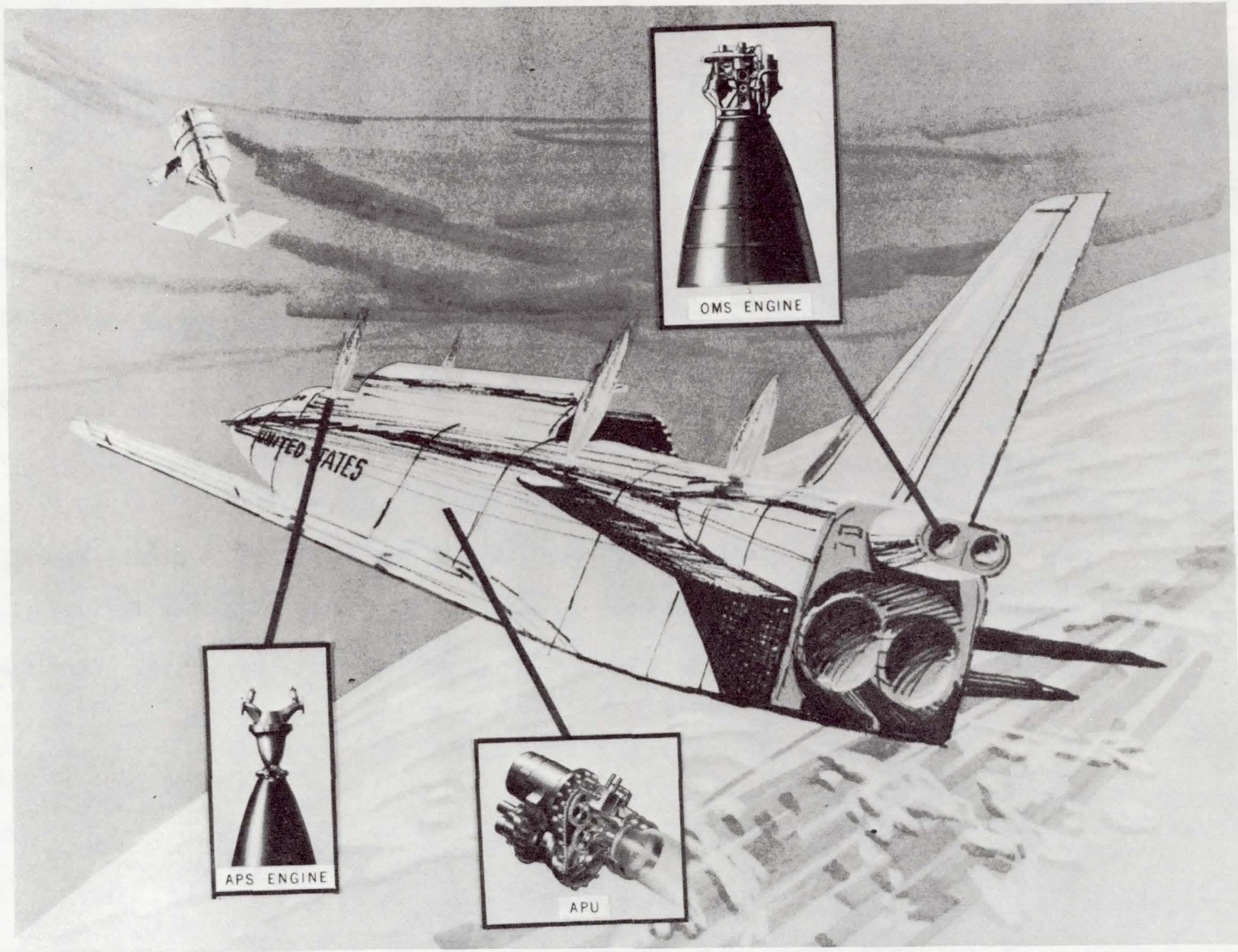
OBJECTIVE: CONDUCT AN ANALYTICAL AND EXPERIMENTATION PROGRAM OF DESIGN, FABRICATION, AND TEST OF GASEOUS HYDROGEN - OXYGEN IGNITION SYSTEMS. OBTAIN DATA FOR WIDE RANGES OF OPERATING CONDITIONS AND COMPLETE A PRELIMINARY DESIGN OF THE APS IGNITION SYSTEM

Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

The selection of oxygen and hydrogen as the propellants for the Space Shuttle has established the requirement for a highly reliable and long life ignition system. Igniters are required for a number of auxiliary propulsion system components. The thrusters, gas generators, conditioners, auxiliary power units, the OMS, and the main engines require igniters. The program discussed in this paper concentrated on the APS thruster ignition system although the technology obtained is applicable to all the Auxiliary Propulsion System components and possibly the main engine. Depending on the amount of redundancy required, up to 150 ignition systems could be required on the Space Shuttle vehicle.

SPACE SHUTTLE ORBITER



The SS/APS igniter is located in the center of the injector face and fires axially along the thruster centerline. Main propellant ignition is accomplished by this igniter torch formed by approximately 1 percent of the thruster flowrate. A fuel-rich torch was selected (M.R. = 1-1.5) based upon existing O_2/H_2 rocket engine experience on the RL-10 and the man-rated J-2 engine.

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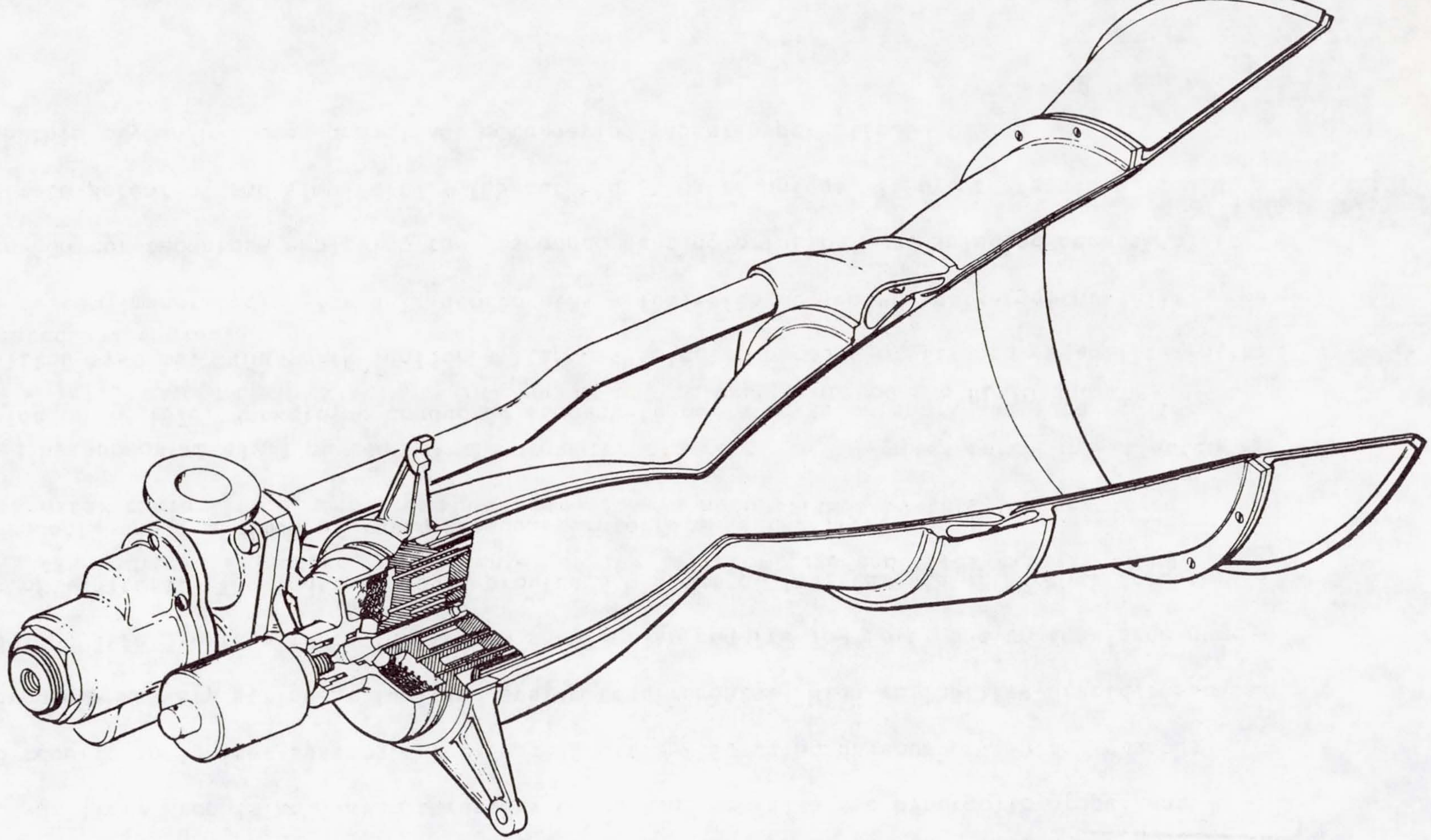
SPACE SHUTTLE APS IGNITION



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Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

The Auxiliary Propulsion System thrusters on current vehicles use hypergolic propellant and require no igniter systems. The Space Shuttle APS, using gaseous oxygen and gaseous hydrogen, has very stringent ignition system requirements. High reliability, rapid response and long life are required. The system should also require low power due to the large number of igniters. The igniter should provide a minimum of interference with other thruster components and a minimum of radio frequency interference (R.F.I.).

During early 1970, Rocketdyne conducted company-funded efforts on spark and auto-igniter ignition systems. The spark igniter system was selected because of its proven ability in the J-2 system (man-rated). The J-2 incorporates a fuel-rich augmented spark igniter (ASI). The auto-igniter technique employing the resonance heating principal was selected because of its ultimate potential and simplicity. The auto-ignition technique, if proven feasible, could make this oxygen/hydrogen propellant combination appear hypergolic.

SPACE SHUTTLE APS IGNITION

REQUIREMENTS

- PROVIDE RELIABLE IGNITION
- HIGH RESPONSE (<50 MILLISECONDS)
- 1,000,000 CYCLE LIFE
- 10 YEARS OPERATION
- SIMPLE
- LOW POWER
- NON-INTERFERENCE

SELECTED APPROACHES

- ADVANCED SPARK IGNITER
- AUTO-IGNITION

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Although the fuel-rich augmented spark igniter is currently used in the man-rated Saturn vehicle, several areas require improvement. These improvements are primarily electrical in nature. The present capacitance discharge electrical units are large, heavy, consume too much power, and cause R.F.I. The goal of the spark igniter was to reduce electrical power requirements and provide a system with low R.F.I., low weight, and a minimum of required maintenance.

275-870
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SPARK IGNITER GOALS

- LOW ELECTRICAL POWER
- LOW RFI
- LOW WEIGHT AND LOW MAINTENANCE

The technical approach adopted during the program was to design the basic igniter with the proper propellant injector technique and spark plug location to obtain reliable ignitions with low delivered energy. To further reduce the electrical power required from the vehicle energy transfer efficiency was increased with improved spark plug design and improved electrical conditioning equipment.

An inductive energy storage technique instead of the conventional capacitance discharge technique was selected to provide high efficiency and low rates of input current change. The low rates of current change will provide low conducted R.F.I. Of great significance is the elimination of a high tension pressurized cable between exciter circuitry and spark plug. This cable is a noise generator and requires periodic maintenance.

TECHNICAL APPROACH

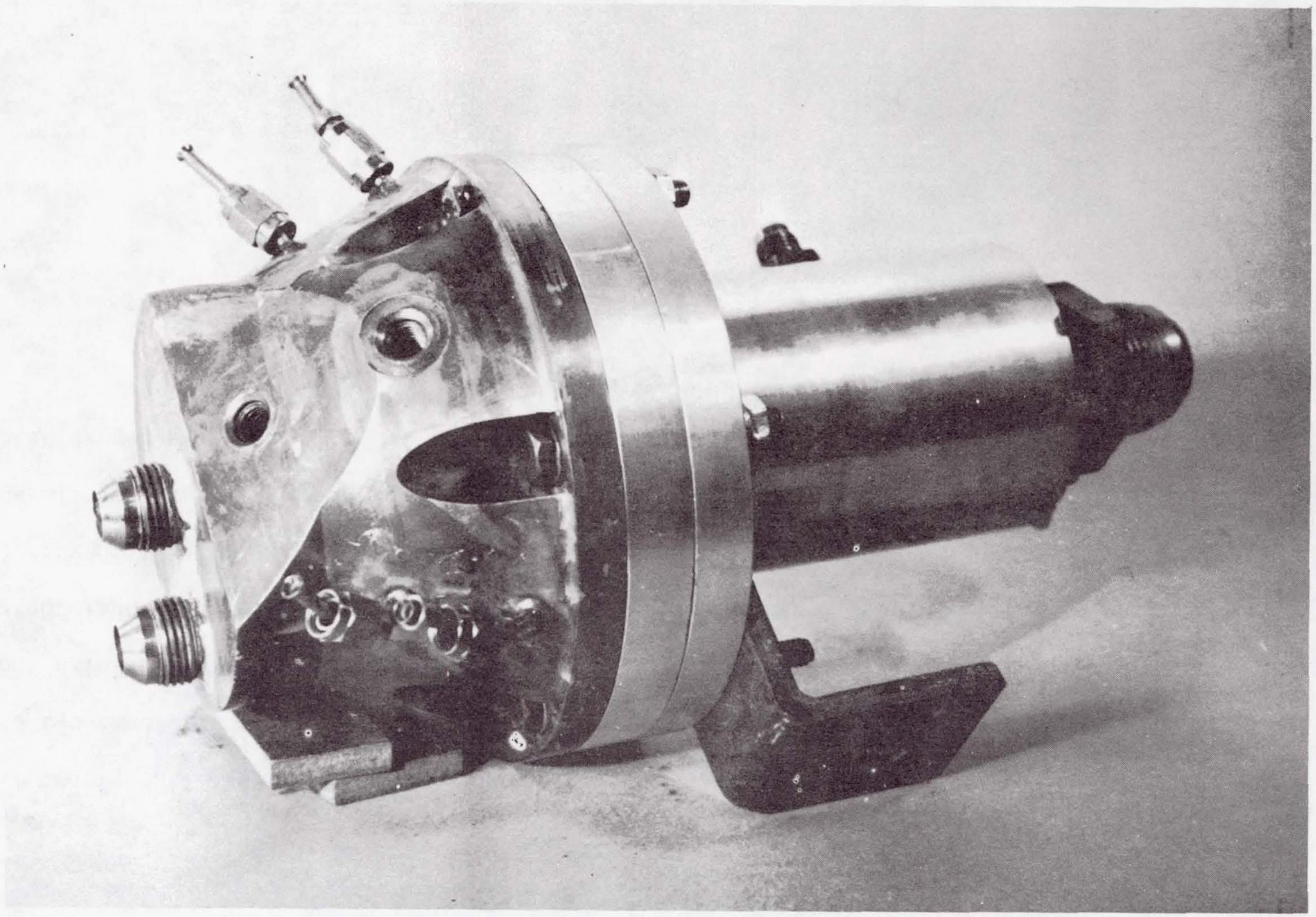
- DESIGN IGNITER TO REQUIRE LOW ENERGY
- INCREASE ENERGY TRANSFER EFFICIENCY
- INDUCTIVE ENERGY STORAGE
- SMALL INTEGRATED EXCITER/PLUG

To obtain reliable spark ignition at low spark energy, it is desirable to have relatively high mixture (> 1.4 o/f) and low velocity gas at the spark plug tip. Several candidate igniter configurations were evaluated. Cold flow tests were conducted using oxygen and helium gases to determine local gas mixture ratio and velocity prior to the igniter ignition. The cold flow hardware of the selected configuration is shown.

SPARK IGNITER COLD FLOW

CONFIGURATION III

683

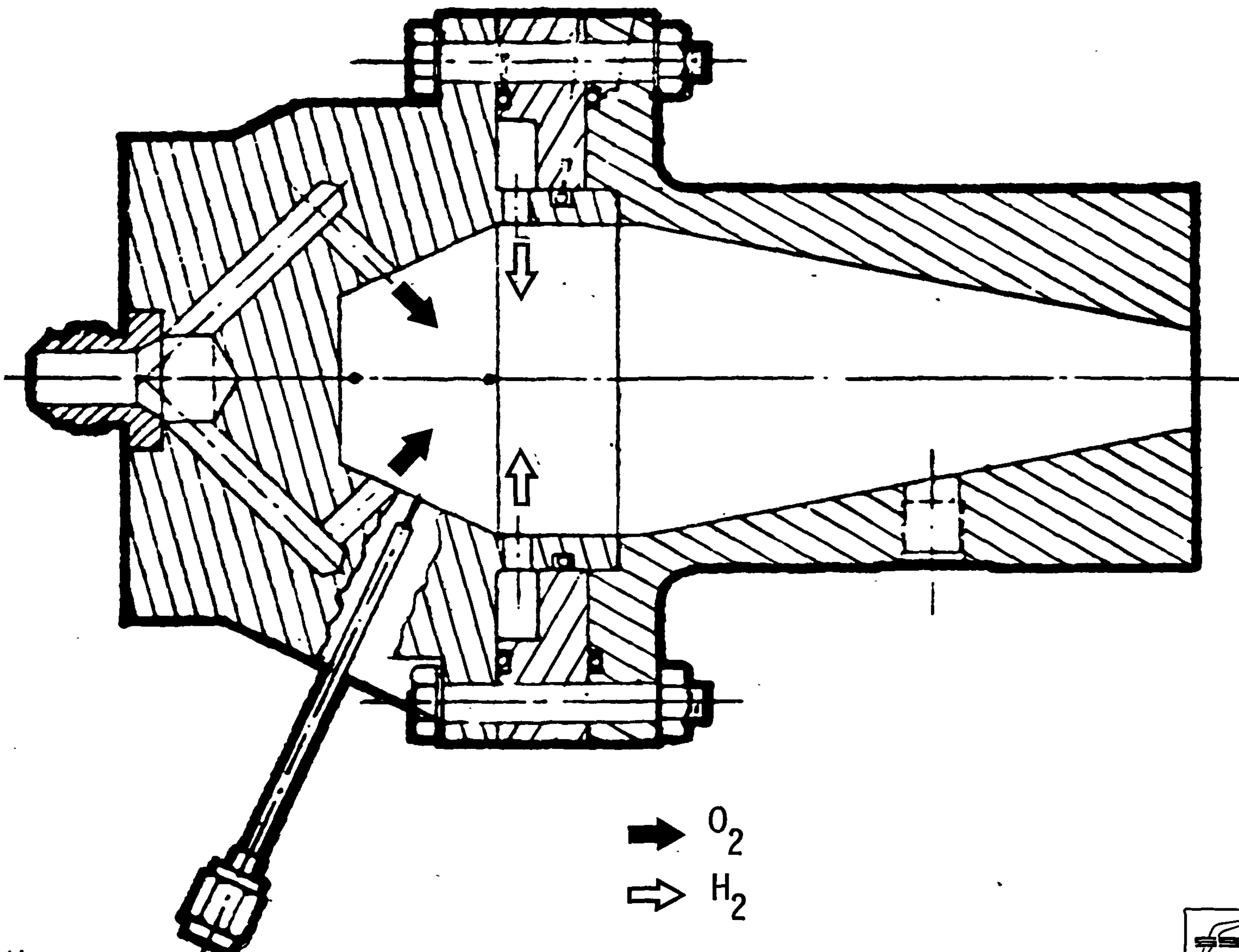


An igniter configuration very similar to that used in the J-2 engine was cold-flow tested and two candidate spark plug locations were selected. In this igniter configuration, oxidizer is injected in two impinging orifices. All the fuel is injected through eight tangential orifices downstream of the oxidizer impingement point. The large majority of the fuel stays next to the igniter wall and flows toward the igniter exit. A small portion of the fuel flows toward the back of the igniter and mixes with recirculated oxygen.

SPARK IGNITER COLD FLOW

CONFIGURATION III

275-872
3-71



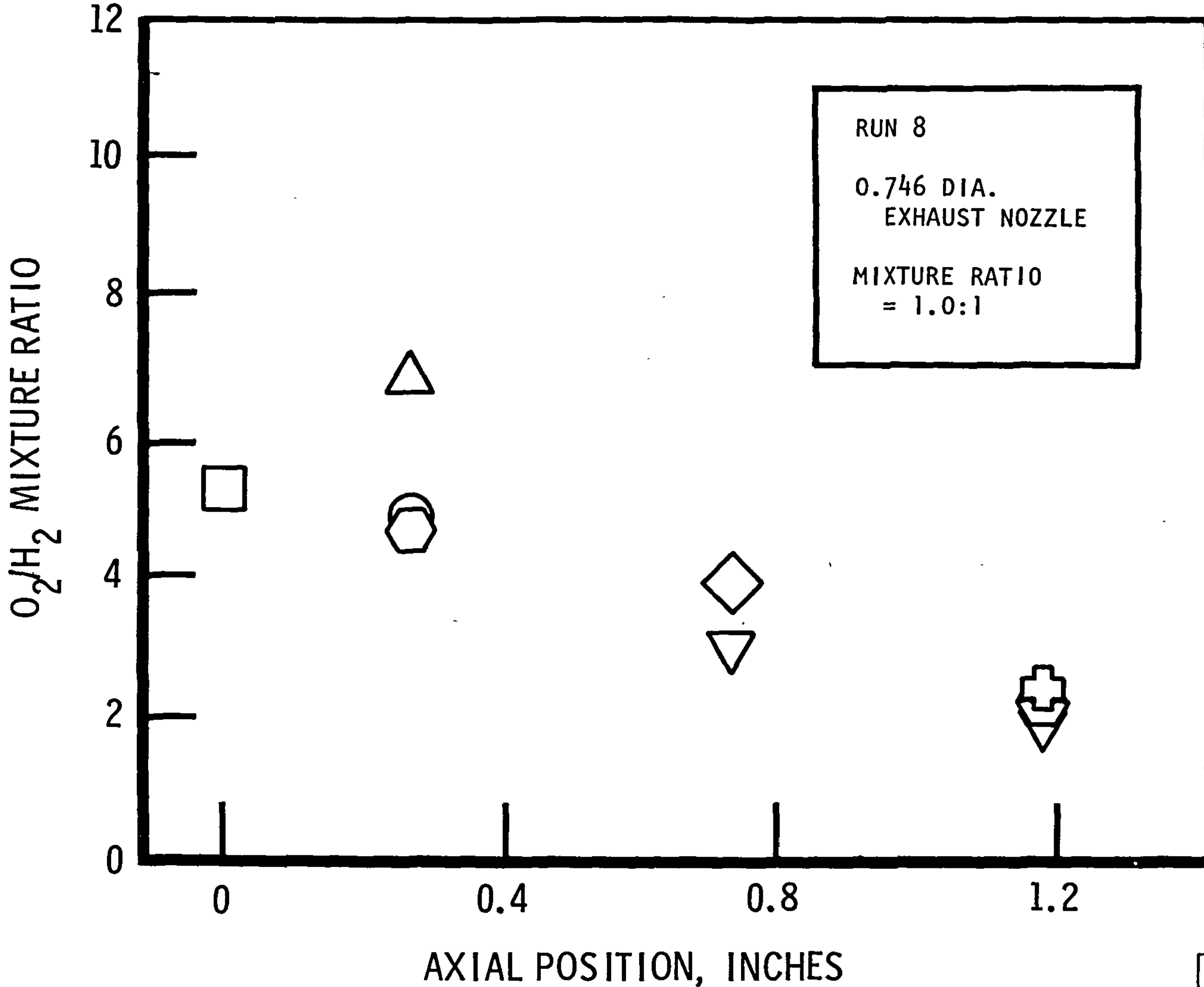
Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

Analysis of the cold flow data indicated the desired high mixture ratio and relatively low velocity gas could be obtained with this configuration in the region upstream of the fuel injection orifices. With an overall igniter mixture ratio of 1.0 o/f, local mixture ratios of approximately 6 were obtained at the back surface of the igniter and locations approximately 0.4 inch downstream. The data points shown are for a number of circumferential locations.

SPARK IGNITER COLD FLOW

CONFIGURATION III



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space shuttle engine
AUXILIARY PROPULSION SYSTEM

Laboratory tests were conducted to evaluate the characteristics of several different spark plug configurations. The objective of this testing was to obtain spark plug efficiency data. Desirable spark plug characteristics for an inductively powered spark are low breakdown voltage (i.e., the voltage required across the gap to initiate the discharge) and a high sustaining voltage (i.e., the voltage that is maintained across the gap during the discharge). It is also desirable to keep the breakdown voltage relatively low (< 10 kv) to aid in transformer design (turns ratio) and to provide high system efficiency. Based upon the laboratory data, the dual recessed gap spark plug was selected due to its high sustaining to breakdown voltage ratio and its breakdown voltage was below 10 kv. This spark plug was used successfully on subsequent hot-firing tests. Dual gap plugs produce two series discharges with each spark. A "floating" intermediate metallic ring acts as the intermediate electrode. Dual gap plugs as a family exhibited greater efficiencies than their single gap counterparts.

Recessed surface gap plugs have additional advantages over radial cap plugs. The breakdown voltage (for small gaps) reaches an asymptote with increasing pressure and electrodes are kept away from hot combustion gases.

All plugs were evaluated for maximum V_{BD} and U_{SUS} variation from 0.2 to 60 psia, 200° R to 560° R GN_2.

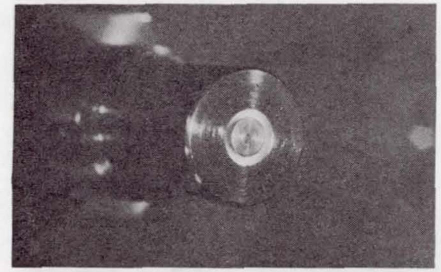
SPARK PLUGS EVALUATED

275-875
3-71

ALL PLUGS
SURFACE GAP

TYPE 1

SINGLE RADIAL GAP

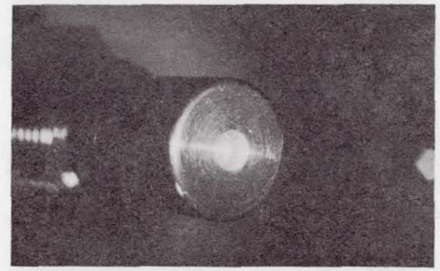


TYPE 2

SINGLE RECESSED GAP

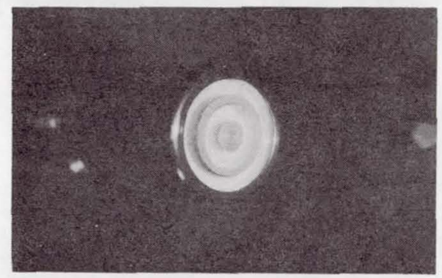
TYPE 2A

SINGLE SHUNTED GAP (J-2)



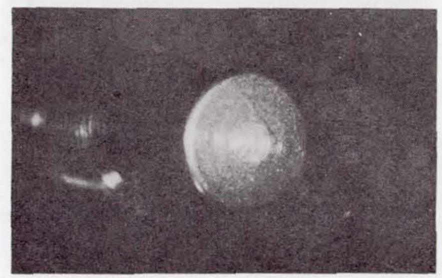
TYPE 3

DUAL RADIAL GAP



TYPE 4

DUAL RECESSED GAP



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AUXILIARY PROPULSION SYSTEM

Drawings of the spark plug electrode configurations described on the previous chart are shown for clarity.

275-904
3-71

SPARK PLUGS EVALUATED

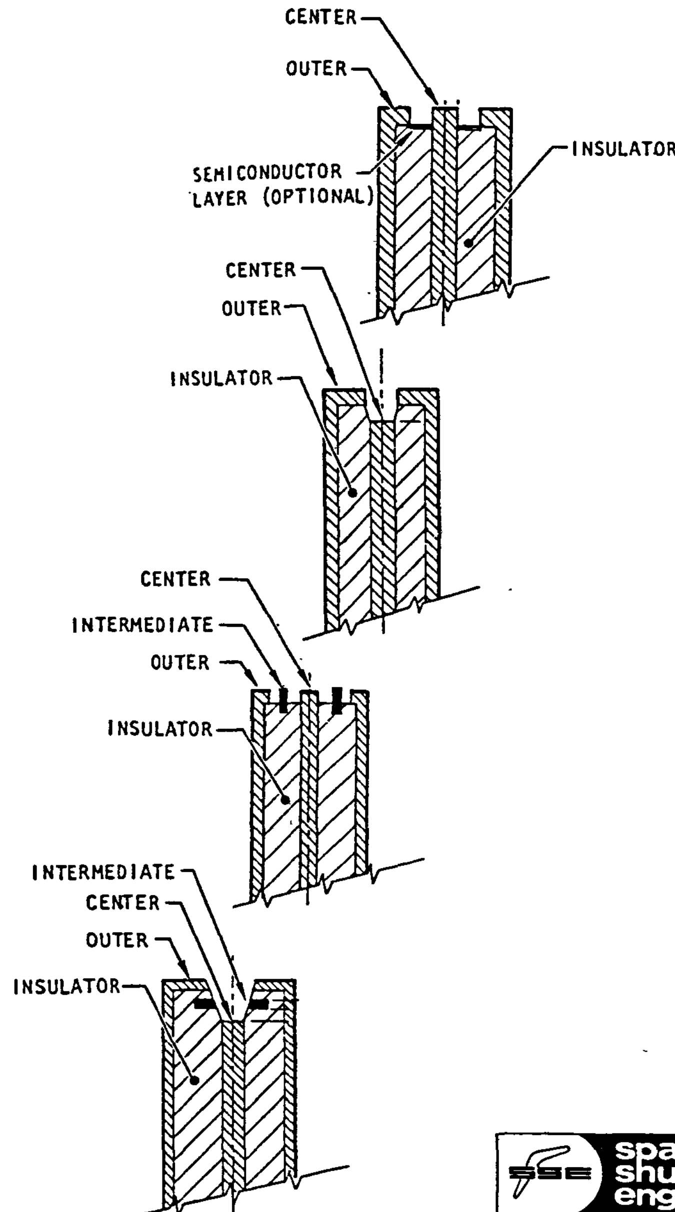
ALL PLUGS SURFACE GAP

TYPE 1
SINGLE RADIAL GAP

TYPE 2
SINGLE RECESSED GAP
TYPE 2A
SINGLE SHUNTED GAP (5-2)

TYPE 3
DUAL RADIAL GAP

TYPE 4
DUAL RECESSED GAP



Rocketdyne
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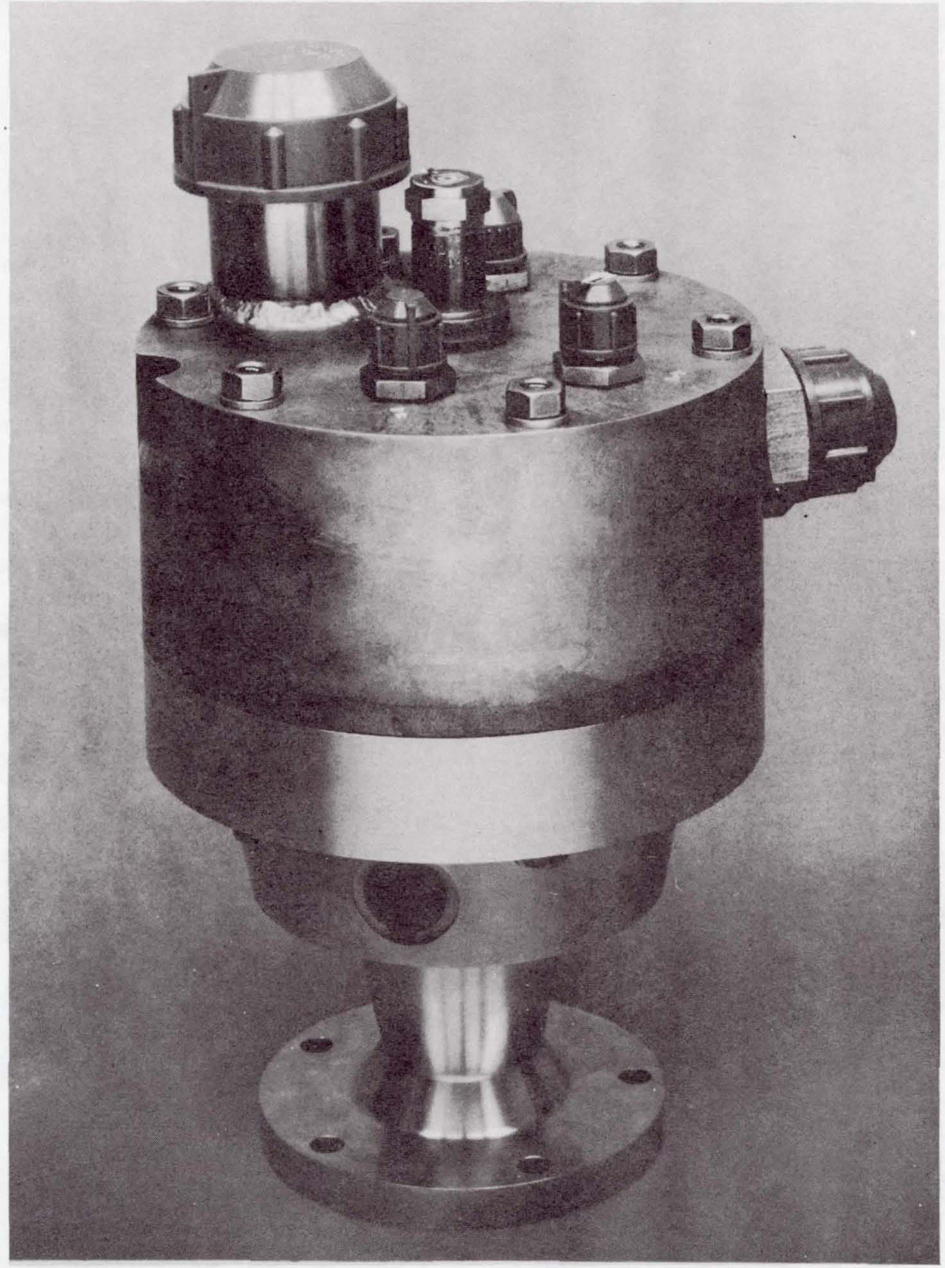
space shuttle engine
AUXILIARY PROPULSION SYSTEM

Based upon the cold flow data, the spark igniters for the high and low pressure SS/APS application were designed and fabricated. The "igniter only" tests were conducted over a range of operational variables. A variable energy and variable spark rate exciter was used for this testing.

The low pressure spark igniter hardware is shown.

275-877
3-71



LOW PRESSURE SPARK IGNITER

Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

103 tests were conducted. The igniter functioned very well, giving rapid response with low energy requirements over the entire range of operational parameters. The most significant data obtained on this test series was that the igniter was insensitive to sequencing (i.e., fuel or oxidizer leads ignited equally well) and that the required energy level was well below the design goal of 20 mj. The fact that the igniter is insensitive to propellant sequencing makes it more reliable and makes it much easier to integrate into the thruster with no separate igniter valves.

275-876
3-71

SPARK IGNITER TEST SUMMARY

(LOW P_c HOT FIRING)

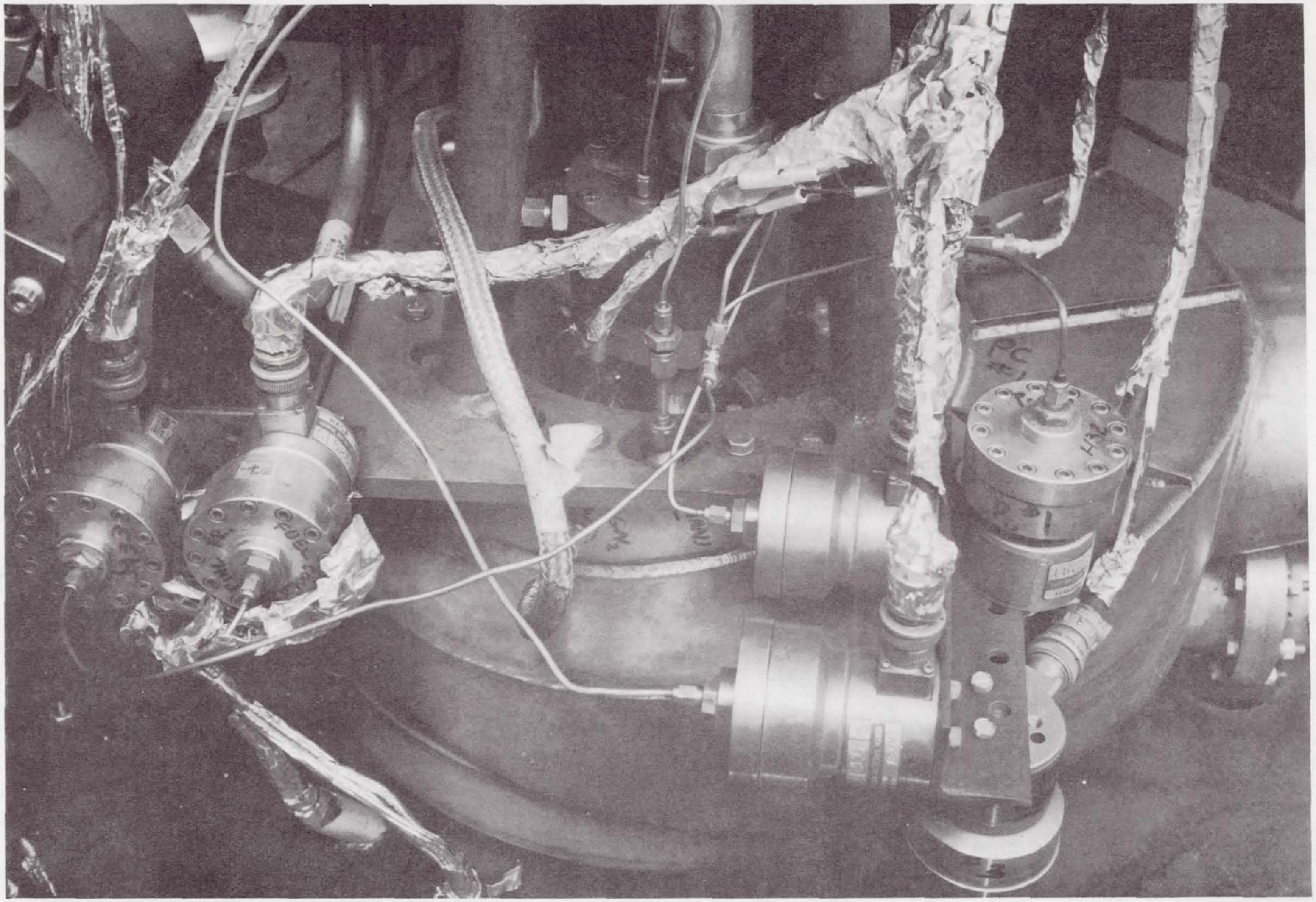
TEST SUMMARY

- 103 TESTS CONDUCTED
- RAPID RESPONSE
- LOW ENERGY
- INSENSITIVE TO SEQUENCING

Upon completion of the low pressure "igniter only" tests, the spark igniter was tested with the complete low pressure thruster assembly. Tests were conducted in the Rocketdyne SS/APS altitude facility. Rapid response of the complete thruster assembly was demonstrated with virtually no igniter delay. The igniter flowrate and igniter mixture ratio were systematically varied to determine the effects of igniter energy on thruster ignition delay. Successful and rapid thruster ignitions were obtained for igniter mixture ratios from 0.8 to 1.5 and igniter flowrates from 0.04 to 0.01 lbs/sec (1 percent to 0.25 percent of thruster flow).

LOW PRESSURE THRUSTER SPARK IGNITER

TEST SETUP



Rocketdyne
North American Rockwell

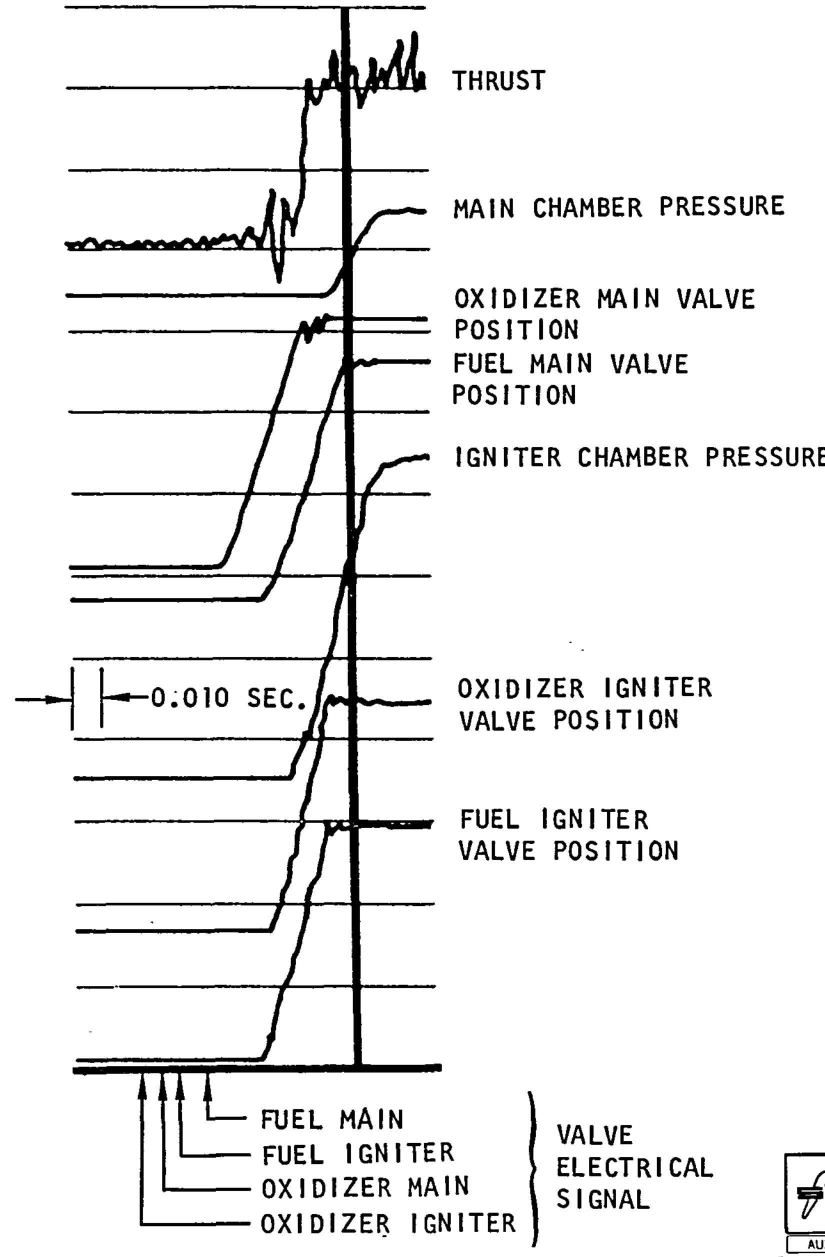
space shuttle engine
AUXILIARY PROPULSION SYSTEM

A typical low pressure thruster start transient is shown. No ignition delay or pressure overshoot is noted in igniter or main thruster chamber pressure. Response was very good, approximately 0.050 sec from electrical signal to 90 percent thrust.

(Although effort on the low pressure APS thruster has been stopped, this technology may be useful on low pressure propellant conditioners or low pressure propulsive vents.)

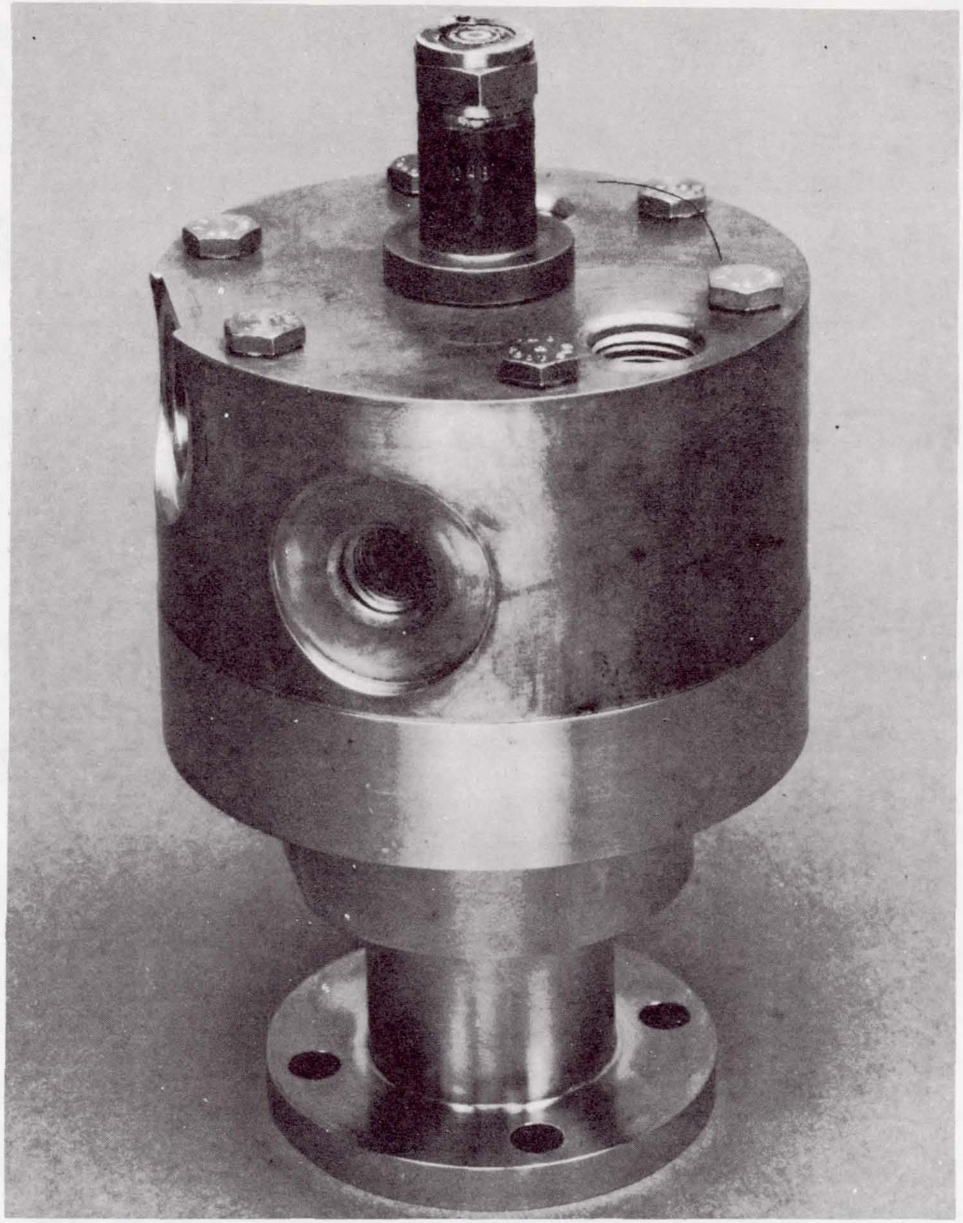
LOW PRESSURE SPARK IGNITION

TYPICAL TRANSIENT



The high pressure spark igniter is very similar in design to the low pressure unit. The igniter has an inside diameter of 1.0 inch. The portion of the igniter which incorporates the propellant injection orifices and the two spark plugs is made of copper; the body is made of stainless steel.

275-884
3-71



HIGH PRESSURE SPARK IGNITER

Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

The high pressure spark igniter was evaluated on 194 tests. Results were very similar to those of the low pressure configuration. Again the response was rapid and the igniter was insensitive to sequencing. Energy requirements were below the 20 mj design goal. The effects of igniter chamber pressure were also investigated. A design chamber pressure of 100 psia was selected because it requires less electrical voltage and energy and this design point will provide a very stiff (275 psid) propellant feed system which makes the igniter less sensitive to propellant inlet conditions.

SPARK IGNITER TEST SUMMARY

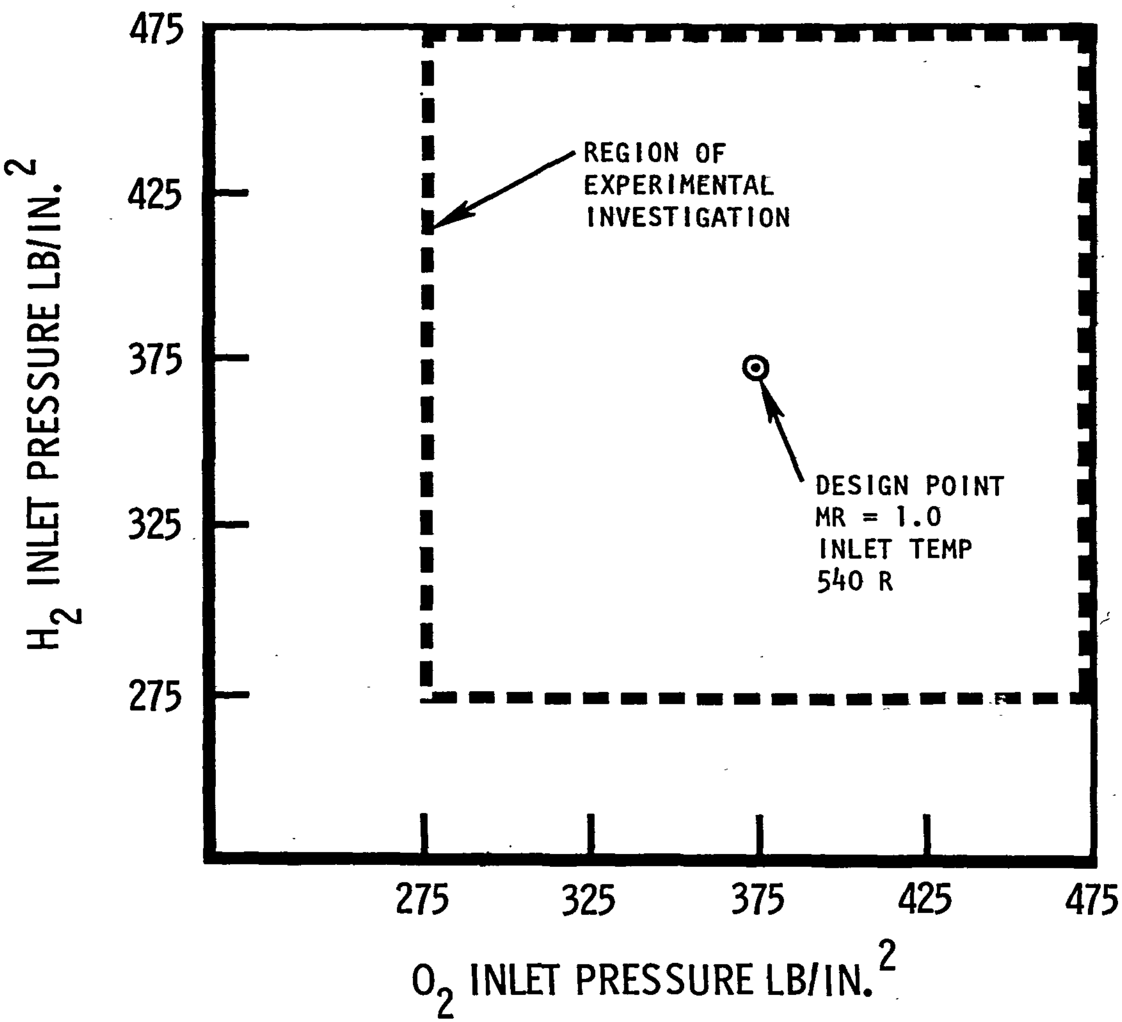
(HIGH P_c HOT FIRING)

TEST SUMMARY

- 194 TESTS CONDUCTED
- RAPID RESPONSE
- LOW ENERGY
- INSENSITIVE TO SEQUENCING

The high pressure spark igniter was tested to evaluate its operation over all combinations of inlet pressures and temperature extremes. Vehicle and auxiliary propulsion system study data were reviewed and a test matrix was established. The pressure and temperature extremes were selected to be outside the expected thruster operation limits. The nominal igniter design point was 540° R propellant inlet temperatures and 375 psia propellant inlet pressures. Each inlet pressure was varied from 275 to 475 psia. As shown on the next page, the propellant inlet temperatures were also varied simultaneously.

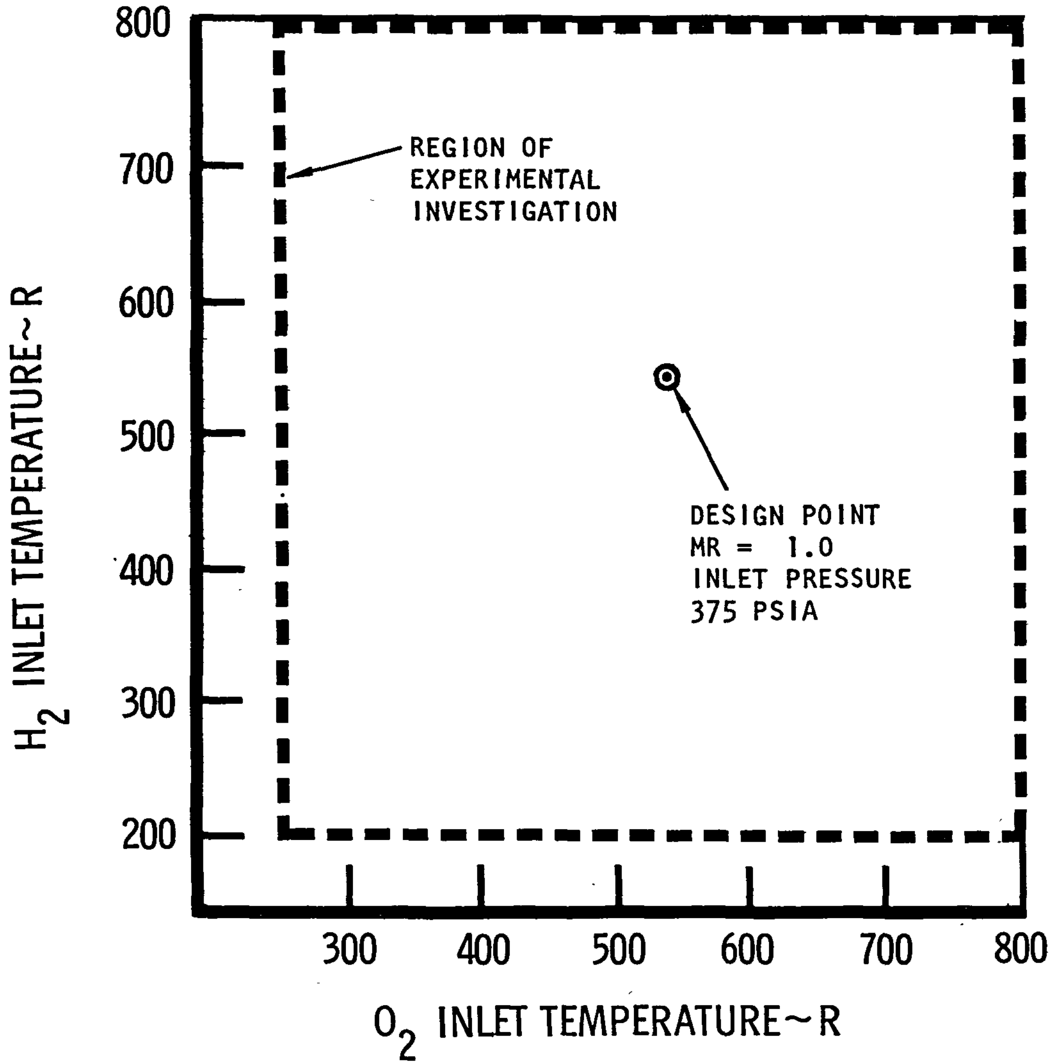
ASI HIGH PRESSURE INLET PRESSURE MATRIX



The fuel inlet temperature was varied from 200° to 800° R. The oxidizer inlet temperature was varied from 250° to 800° R. All corners of the matrix were successfully tested. Resultant igniter mixture ratios varied from 0.3 to 3.0 o/f. Initial igniter hardware temperatures were also evaluated. Successful ignitions were again obtained at all combinations of inlet temperature and pressure with hardware temperatures down to 400° R.

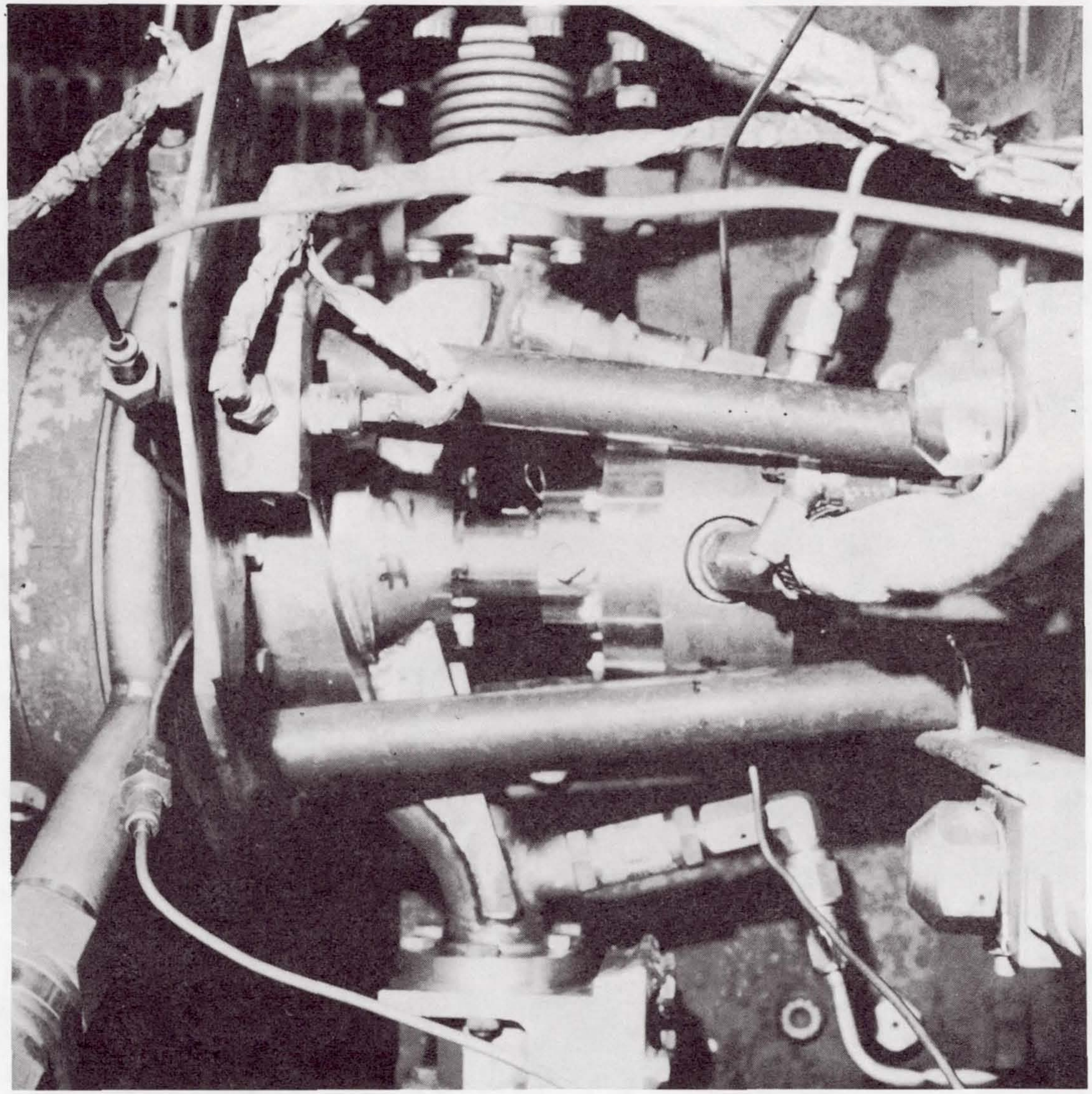
275-882
3-71

ASI HIGH PRESSURE PROPELLANT INLET TEMPERATURE MATRIX



The high pressure spark igniter was installed with the SS/APS thruster. Sea level checkout tests were conducted and altitude testing is currently being conducted. 33 tests have been conducted to date.

275-903
3-71



HIGH PRESSURE IGNITER/THRUSTER

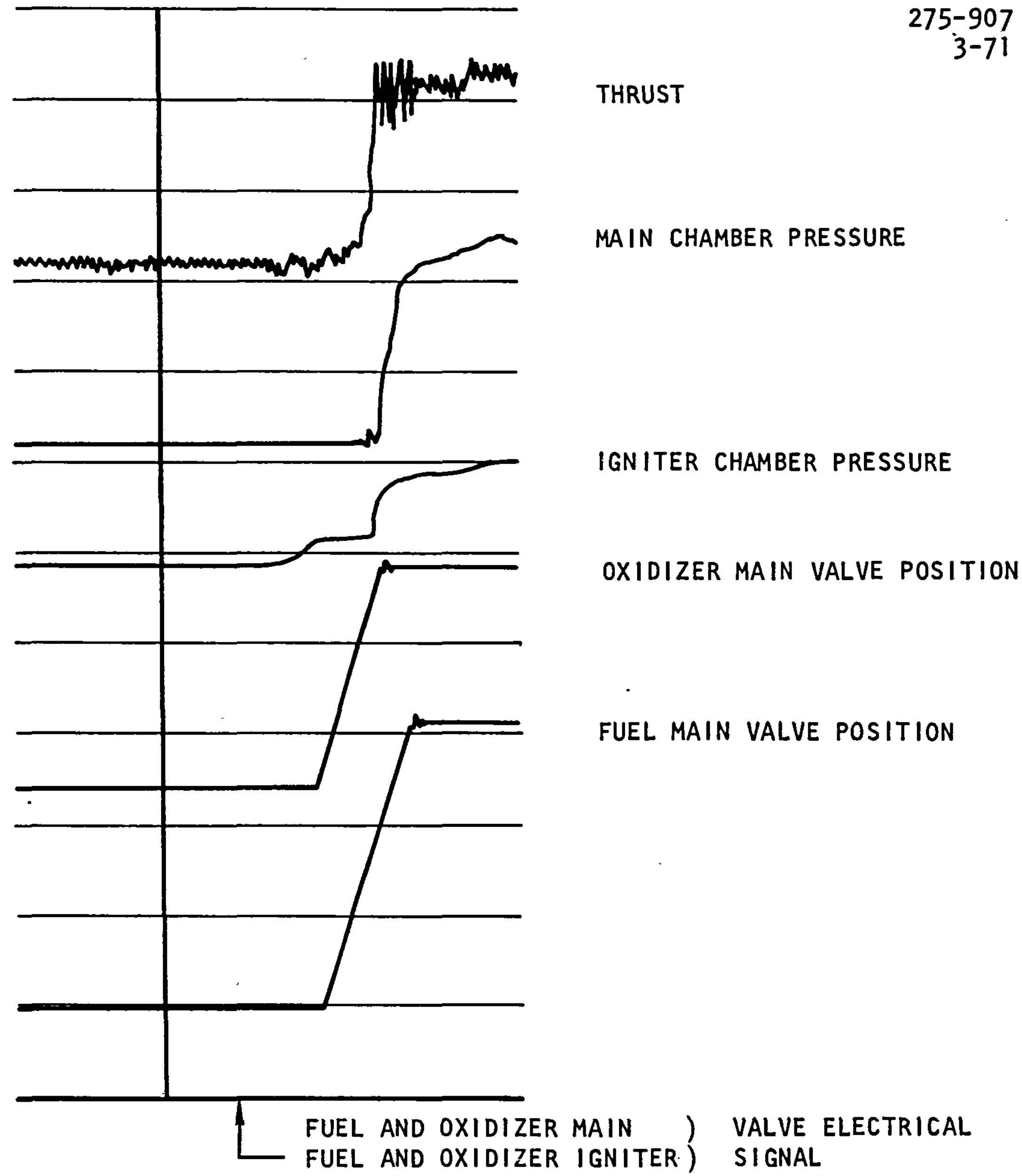
Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

Ignition operation has been 100 percent reliable. Very rapid response has been obtained by sequencing all valves open simultaneously. Approximately 0.040 sec response time from electrical signal to 90 percent thrust has been demonstrated.

HIGH PRESSURE THRUSTER IGNITION

TYPICAL TRANSIENT



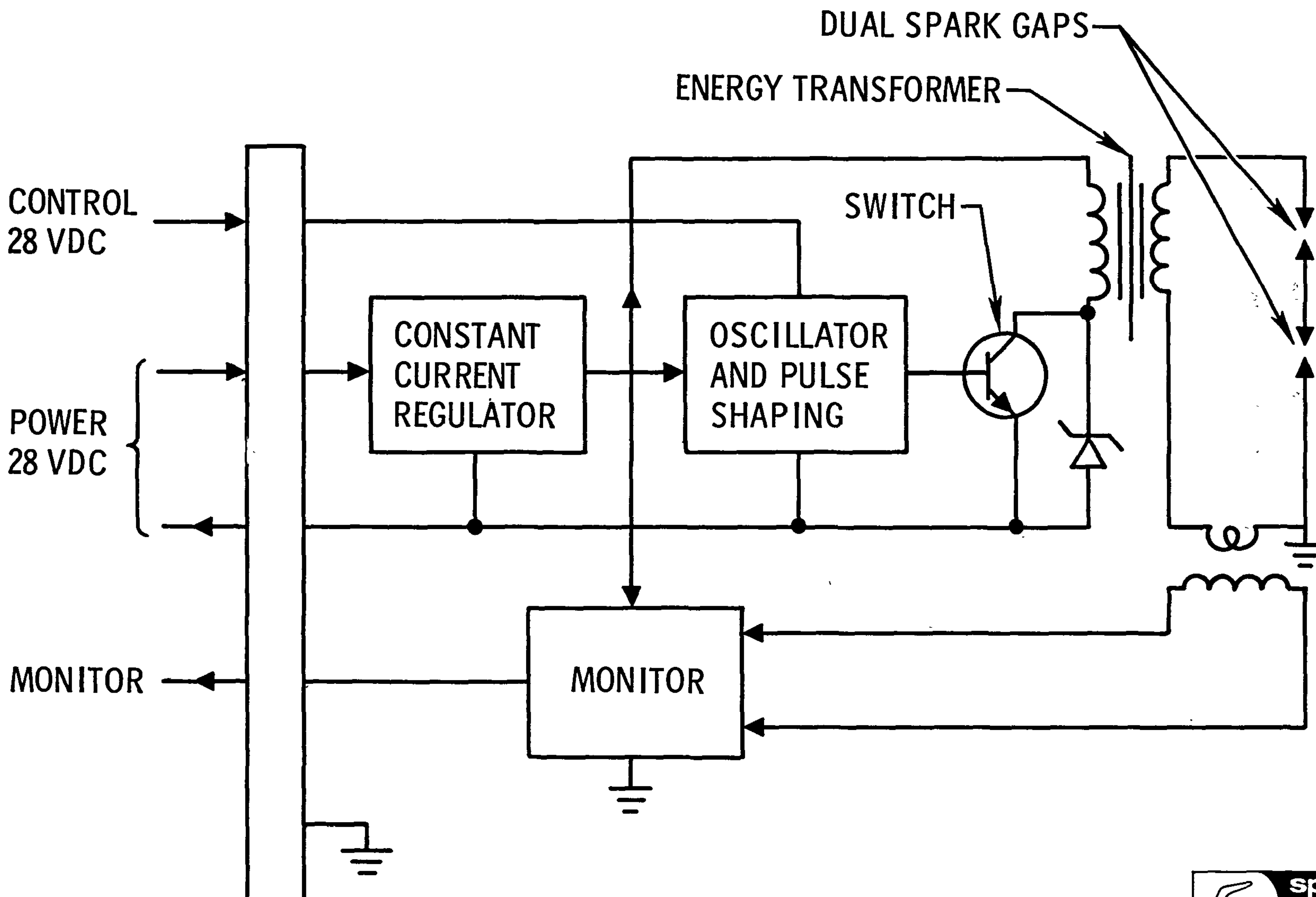
Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

The overall goal of the spark igniter report was to reduce the system size, weight, power requirements and R.F.I. Data obtained during the igniter testing with the workhorse variable energy and variable spark rate exciter was used to establish the design point for the integrated plug/exciter unit. A block diagram of the integrated unit is shown. A four-pin connector provides 28 vdc power, a control signal to turn the unit on, and a monitor signal. The constant current regulator holds the current flow from the vehicle at a low constant rate which virtually eliminates conducted R.F.I. The oscillator generates a 200 cps pulse which switches the transistor. The transformer stores energy (inductively) and also steps the voltage to a sufficient level to break down the dual gaps in the spark plug.

275-897
3-71

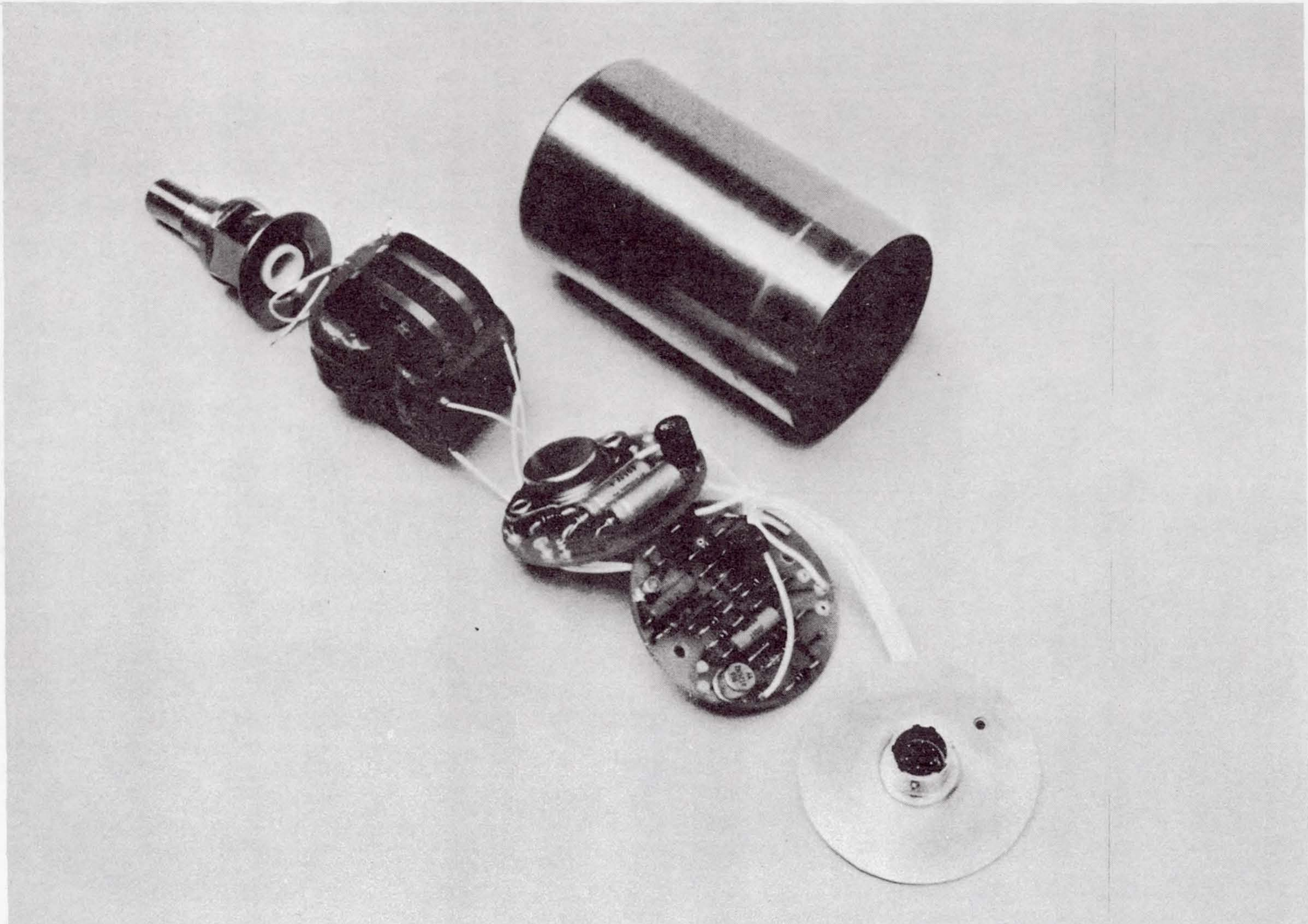
INTEGRATED PLUG AND EXCITER BLOCK DIAGRAM



713

The electronic components are shown prior to final assembly. Electronic components are installed on circuit boards and these boards are installed with the transformer in a small, well-shielded metal case. This hermetically sealed case is welded to the spark plug. Input lines to this unit are low voltage and low current design, requiring no special insulation or shielding.

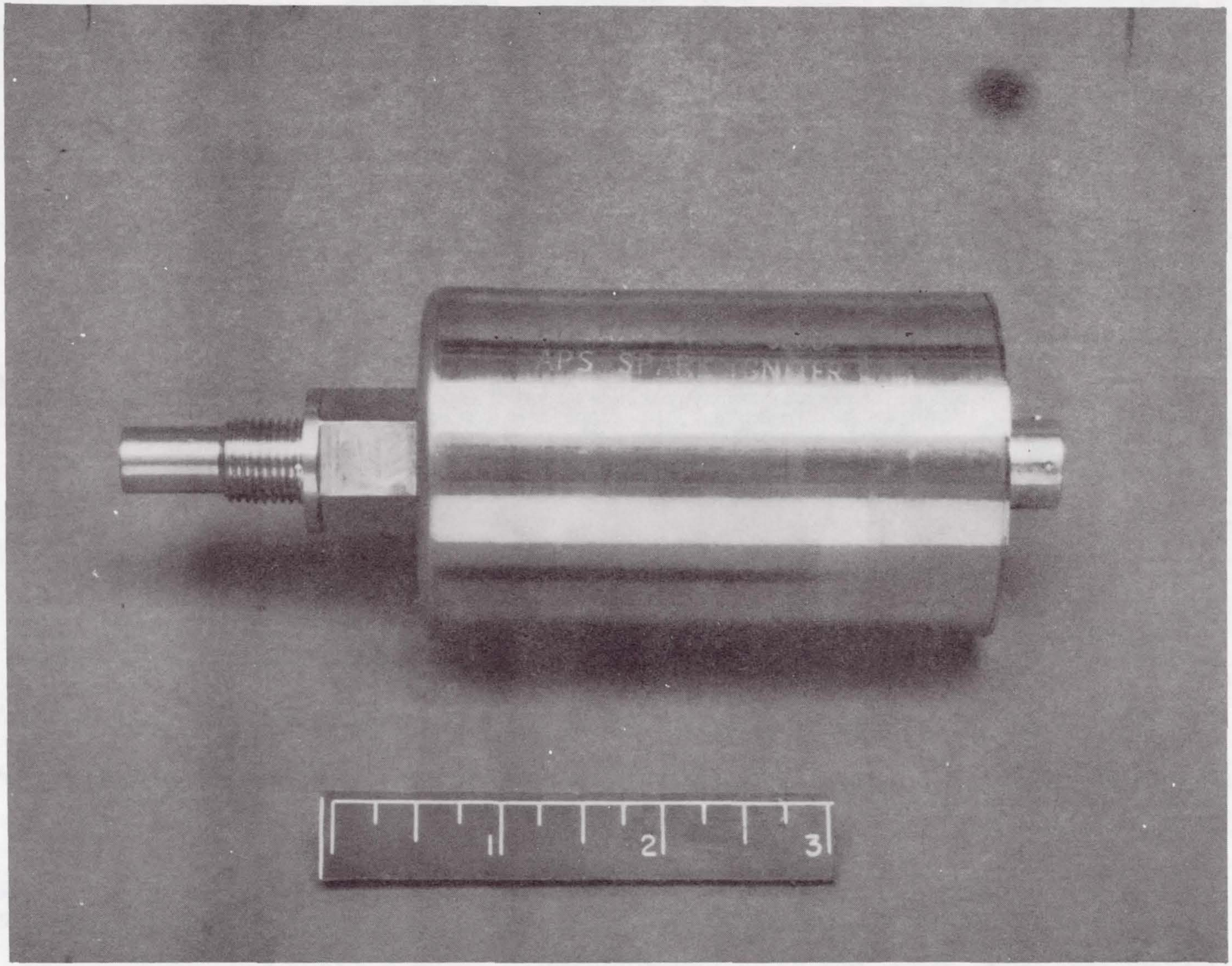
INTEGRATED PLUG AND EXCITER COMPONENTS



The complete unit is shown. This unit includes the spark plug (left end) and all the electrical conditioning equipment.

275-899
3-71

INTEGRATED PLUG AND EXCITER

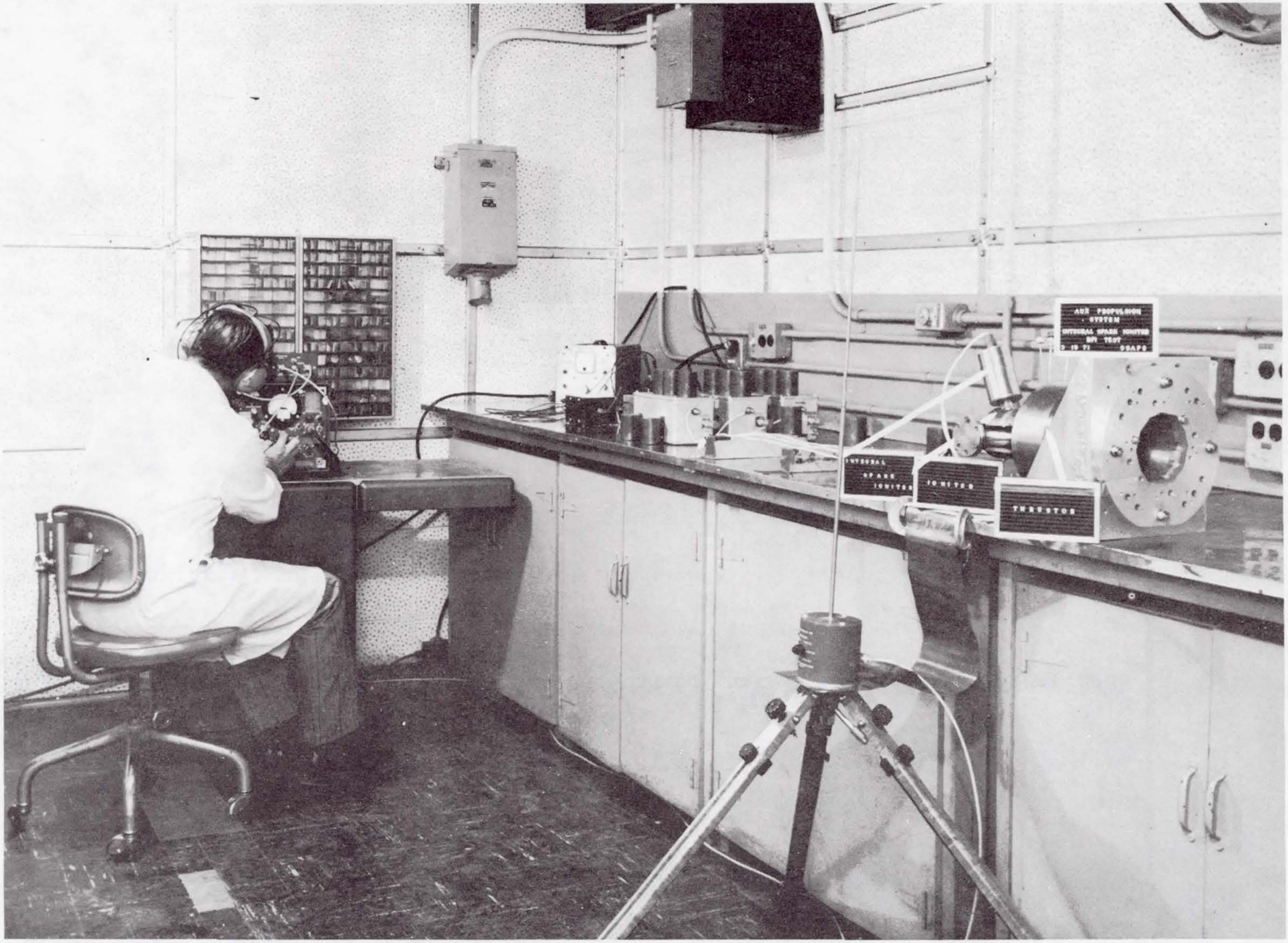


Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

The integrated plug/exciter unit and spark igniter were installed in the 1500-lb thrust SS/APS thruster assembly and R.F.I. tests were conducted. These tests were run in the Rocketdyne screen room to determine radiated as well as conducted R.F.I.

SPARK IGNITER RFI TESTING

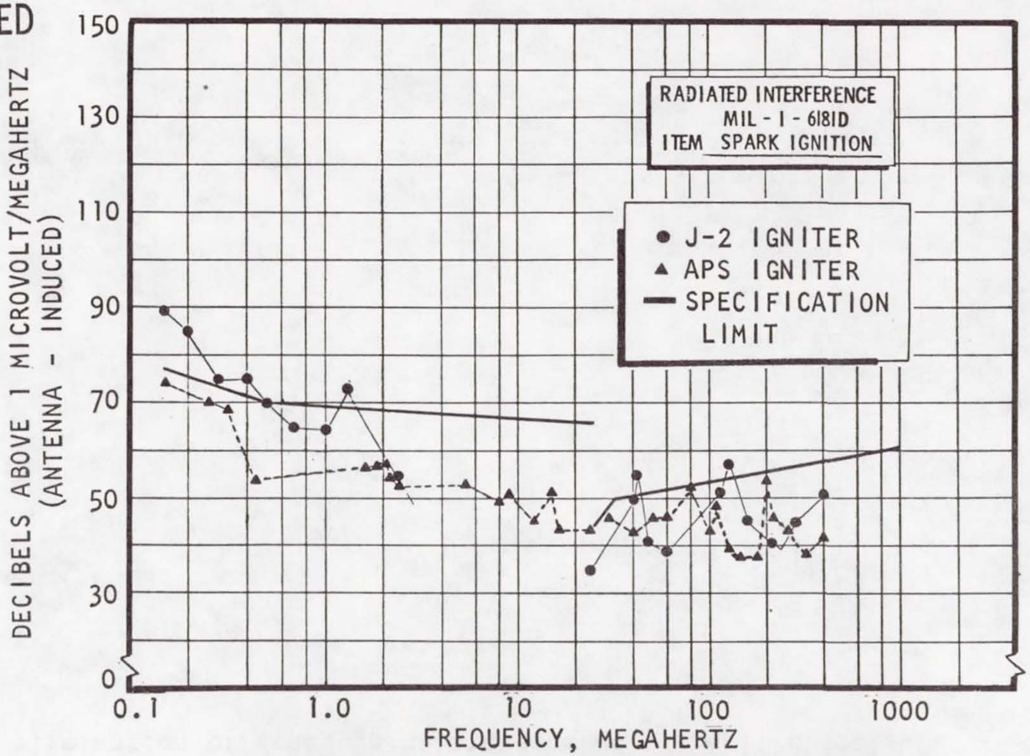


Rocketdyne
North American Rockwell

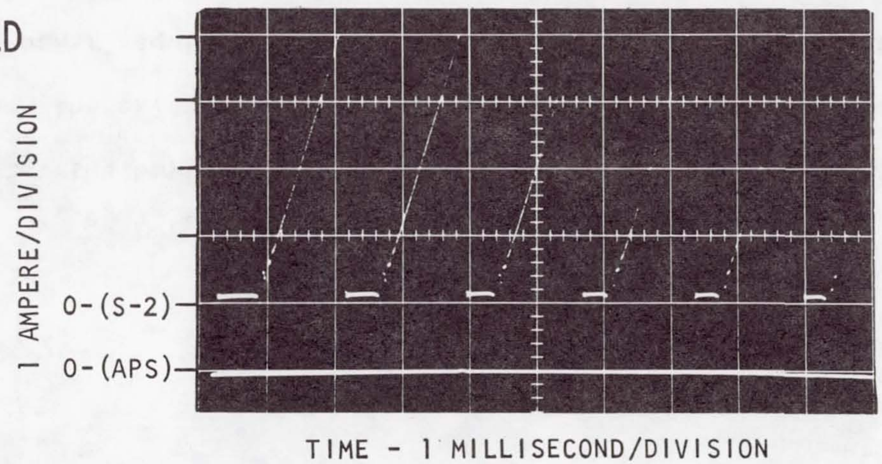
The results of the radiated R.F.I. tests are compared to the Mil Spec and a conventional capacitance discharge spark system. The input current to the SS/APS exciter and the conventional-type exciter were also monitored. The high rates of current change with time for the conventional exciter is indicative of high conducted R.F.I. The almost imperceptible input current for the SS/APS exciter will produce virtually no conducted R.F.I.

RFI TEST RESULTS

RADIATED



CONDUCTED



To summarize, the basic objective of the spark igniter portion of the program was to improve upon the existing state-of-the-art spark ignition system. A reduction of the electrical power required, the elimination of R.F.I., the reduction of system weight, and the elimination of schedule maintenance were specific goals.

275-883
3-71

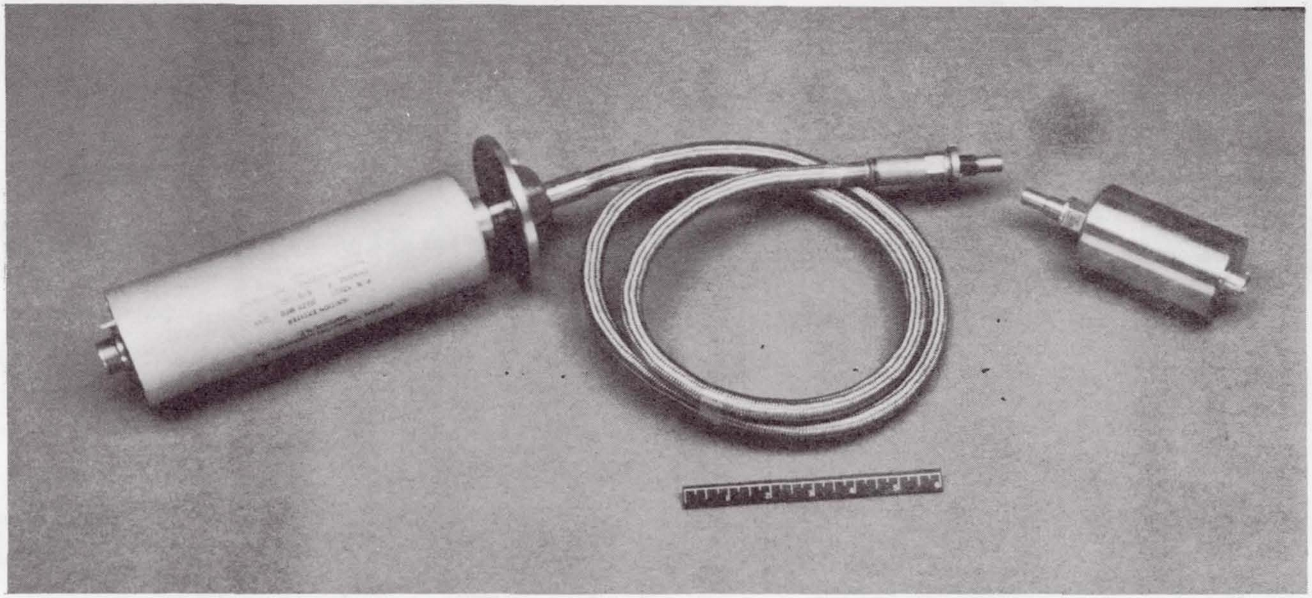
SPARK IGNITER GOALS

- LOW ELECTRICAL POWER
- ELIMINATE RFI
- LOW WEIGHT AND LOW MAINTENANCE

The spark igniter program goals have been achieved. An integrated plug/exciter has been designed, fabricated, and tested for the SS/APS application. It provides low input current and voltage, low R.F.I., improved reliability, and low maintenance. Thus, this design offers the solution to the only major problems of using the proven (man-rated) fuel-rich, augmented spark igniter for the Space Shuttle Auxiliary Propulsion System thrusters.

To best illustrate how well these goals have been met, a comparison is made between the J-2 and the SS/APS spark systems. Electrical power has been reduced (4.5 amperes vs 0.12 amperes). R.F.I. has been reduced (20 db over MIL-6-6181 to 3 db under). The weight has been reduced (6 lbs to 0.6 lbs). The pressurized 48-inch-long, high voltage cable has been eliminated and the SS/APS requires no scheduled maintenance.

INTEGRATED APS SPARK PLUG EXCITER
COMPARISON WITH J-2 SPARK SYSTEM



J-2		SS/APS
• 28	INPUT VDC	28
• 4.5 AMPERES	INPUT CURRENT	.12 AMPERES
• 360 MILIJOULES	STORED ENERGY	22 MILLIJOULES
• 50/SEC	SPARK RATE	200/SEC
• 6.0	WEIGHT, POUNDS	<0.6
• LARGE UNIT WITH 48 INCH HIGH VOLTAGE CABLE	RFI	SMALL WELL-SHIELDED UNIT
• 20 DB OVER MIL-L-6181		COMPLIES WITH MIL-1-6181
• 28%	ENERGY TRANSFER EFFICIENCY	70%
• 2%	SYSTEM EFFICIENCY	50%

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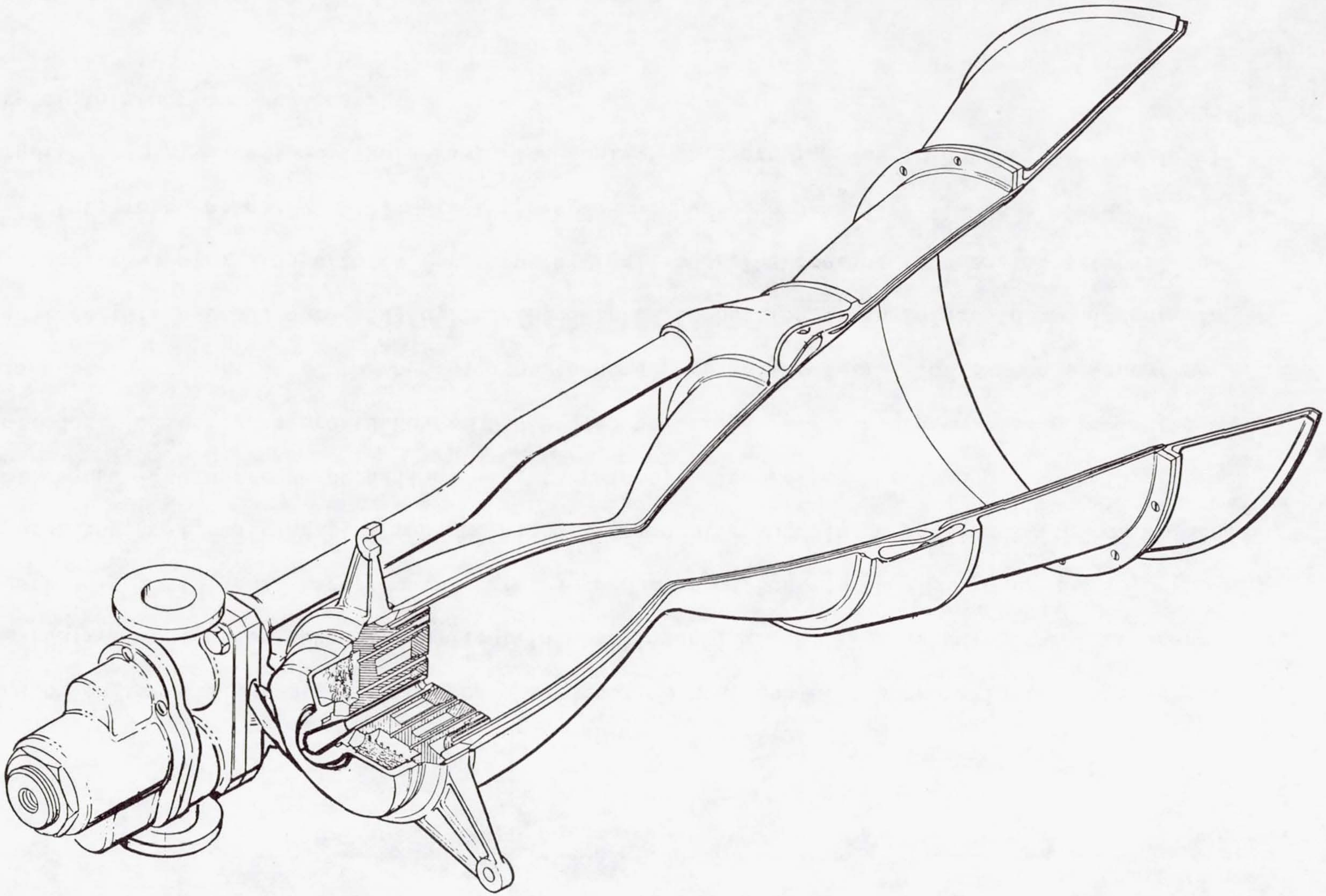
The SS/APS thruster assembly is shown with the auto-igniter installed. Main thruster propellant ignition is obtained by a torch (identical to the spark igniter). However, the initial ignition of the torch or igniter propellant is accomplished in a small preburner without the use of a spark plug or any other external energy. The energy for ignition is obtained from the high pressure propellants.

275-913
4-71

SPACE SHUTTLE APS IGNITION



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The auto-igniter employs a gas resonance phenomenon to heat one of the propellants. The igniter is basically two opposed tubes (propellant inlets) injecting into a larger diameter tube where combustion is sustained. Experimental efforts conducted on Rocketdyne IR&D in early 1970 indicated the most rapid heating and the highest temperatures could be obtained with low molecular weight gases. The hydrogen propellant was, therefore, selected for the lead gas. The hydrogen is introduced slightly before (0.005 to 0.010 sec) the oxygen. The hydrogen resonates in the oxygen tube and a portion of the hydrogen gas becomes heated (Mode I). The oxygen is then injected into this hot hydrogen and ignites (Mode II). Combustion is sustained in the larger diameter tube, thus providing a torch for thruster ignition. The concept requires no external power and, if integrated properly, will make the oxygen and hydrogen propellants appear virtually hypergolic from a systems point of view. The goal of this program was to demonstrate the feasibility of this unproven concept.

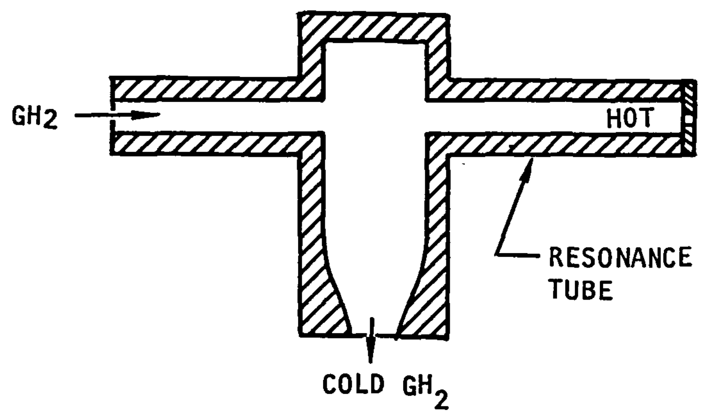
275-185B
3-71

O_2/H_2 AUTO IGNITER (RESONANCE)

729

MODE I

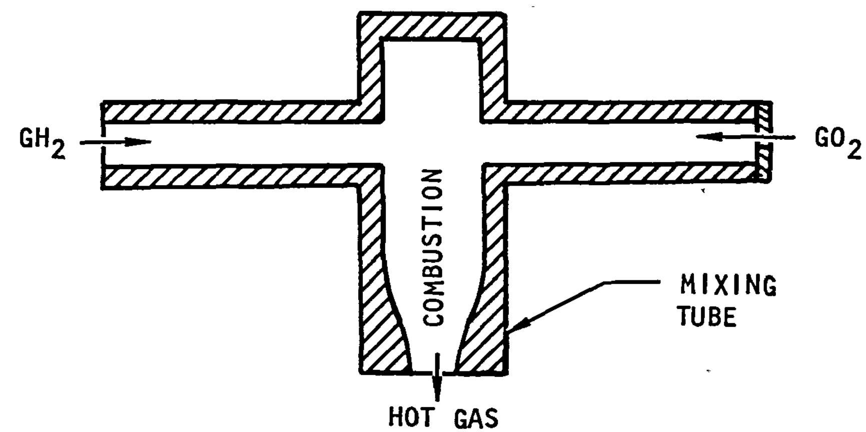
(HEATING)



- GH_2 ONLY FLOWING
- RESONANCE HEATING

MODE II

(COMBUSTION)



- GH_2 AND GO_2 FLOWING
- HOT GAS TORCH FOR THRUSTER IGNITION

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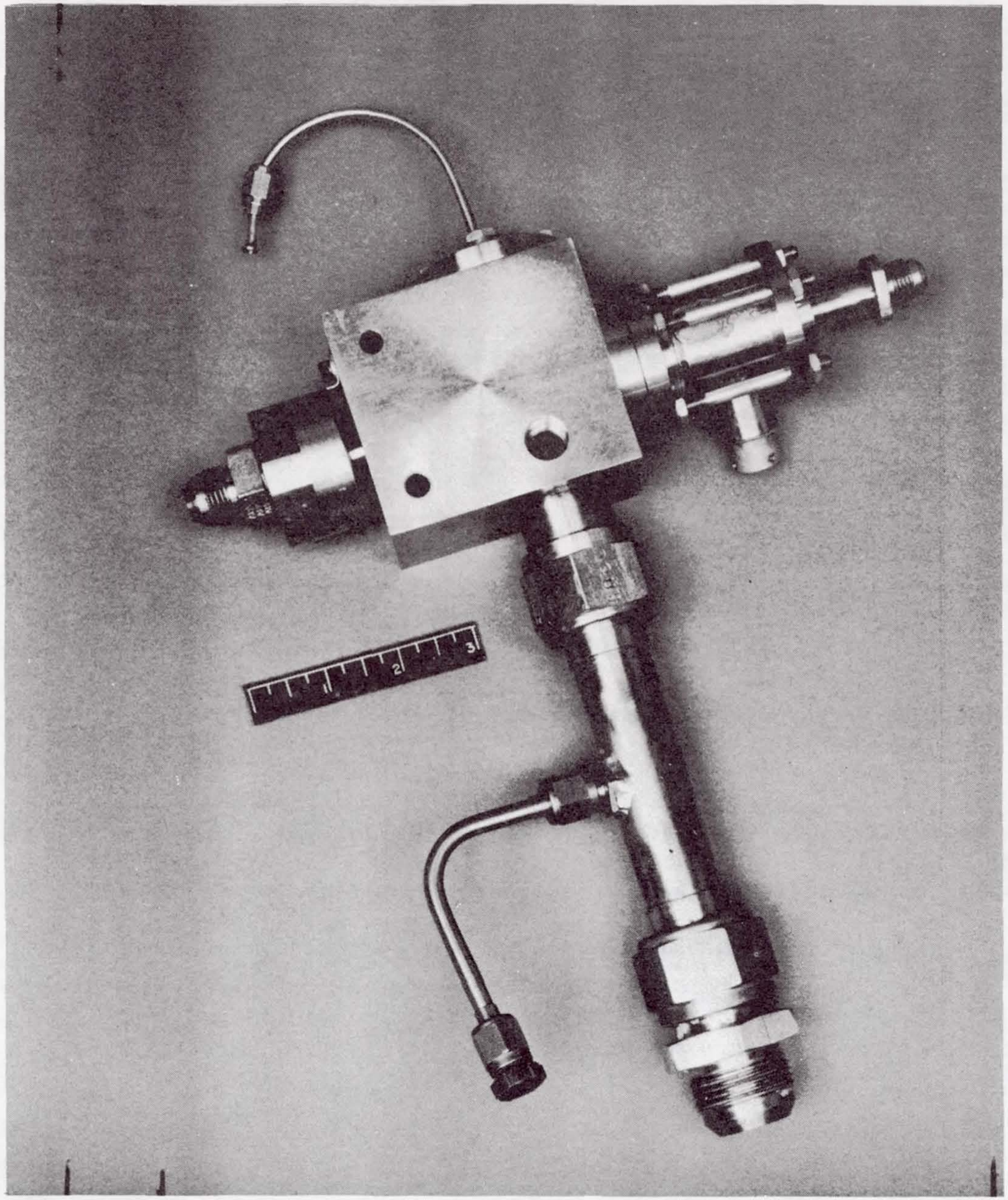
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

Tests were conducted using variable geometry test hardware. During the initial heating evaluation tests, hydrogen was injected into the resonance cavity. A fast-response thermocouple was placed at the end of the resonance cavity to measure temperature response and absolute value of temperature. Resonance cavity configurations and other igniter variables (gap ratio and pressure ratio) were evaluated over a range of inlet pressures and flowrates. The geometry was optimized. Temperatures up to 3000° R were recorded. Fast response was indicated by reaching indicated temperatures of 1000° F in approximately 20 msec. (Gas temperature response was less than that indicated by the thermocouple.)

275-887
3-71



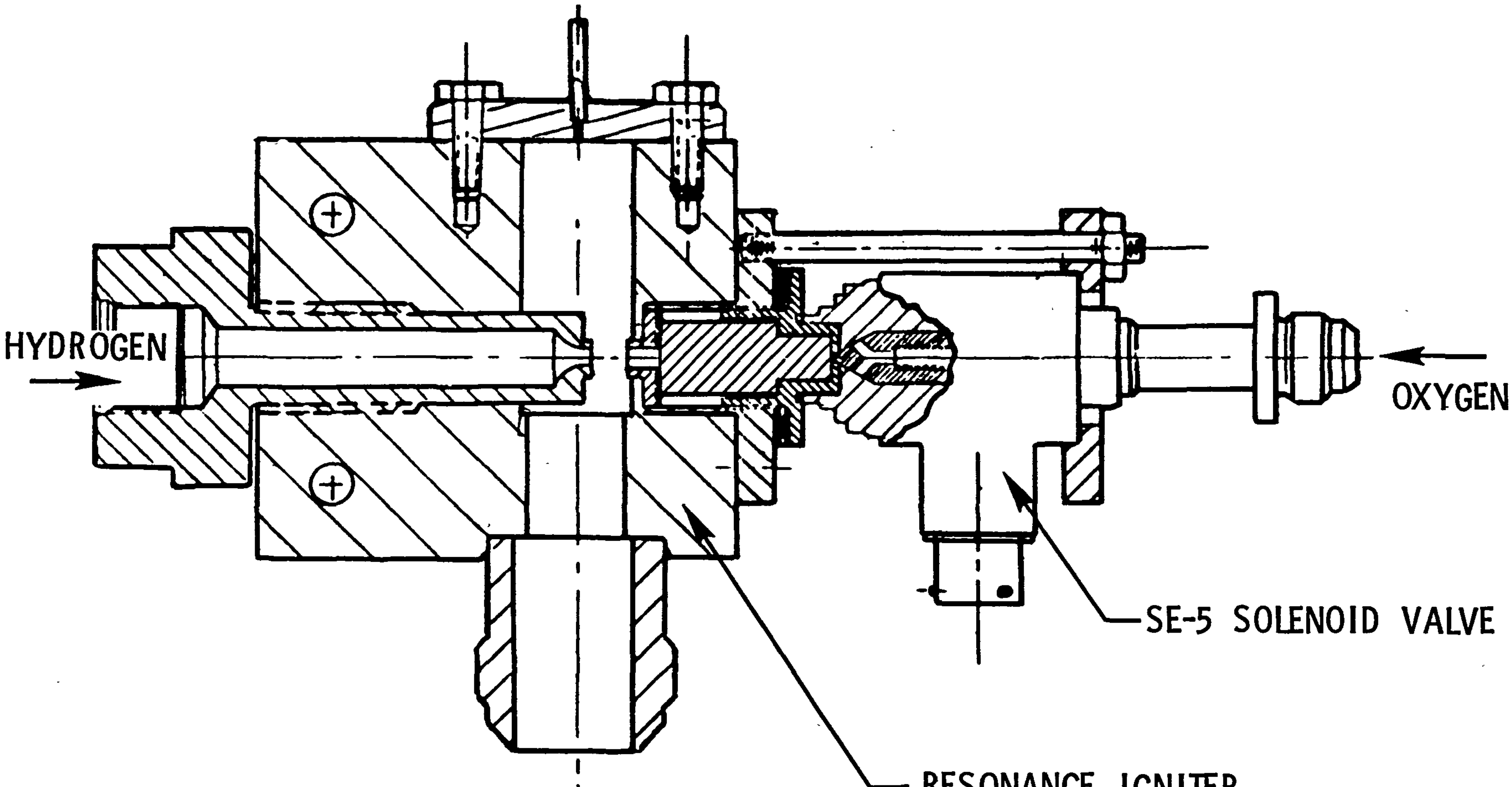
TEST HARDWARE

Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

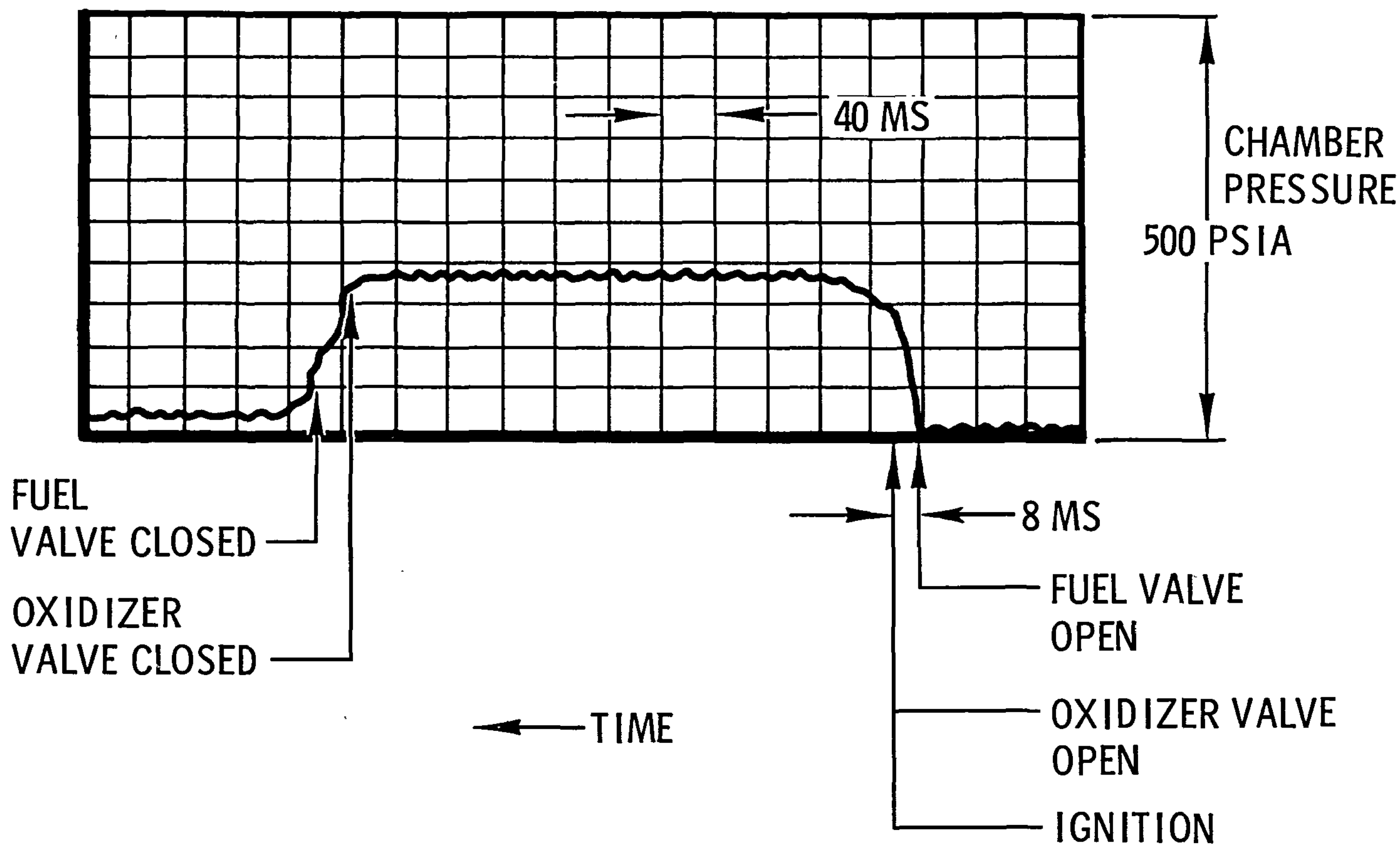
The variable geometry test hardware was also used for combustion tests with only minor modifications. An oxidizer valve was installed at the end of the resonance cavity where the thermocouple had been previously. 106 combustion tests were conducted. Hydrogen leads down to 3 msec resulted in ignitions. Response from electrical signal to 90 percent of igniter chamber pressure of from 0.020 to 0.030 sec were obtained. The basic feasibility of the concept was demonstrated with this workhorse-type hardware. These data were then used to design an igniter which would integrate into the SS/APS injector and thrust chamber.

TEST HARDWARE



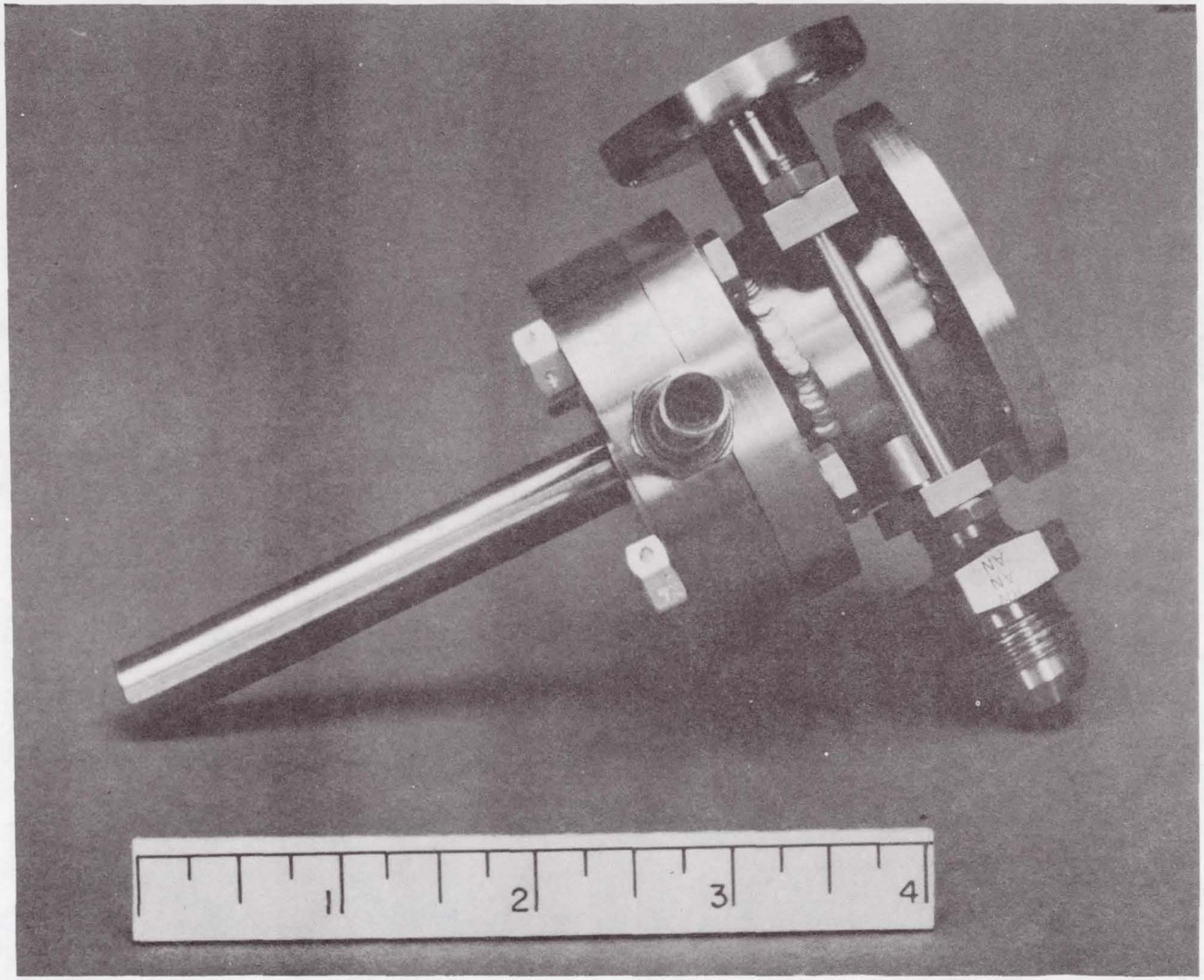
A test record of a typical start transient demonstrates the rapid response of the auto-igniter. Analysis of the chamber pressure trace indicates no ignition delay or pressure overshoot. The hydrogen lead (heating time) on this test was only 0.008 sec.

TEST RESULTS



Based upon the test data obtained with the variable geometry test hardware, a fixed geometry auto-igniter was designed and fabricated.

AUTO-IGNITER HARDWARE



737

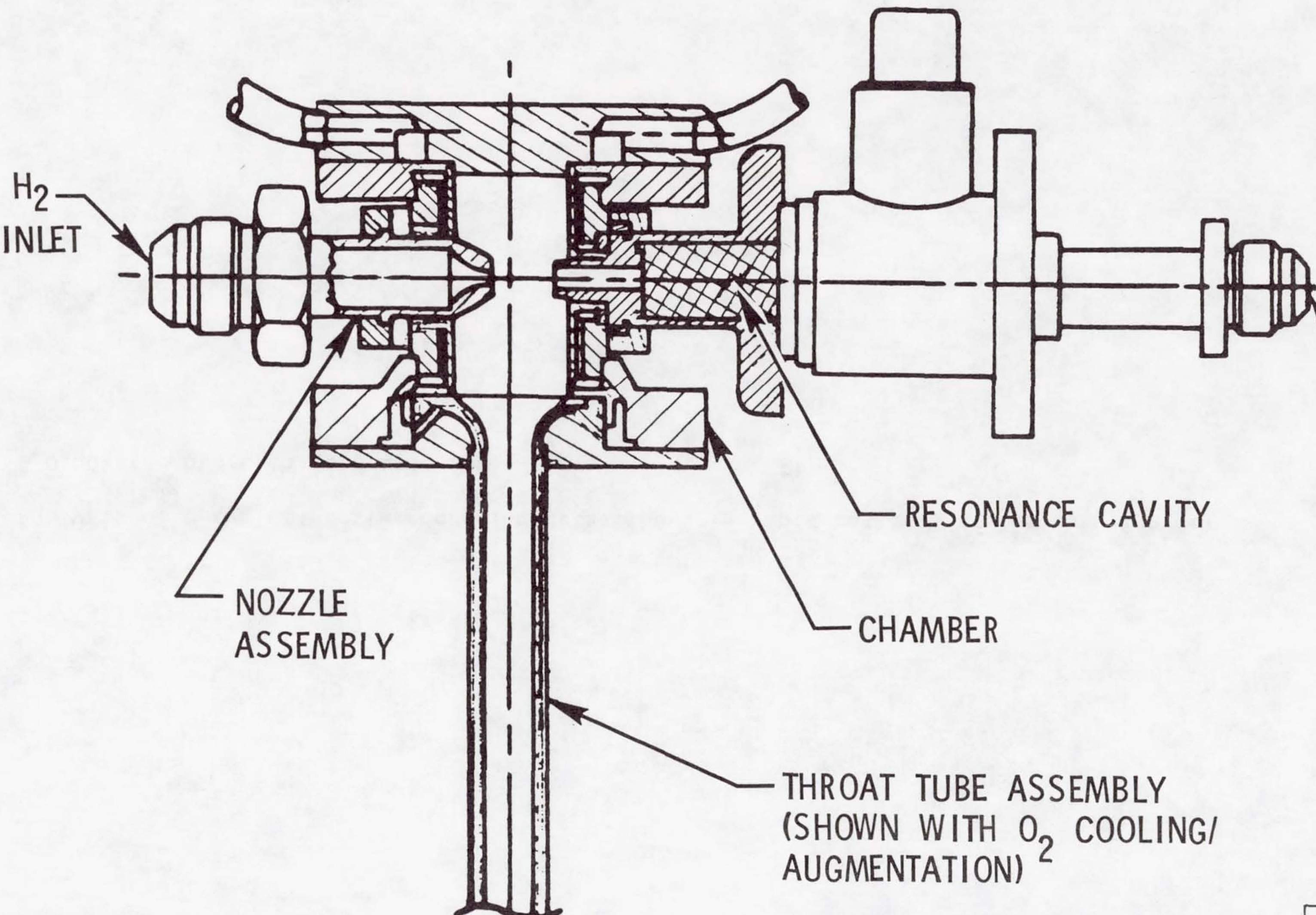
Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

The design is similar to the workhorse-type igniter. The resonance cavity and other critical dimensions are identical. Oxidizer is used to cool the igniter and is injected into the igniter exhaust at the thruster injector face plane. 194 combustion tests were conducted over a range of inlet conditions and flowrates. Tests were conducted successfully with hydrogen propellant temperatures down to 201° R and oxygen propellant temperatures down to 277° R.

AUTO-IGNITER HARDWARE

275-888
3-71



739

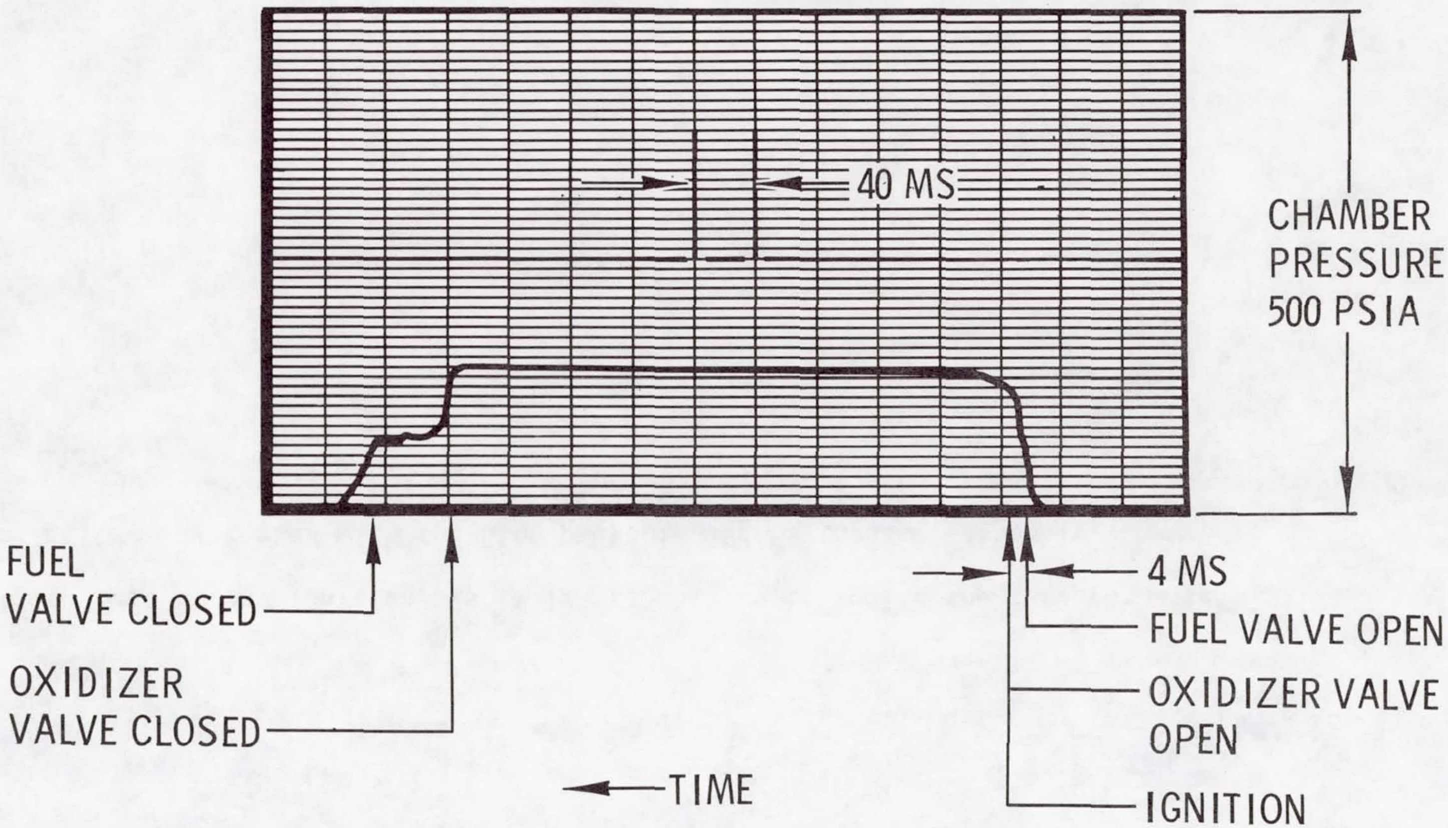
Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

Analysis of a typical test record indicates the very rapid response of the auto-igniter.

A 0.004 sec hydrogen lead was used.

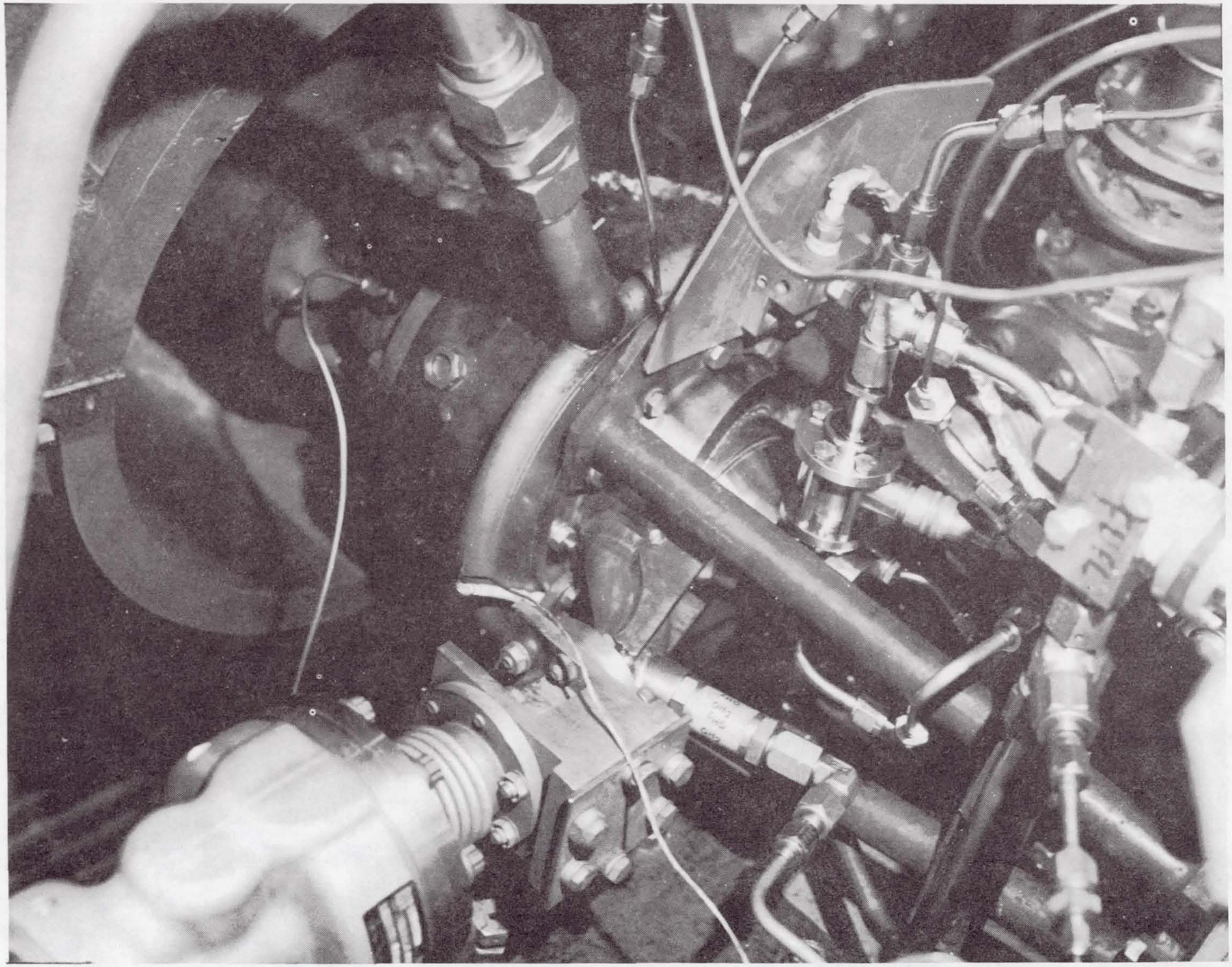
AUTO IGNITER TEST RESULTS



The ultimate test of any ignition system is its ability to ignite the thruster assembly.
The auto-igniter was tested with the high pressure SS/APS thruster assembly.

HIGH PRESSURE THRUSTER IGNITION
(AUTO-IGNITER)

275-901
3-71



Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

This test program was initiated with eight "igniter only" tests at sea level conditions to evaluate sequencing (all previous tests were conducted on an "igniter only" facility at altitude conditions). All eight tests resulted in rapid ignitions. A series of six tests was then conducted with ambient propellants. All tests resulted in successful thruster ignitions. The final series of tests was successfully conducted with cold propellants (250° R hydrogen and 375° R oxygen). During these 14 tests with conditioned thruster propellants, igniter mixture ratio and flowrates were varied.

TEST SUMMARY

- 28 TESTS CONDUCTED
- RAPID RESPONSE
- SEA LEVEL AND ALTITUDE OPERATION DEMONSTRATED

SUMMARY

The auto-igniter technique employing resonance heating has been systematically investigated. Variable geometry test hardware was used to optimize igniter geometry. An auto-igniter was tested in an "igniter only" facility to explore operational limits. Complete thruster assembly tests were conducted at simulated altitude conditions to demonstrate the feasibility of the auto-igniter to ignite the SS/APS thruster without external power and without catalytic agents.

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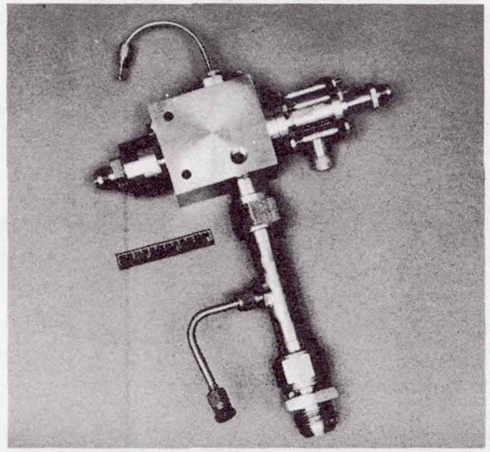
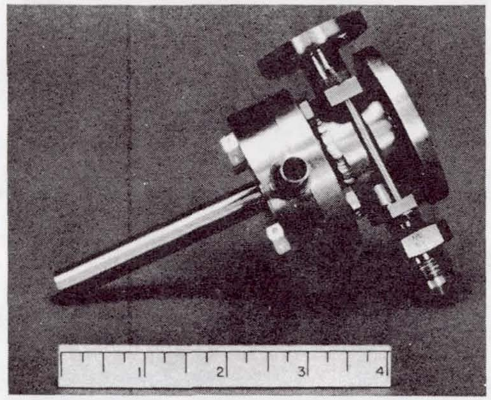
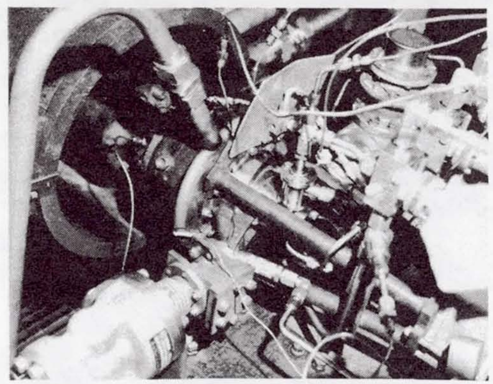
AUTO-IGNITER HARDWARE







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SUMMARY

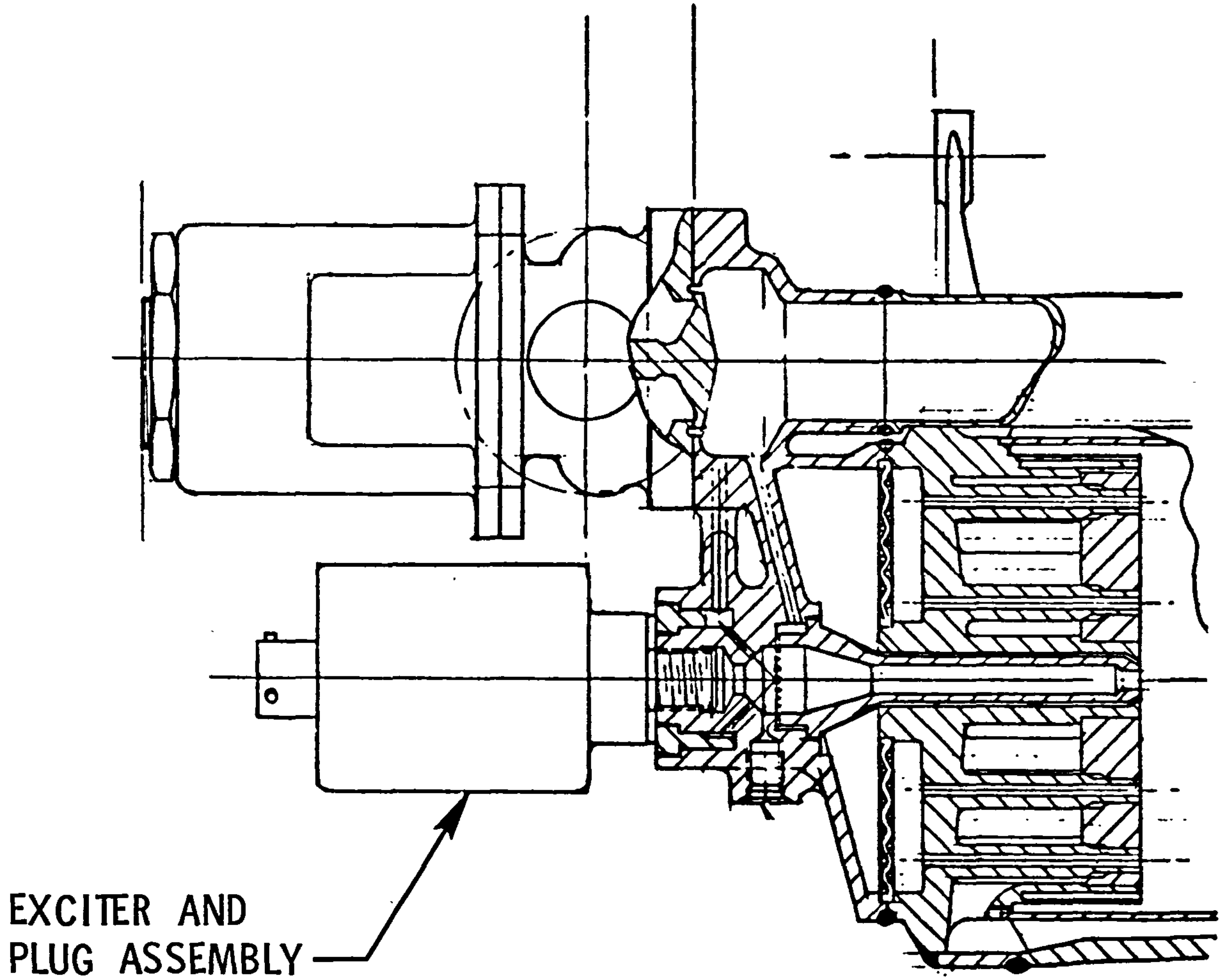
Auto-igniter technique employing resonance heating has been systematically investigated. It has progressed from a conceptual idea to a demonstrated feasible SS/APS ignition technique. The basic principles of operation and design variables have been investigated experimentally. The current igniter design has been optimized for ambient temperature propellants. Its operation at this design point has been exceptional. Further efforts to optimize this igniter for low temperature and incorporate additional design improvements appear warranted.

AUTO-IGNITER SUMMARY

- HEATING AND RAPID RESPONSE DEMONSTRATED
- GEOMETRY EFFECTS INVESTIGATED
- IGNITION OBTAINED AND COMBUSTION SUSTAINED
- OPERATIONAL LIMITS EXPLORED (TEMP. AND PRESSURE)
- THRUSTER IGNITIONS DEMONSTRATED
- AUTO-IGNITER FEASIBILITY DEMONSTRATED

A preliminary ignition system design effort is currently being conducted. This effort is scheduled to be completed 30 April. A spark igniter system is shown closely integrated with the thruster assembly. The igniter propellants are manifolded to the igniter immediately downstream of the main propellant valves. A single integrated exciter/plug unit is shown. A second unit can be installed if reliability analysis indicated a requirement for redundancy.

PRELIMINARY SPARK IGNITION SYSTEM DESIGN
IGNITION ONLY



CHAMBER PRESSURE

If the igniter is required to perform the A.C.S. function (very small impulse bits), separate igniter valves and higher flowrates are required. This igniter would provide approximately 50 to 100 lbs thrust at the overall mixture ratio of 4.0. The exciter and plug unit is shown mounted in the alternate size location. At this location, two units can be used 180° apart and provide redundancy.

PRELIMINARY SPARK IGNITION SYSTEM DESIGN
IGNITION AND PULSE MODE

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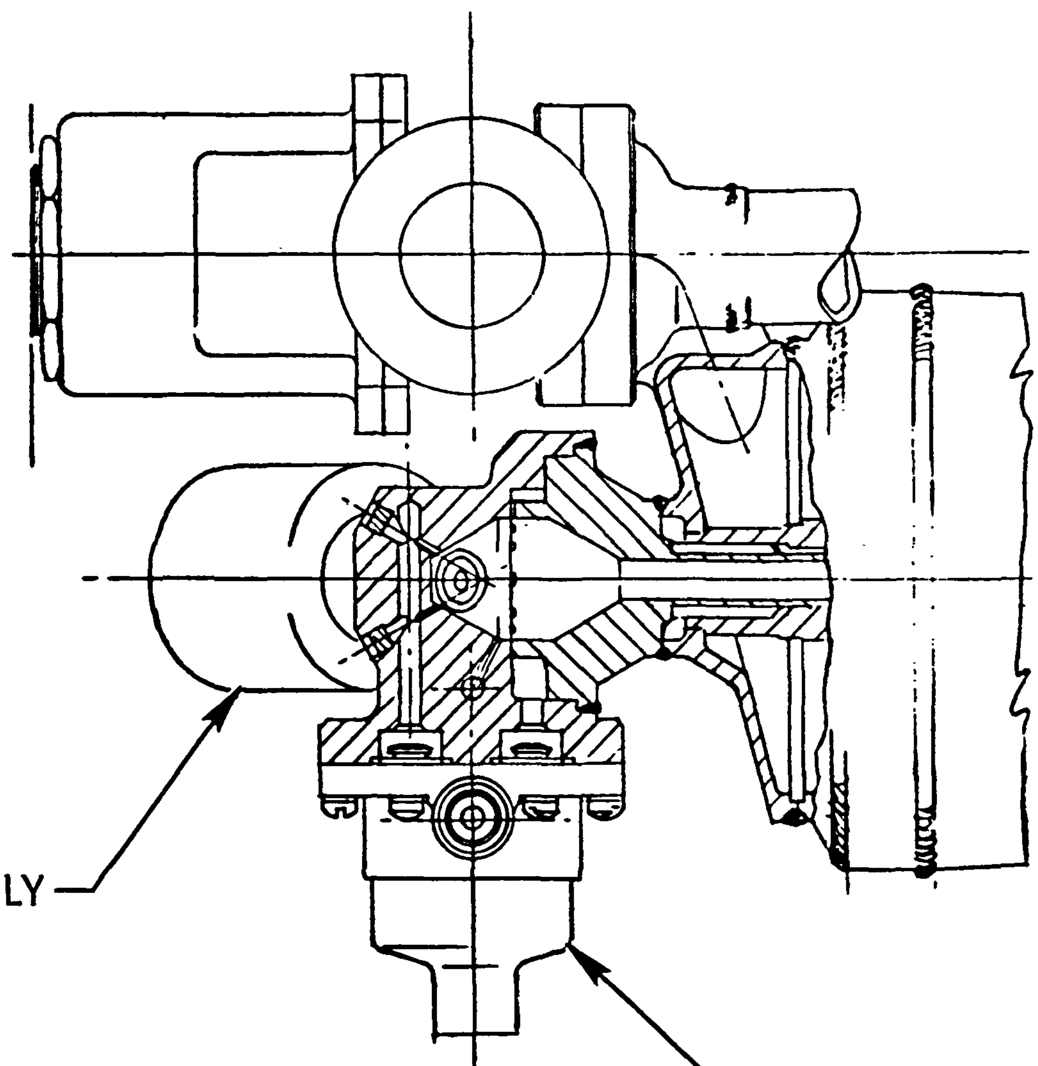

A preliminary auto-igniter system design is also being made for the high pressure SS/APS thruster application. For the "ignition only" application, no electrical power is required. The propellants are manifolded to the auto-igniter immediately downstream of the main propellant valves. Manifold volumes and volumetric flowrates will provide a short hydrogen lead to the auto-igniter. A check-valve is employed at the end of the resonance cavity on the current designs. Advanced resonance igniter designs are currently being evaluated which will eliminate the need for the check-valve and will potentially operate with any propellant sequencing.

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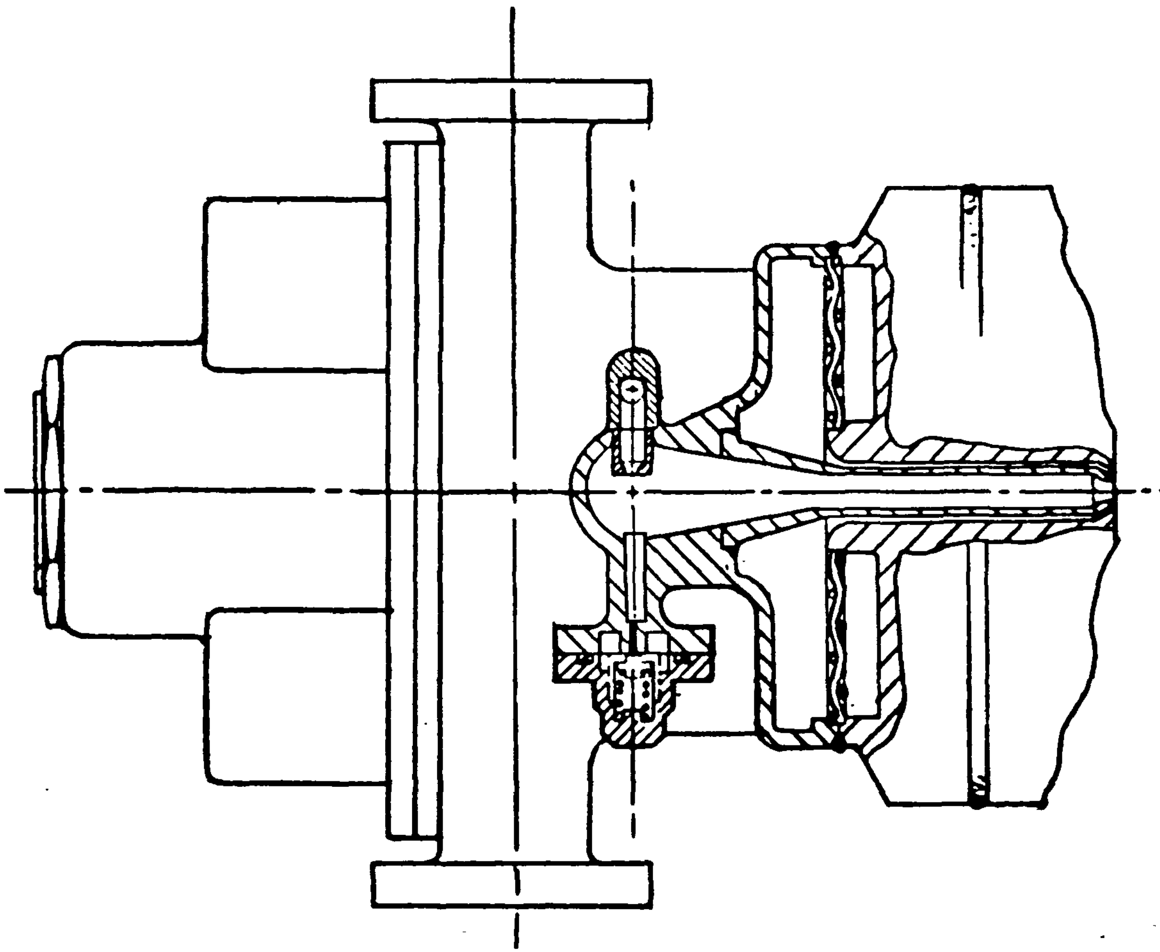
PRELIMINARY AUTO-IGNITION SYSTEM DESIGN

IGNITION ONLY



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If the igniter is required to perform the A.C.S. function (very small impulse bits), separate igniter valves and higher flowrates are required. The igniter provides 50 to 100 lb of thrust at an overall mixture ratio of 4.0. The only electrical power required for this system is for valve actuation. This configuration with close-coupled valves is virtually identical to that evaluated experimentally on this program.

PRELIMINARY AUTO-IGNITION SYSTEM DESIGN

IGNITION AND PULSE MODE

275-896
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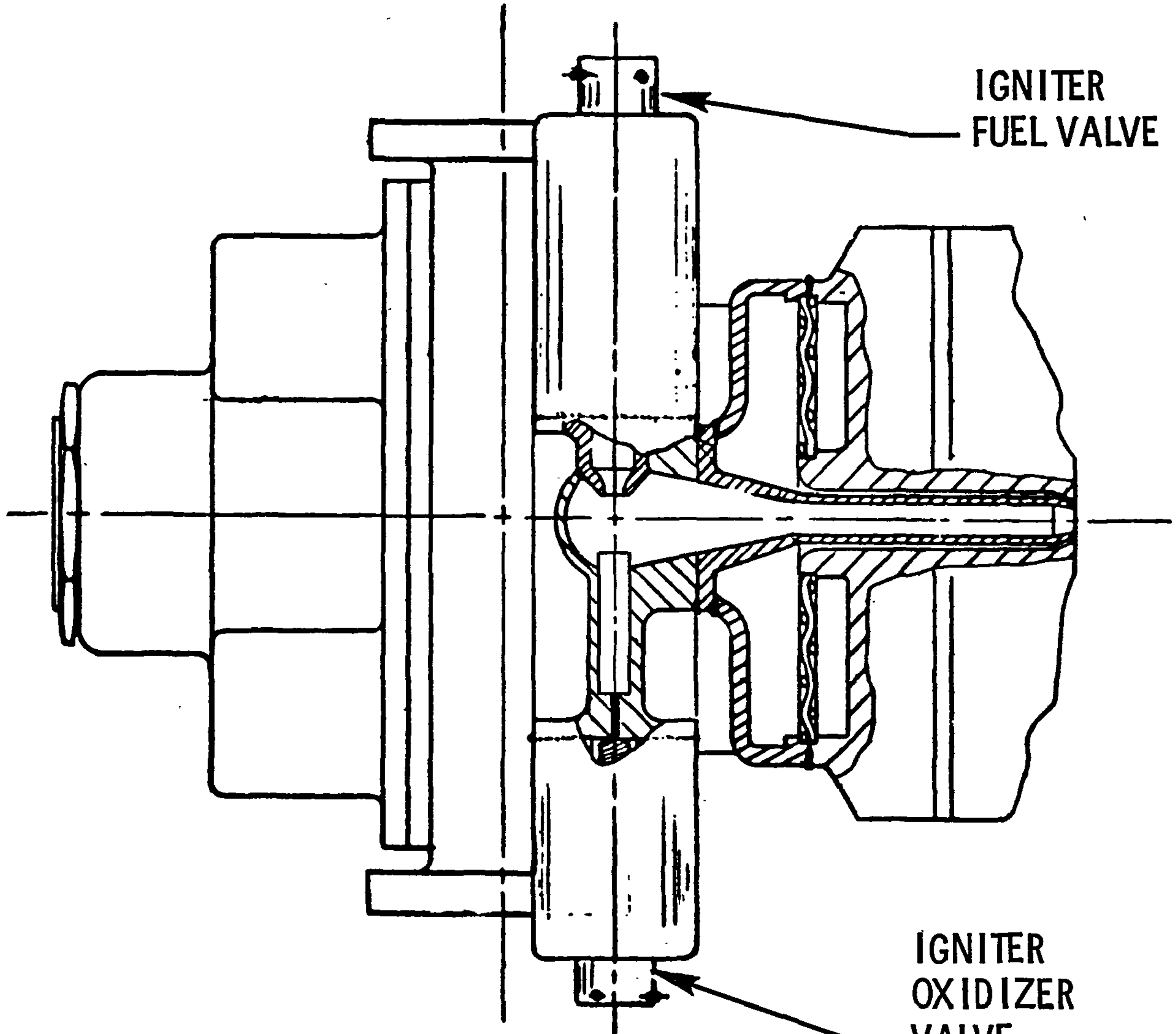

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CONCLUSIONS AND RECOMMENDATIONS

All program objectives have been met. The feasibility of an advanced spark igniter system has been demonstrated. Very significant improvements have been made in the electrical equipment. Electrical power requirements and system weight have been drastically reduced. R.F.I. and scheduled maintenance have been eliminated. The feasibility of the auto-igniter to provide reliable and rapid SS/APS thruster ignitions has been demonstrated. The effects of igniter geometry and the operational limits have been explored. Additional efforts to further evaluate the flight-type igniters of both the spark and auto-igniter type are recommended.

"CATALYTIC IGNITION/THRUSTER INVESTIGATION"

R. J. JOHNSON

TRW

TECHNICAL MANAGER

P. N. HERR

LEWIS RESEARCH CENTER

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HYDROGEN-OXYGEN CATALYTIC IGNITION AND THRUSTER INVESTIGATION

TRW SYSTEMS GROUP, TRW INC.

R. J. JOHNSON, PROJECT ENGINEER

CONTRACT NAS 3-14347

In support of NASA Lewis Research Center Space Shuttle Auxiliary Propulsion Engine Technology, TRW Systems Group has been performing analyses and test firings of gaseous hydrogen-oxygen igniters and thrusters since June of 1968 (NAS 3-11227). The program presently being conducted is an experimental and supporting analytical evaluation of catalyst bed operational limitations, igniter scaling criteria, and delivered performance for a flightweight gaseous hydrogen-oxygen thruster. Performance of this contract was initiated in July of 1970. The program objectives, results of the tasks performed to date, and the remaining tasks to be completed are described in this presentation.

PROGRAM OBJECTIVES

- ESTABLISH CATALYST OPERATIONAL LIFE, BOTH STEADY-STATE AND PULSE-MODE
- DEVELOP IGNITER FLASHBACK PREVENTION CRITERIA
- INVESTIGATE METHODS OF IGNITER RESPONSE ENHANCEMENT
- PROVIDE GENERALIZED DESIGN GUIDELINES FOR CATALYTIC IGNITION OF H_2/O_2 THRUSTERS
- EVALUATE OVERALL PERFORMANCE OF A COOLED FLIGHTWEIGHT H_2/O_2 THRUSTER

The first program task was to determine the catalyst operational life with gaseous hydrogen-oxygen propellants for both steady-state and pulse-mode firings. Tests were performed under simulated altitude conditions at chamber pressures of 100 psia and reaction temperatures of 1800°F. The reactor test hardware fabricated during the performance of NAS 3-11227 was to be utilized for these tests. This hardware was designed to accommodate catalyst beds of various length, and is instrumented for pressure and temperature measurements upstream, downstream, and within the catalyst bed.

CATALYST LIFE EVALUATION TESTS

- DETERMINE OPERATIONAL LIFE OF SHELL 405-ABSG AND ENGELHARD MFSA CATALYSTS
- STEADY STATE LIFE TESTS - EACH CATALYST

  4000 SECONDS CONTINUOUS FIRING, 100 PSIA CHAMBER PRESSURE
  AMBIENT PROPELLANTS, 1800°F REACTION TEMPERATURE

- CYCLIC LIFE TESTS - 5000 PULSES EACH

  SHELL 405-ABSG, ENGELHARD MFSA: 70°F PROPELLANTS, REPEATED REACTION TO 1800°F, FOLLOWED BY EXTERNAL COOLING TO 1000°F

  SHELL 405-ABSG: -250°F PROPELLANTS, REPEATED REACTION TO 1800°F, FOLLOWED BY EXTERNAL COOLING TO -250°F

- DETERMINE EXTENT OF PHYSICAL DAMAGE, VOLUME LOSS, AND LOSS OF ACTIVITY OF THE CATALYSTS, USING TECHNIQUES DEVELOPED IN NAS 3-11227.

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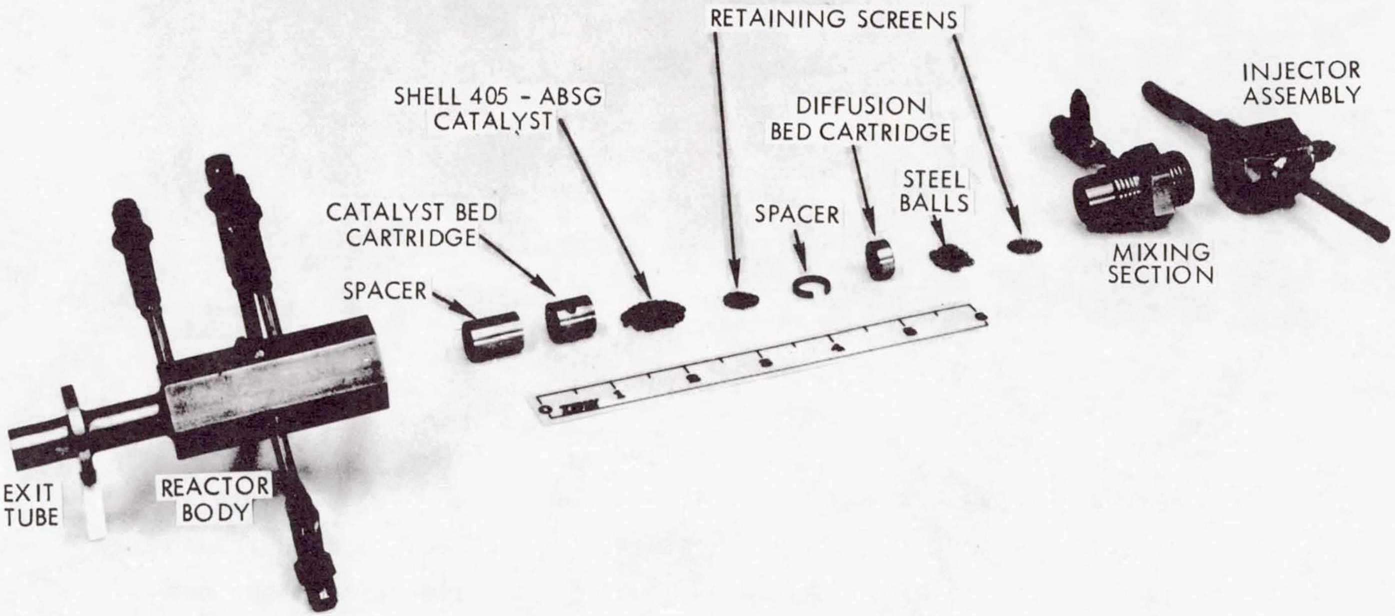
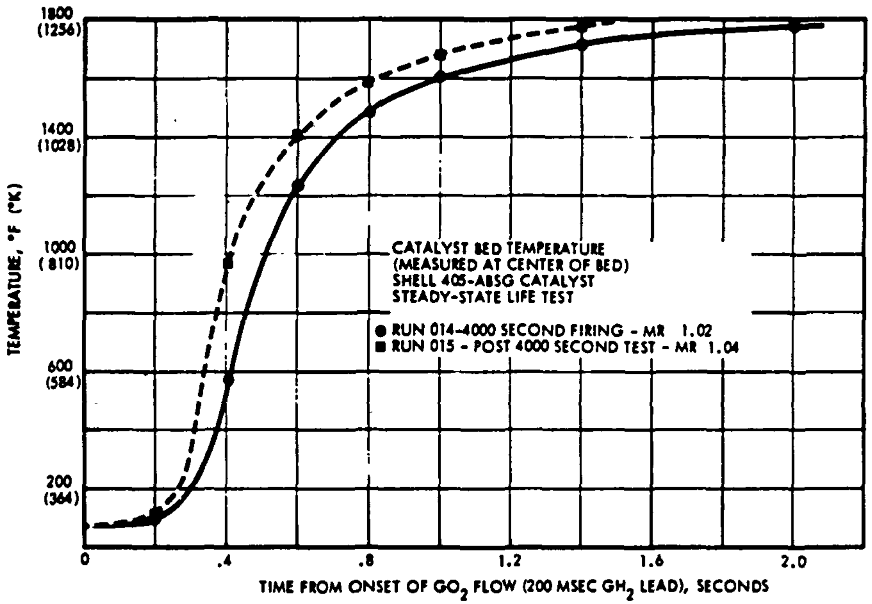
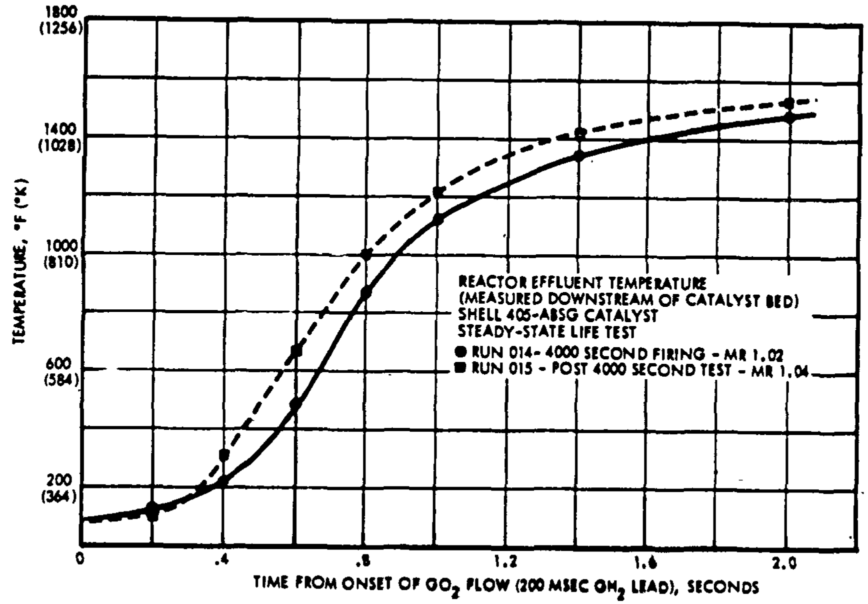


Disassembled Catalytic Reactor Igniter - High Chamber Pressure

Continous 4000 second firings were completed with both the Shell and Engelhard catalysts. Immediately following each long-duration firing, after cooldown of the catalyst bed to ambient temperature, a 20-second firing was conducted to determine if the thermal response of the catalyst bed had been affected by the extended duration firing. These figures compare the catalyst bed and reactor effluent thermal response before and after 4000 seconds of operation with the Shell catalyst. The results indicate that the long duration firings had no detrimental effect on the reactor thermal response with ambient temperature propellants.



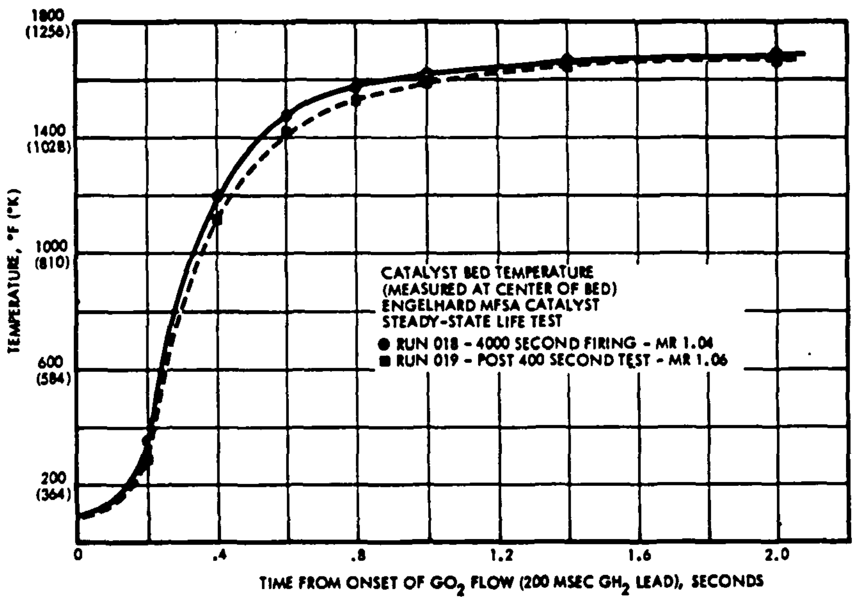
Catalyst Bed Response Data - Shell Catalyst Steady-State Life Tests



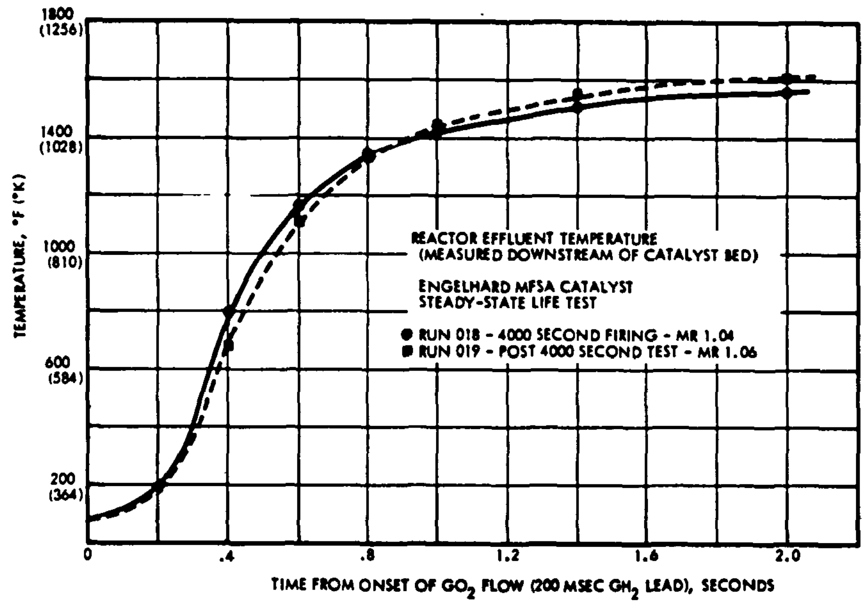
Reactor Effluent Response Data - Shell Catalyst Steady-State Life Tests

Data from the Engelhard catalyst steady-state life tests also indicate that 4000 seconds of steady-state operation did not reduce the capability of the catalyst to initiate reaction with gaseous hydrogen-oxygen propellants.

Visual examination of each catalyst load after the 4000 second firings revealed no crumbling or erosion of the catalyst pellets. No apparent differences could be seen between the fired and "as received" catalyst of each type. Measured weight loss of each catalyst bed was less than five percent of the initial bed load, and less than one-tenth of a gram in each case.



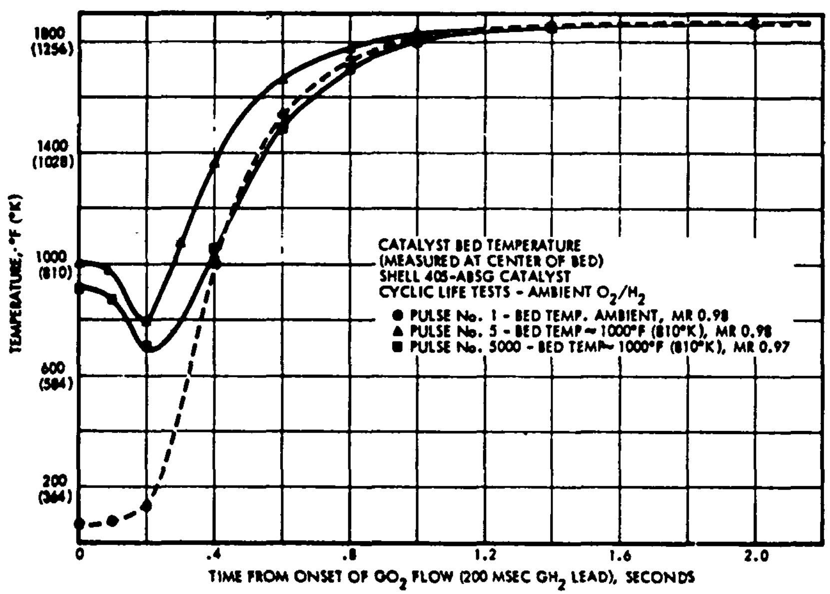
Catalyst Bed Response Data - Engelhard Catalyst Steady-State Life Tests



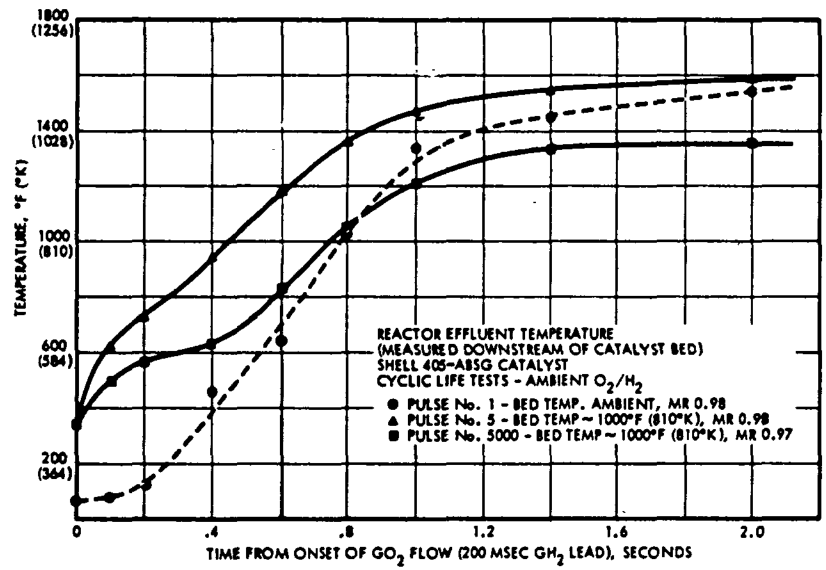
Reactor Effluent Response Data - Engelhard Catalyst Steady-State Life Tests

Ambient temperature propellant cyclic life evaluations were performed with cooldown of the catalyst bed to 1000°F or below between pulses. Data from the Shell catalyst cyclic life tests compare the catalyst bed and reactor effluent temperature response before and after 5000 pulses (15,000 seconds total firing time). It should be noted that the life test reactor hardware and catalyst bed configurations were not designed for optimum bed response, and that these test results are not indicative of the minimum attainable catalyst bed thermal response times.

The data indicate that some degradation of catalyst bed response and reduction in reactor effluent temperatures occurred after 5000 cycles of operation; however, the catalyst was still able to support the reaction of ambient temperature hydrogen-oxygen.



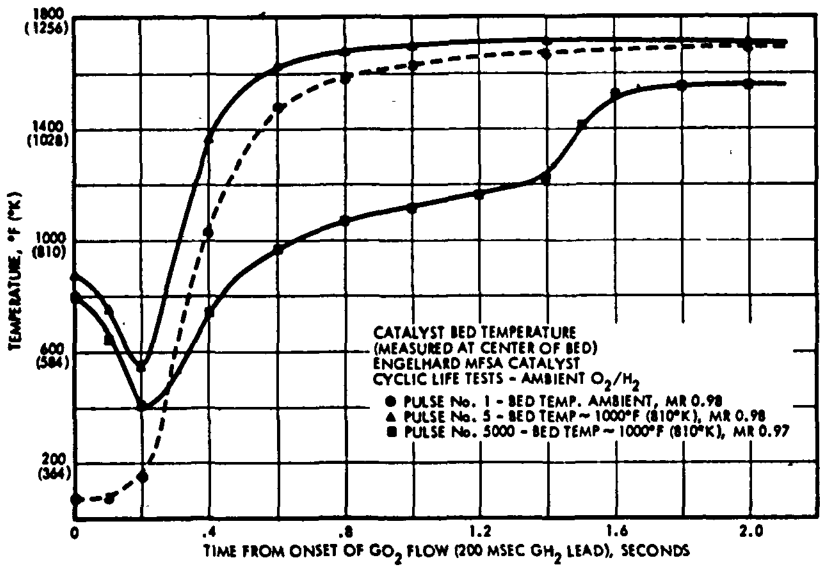
Catalyst Bed Response Data -
Shell Catalyst Cyclic Life Tests



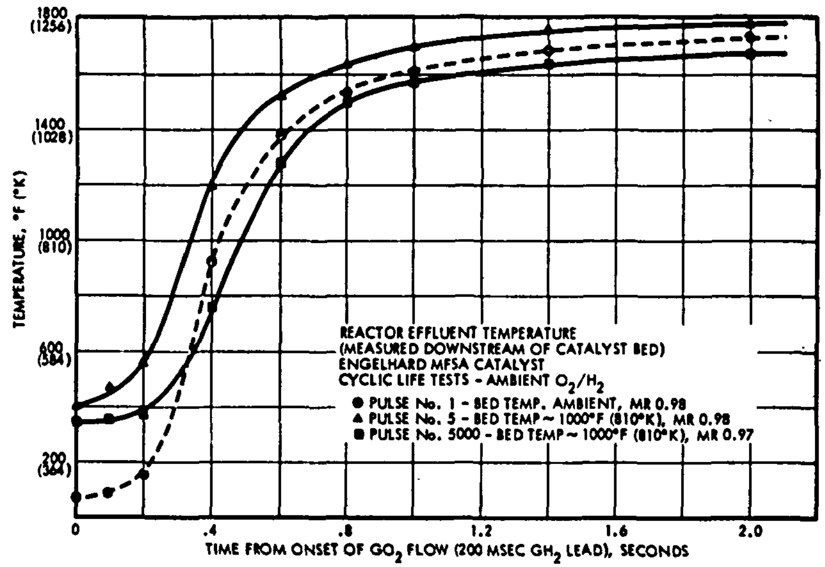
Reactor Effluent Response Data -
Shell Catalyst Cyclic Life Tests

Results of the cyclic life tests with the Engelhard catalyst also indicate that some degradation in bed response occurred after 5000 pulses, although the effluent temperature measured on the last pulse was not appreciably reduced.

Visual examination of each catalyst load after the cyclic life tests revealed slight erosion of the catalyst pellets. Some sintering together of the Shell catalyst was observed, although most of the catalyst bed remained as individual pellets. Weight loss of each catalyst bed was approximately seven percent.



Catalyst Bed Response Data -
Engelhard Catalyst Cyclic Life Tests



Reactor Effluent Response Data -
Engelhard Catalyst Cyclic Life Tests

After completion of the steady-state and ambient propellant cyclic life tests, each catalyst bed was subjected to a series of laboratory tests to determine the total and active surface area. A constant volume gas adsorption apparatus was used to measure total surface areas by the physical adsorption technique of Brunauer, Emmett, and Teller (BET Theory)*.

This table shows the results of the total surface area measurements on both the Shell and Engelhard catalysts. The results show that the Shell catalyst lost a greater portion of its total surface area than the Engelhard catalyst during the 4000 second steady-state firings. The 5000 pulse life tests (15,000 seconds firing time) resulted in a loss of nearly 90% of the surface area of the Shell catalyst, while the Engelhard catalyst retained about 50% of its original surface area.

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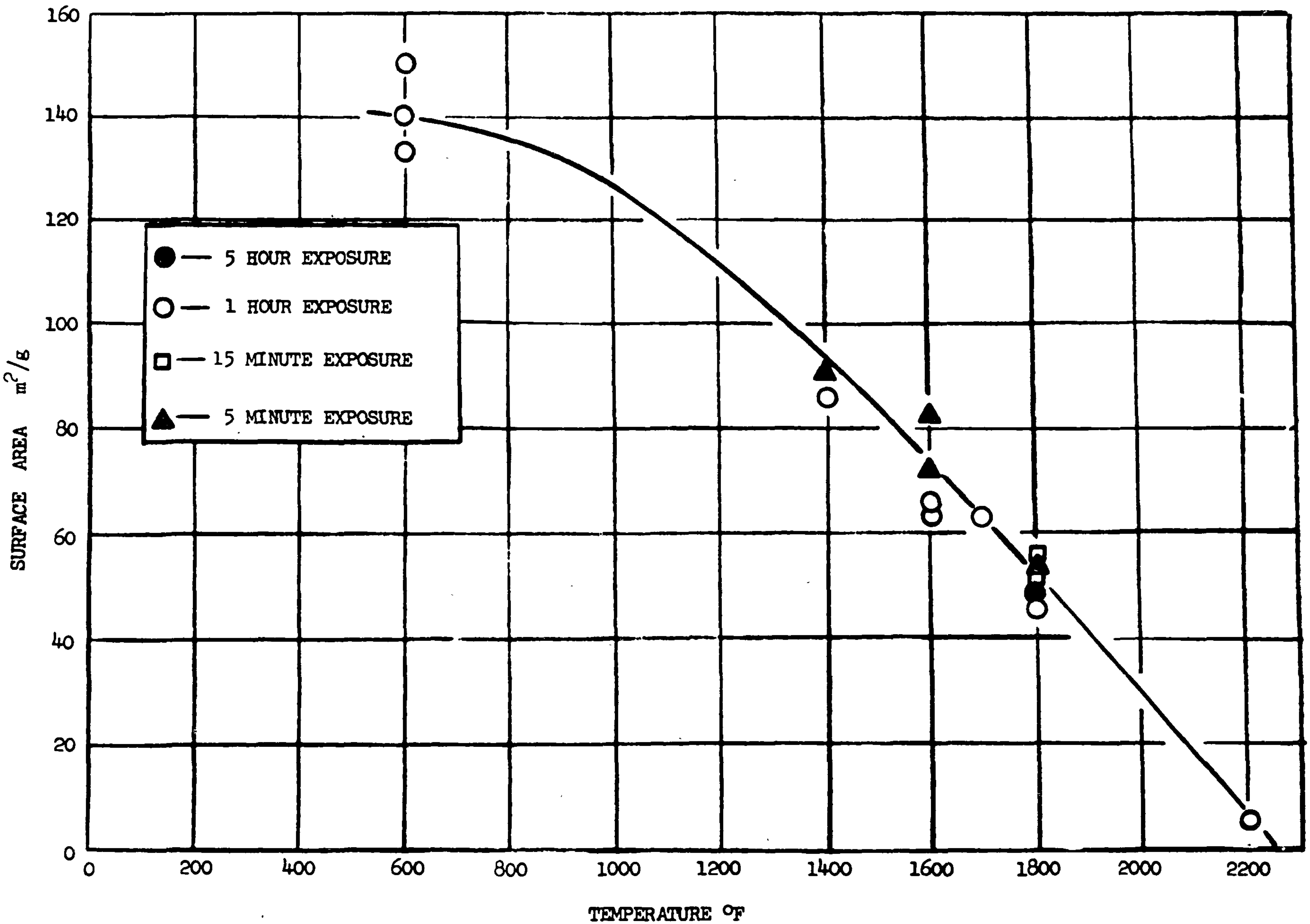
*Brunauer, S., Emmett, P.H., and Teller, E.J., J. American Chemical Society, 60, pp. 309, (1938)

SUMMARY OF CATALYST SURFACE AREA MEASUREMENTS

CATALYST	TOTAL SURFACE AREA - SQUARE METERS/GRAM		
	AS RECEIVED	AFTER 4000 SEC STEADY-STATE FIRING	AFTER 5000 PULSES (15,000 SEC ON TIME)
SHELL 405-ABSG	112	40	15
ENGELHARD MFSA	180	93	85

The data in this figure was taken from previous TRW sponsored catalysis research with Shell 405, which is also used for decomposition of hydrazine. These results indicate that the loss of surface area is strongly dependent on operating temperature, and is not affected by length of exposure time from five minutes to at least five hours.

The Shell catalyst from the 4000 second steady-state life test was operated at bed temperatures up to 1870°F and had a post-test measured surface area of 40 sq. meters/gram, which agrees very well with the data in this figure. Measured surface area after the 5000 pulse tests (15,000 seconds or about 4 hours firing time) with the Shell catalyst was 15 sq. meters/gram, indicating that pulse-mode operation has more effect on surface area loss of the Shell catalyst than continuous reaction.



Surface Area vs. Temperature for Short Term Exposure of Shell 405 Catalyst

Under NASA Contract NAS 7-520, it was found that the determination of adsorption isotherms of hydrogen on catalysts provide a very sensitive measure of the amount of active surface metal available to catalyze chemical reactions. For this contract, chemisorption measurements provided a simple and rapid method of determining what effect the life test firings had on the active catalyst surface. This table is a summary of the hydrogen chemisorption data for both the Shell 405 and Engelhard MFSA catalysts. Since the amount of hydrogen chemisorption was proportional to the amount of active surface area, made up of active surface metal available to promote the reaction, a like amount of reduction in active surface metal occurred. The table also reveals that a severe loss in active surface area of the Shell catalyst occurred during the cyclic life tests, while the Engelhard catalyst was not as significantly affected, having much less active metal content originally than the Shell catalyst. These results are consistent with the measured changes in catalyst total surface area, as listed in the previous table.

SUMMARY OF HYDROGEN CHEMISORPTION ISOTHERM EXPERIMENTS

CATALYST	SAMPLE HISTORY	EQUILIBRIUM HYDROGEN ADSORPTION AT 473°K AND 200 TORR (MM. HG) HYDROGEN ATOMS X 10^{-18}
SHELL 405-ABSG	AS RECEIVED	415 $\pm$ 5*
SHELL 405-ABSG	AFTER 4000 SEC RUN	128 $\pm$ 8
SHELL 405-ABSG	AFTER 5000 PULSES (15,000 SEC ON TIME)	9.7 $\pm$ 1.0
ENGELHARD MFSA	AS RECEIVED	22.3 $\pm$ 3.0
ENGELHARD MFSA	AFTER 4000 SEC RUN	14.6 $\pm$ 1.5
ENGELHARD MFSA	AFTER 5000 PULSES (15,000 SEC ON TIME)	11.3 $\pm$ .4

*DUPLICATE EXPERIMENTS WERE RUN FOR EACH OF THESE CASES AND THE LISTED UNCERTAINTY REPRESENTS THE SPAN BETWEEN THE EXPERIMENTS.

The steady-state and cyclic life tests with ambient temperature propellants were successfully completed with each catalyst type. Test results indicated that both catalysts were capable of continued operation at the completion of the life test firings.

Cyclic tests were conducted with the Shell 405-ABSG catalyst at propellant and initial bed temperatures of -250°F (117°K). The objective of this test series was to complete 5000 pulses of the reactor, during which the catalyst would be heated to 1800°F and then cooled by external means to -250°F prior to the next pulse. Pulse ignitions were achieved at -250°F propellant and bed temperatures for 700 successive pulses; however, fatigue cracks developed in the reactor wall between the catalyst bed and the liquid nitrogen cooling manifold. Completion of this task has been rescheduled to allow for modification of the reactor hardware.

CONCLUSIONS FROM CATALYST LIFE TESTS

- BOTH THE SHELL 405 ABSG AND THE ENGELHARD MFSA CATALYSTS EFFECTIVELY PROMOTE THE REACTION OF GASEOUS HYDROGEN AND OXYGEN FOR STEADY-STATE RUN DURATIONS OF AT LEAST 4000 SECONDS.

- AT LEAST 5000 PULSES WITH AMBIENT TEMPERATURE PROPELLANTS CAN BE ACHIEVED WITH EITHER CATALYST, ALTHOUGH LOSS OF SURFACE AREA MAY AFFECT LOW TEMPERATURE REACTION.

- SATISFACTORY REACTION CAN BE ATTAINED WITH THE SHELL 405-ABSG CATALYST AT -250°F.

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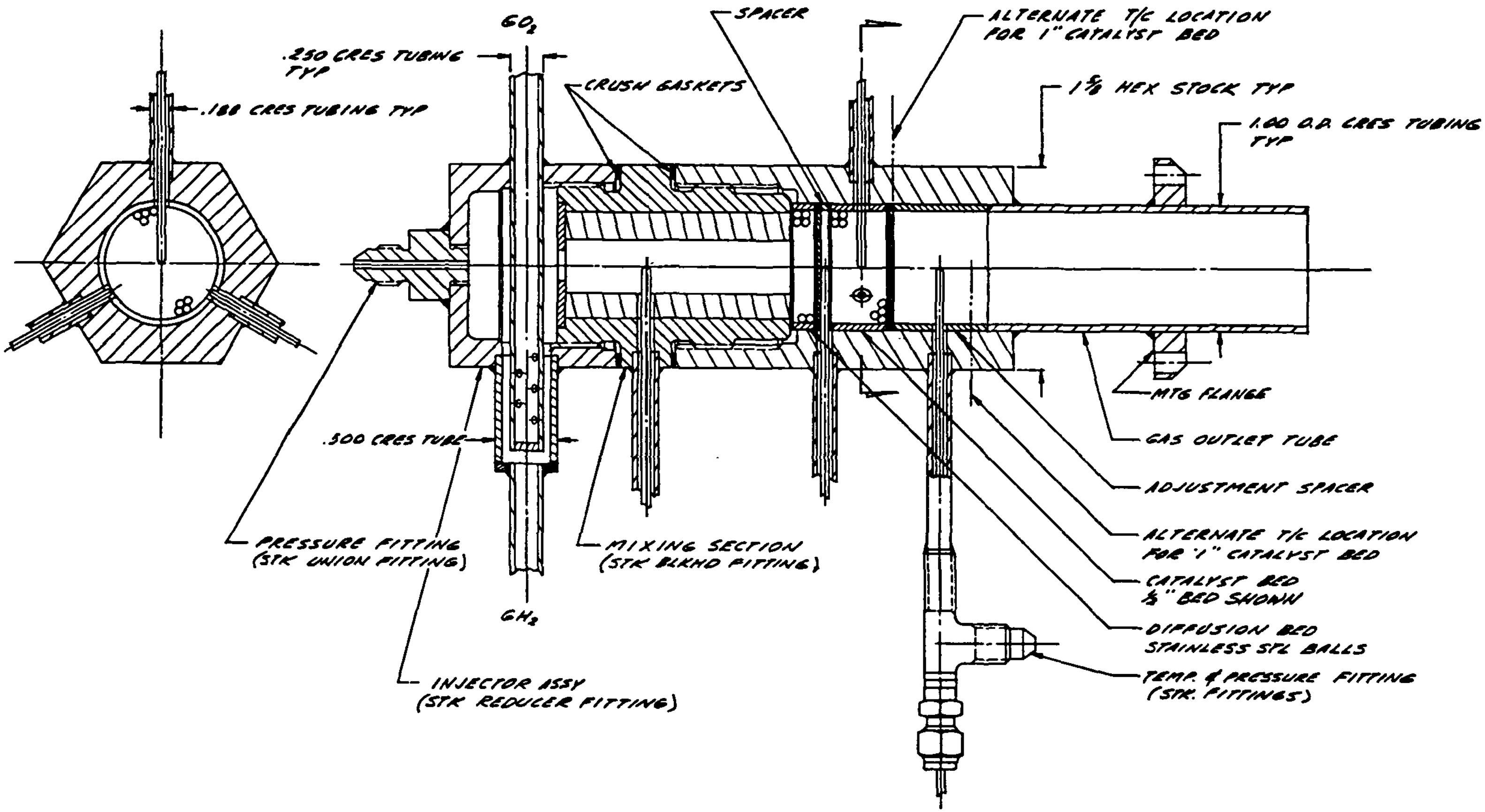
CATALYTIC IGNITER INVESTIGATIONS

- CONDUCT TESTS TO DETERMINE VARIABLES INFLUENCING/CONTROLLING FLASHBACK OF FLAME FRONT FROM CATALYST BED TO MIXING INJECTOR.
- DETERMINE ATTAINABLE CATALYST BED RESPONSE AND EVALUATE METHODS OF IMPROVING OVERALL IGNITER RESPONSE.
- DEVELOP GENERALIZED DESIGN GUIDELINES FOR CATALYTIC IGNITERS INCLUDING SCALING CRITERIA.

This figure is a section view of the low chamber pressure reactor used for the injector flashback and response enhancement tests. The high Pc reactor is identical in concept but reduced in size. The mixing section between the injector and catalyst bed contains interchangeable sleeve and orifice plate inserts, which provide the capability of varying the gas velocity through the mix section. Further variation of velocity was accomplished by using a motorized throat plug to increase pressures, which results in decreased flow velocity at constant propellant mass flow rates (controlled by sonic orifices).

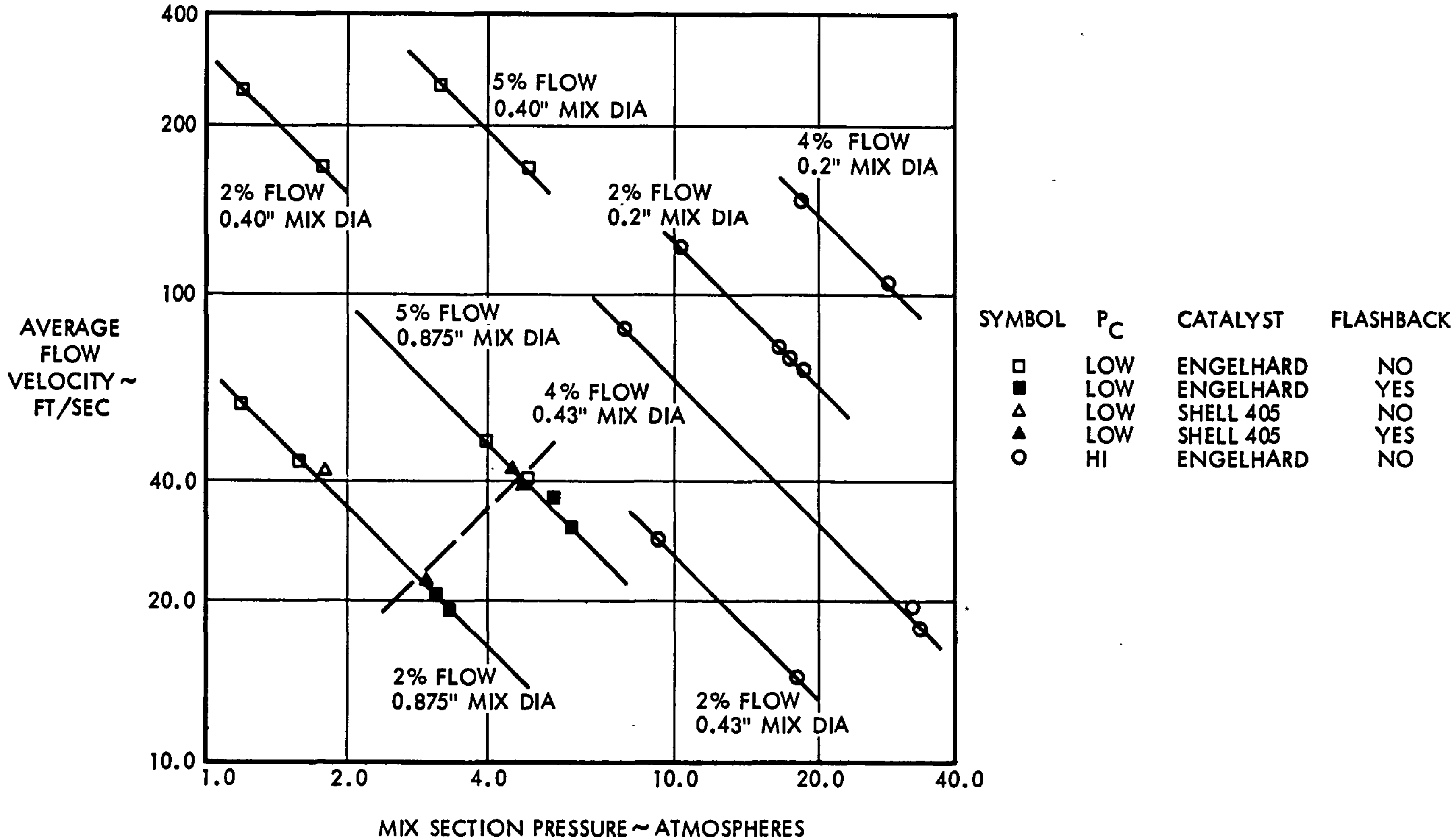


Catalytic Reactor Igniter - Low Chamber Pressure Thruster

A series of flashback investigation tests were performed with each pressure level hardware by varying the propellant flow rates (shown as percent of main thruster flows), mixing zone diameters, and operating pressures in attempting to initiate flashback from the catalyst bed to the injector. These tests were conducted at a nominal O/F mixture ratio of 1:1 and with a hydrogen lead on startup and lag on shutdown.

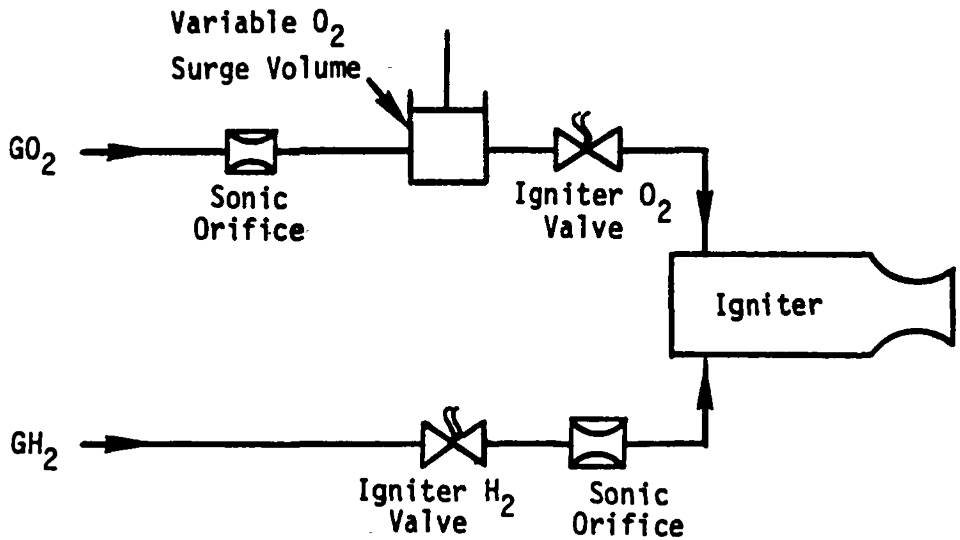
Flashback occurred with the 15 psia igniter only at pressures above 45 psia and with a mix diameter of .875 inches, which was equal to the catalyst bed diameter. The type of catalyst used did not effect the flashback limits. No flashback occurred with the 100 psia igniter during these tests, which included firings at over 500 psia with mix diameters also equal to the catalyst bed diameters.

H_2/O_2 IGNITER FLASHBACK DATA



The effects of ignition transients on the flashback limits were also investigated. This figure shows the technique used to attain a high O/F mixture ratio during the start transient. The variable volume plenum between the igniter oxygen fire valve and the oxidizer sonic flow control orifice provides a controlled surge of oxygen during the start transient, while the fuel flow rate remains constant. Valve timing was set to provide a fuel lead on startup, followed by an oxygen surge of less than 100 milliseconds, then an overall mixture ratio of 1:1.

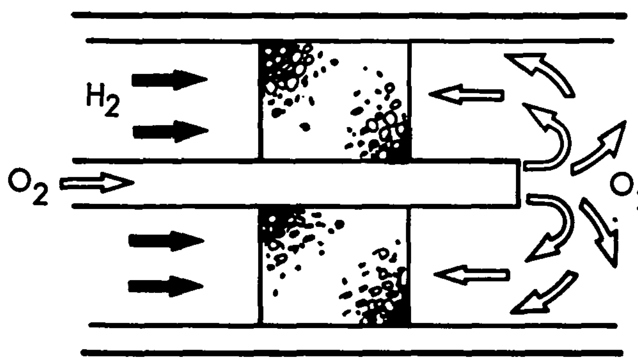
It was found that flashback could be instigated with either pressure level igniter by increasing the oxidizer surge volume. A high initial mixture ratio was also under evaluation as a means of improving response; however, these test results indicated that flashback could occur unless initial mixture ratios were carefully controlled.



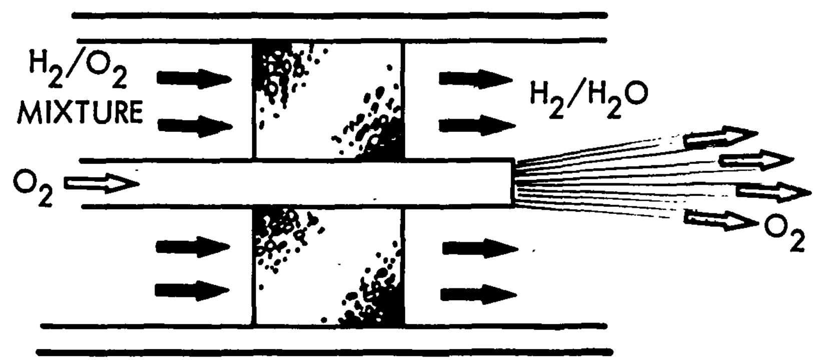
Schematic - High Initial Mixture Ratio Tests

Another technique of igniter response enhancement being evaluated is injection of oxygen downstream of the catalyst bed. Since most of the response time of the igniter consists of thermal delay of the catalyst bed, overall igniter response could be improved if ignition could be initiated downstream of the catalyst bed, as shown. This was attempted by injecting oxygen downstream of the catalyst bed while hydrogen was injected from upstream, first with a hydrogen lead, then with simultaneous injection. The upstream oxygen flow was then initiated to maintain a mixture ratio of 1:1 within the catalyst bed. Overall mixture ratios as high as 50:1 have been investigated.

IGNITER RESPONSE ENHANCEMENT
DOWNSTREAM O_2 INJECTION



(a) IGNITION AT DOWNSTREAM SURFACE OF CATALYST BED, M.R > 10



(b) STEADY-STATE OPERATION:

H_2/O_2 MIXTURE THROUGH CATALYST BED, M.R. ~ 1.0

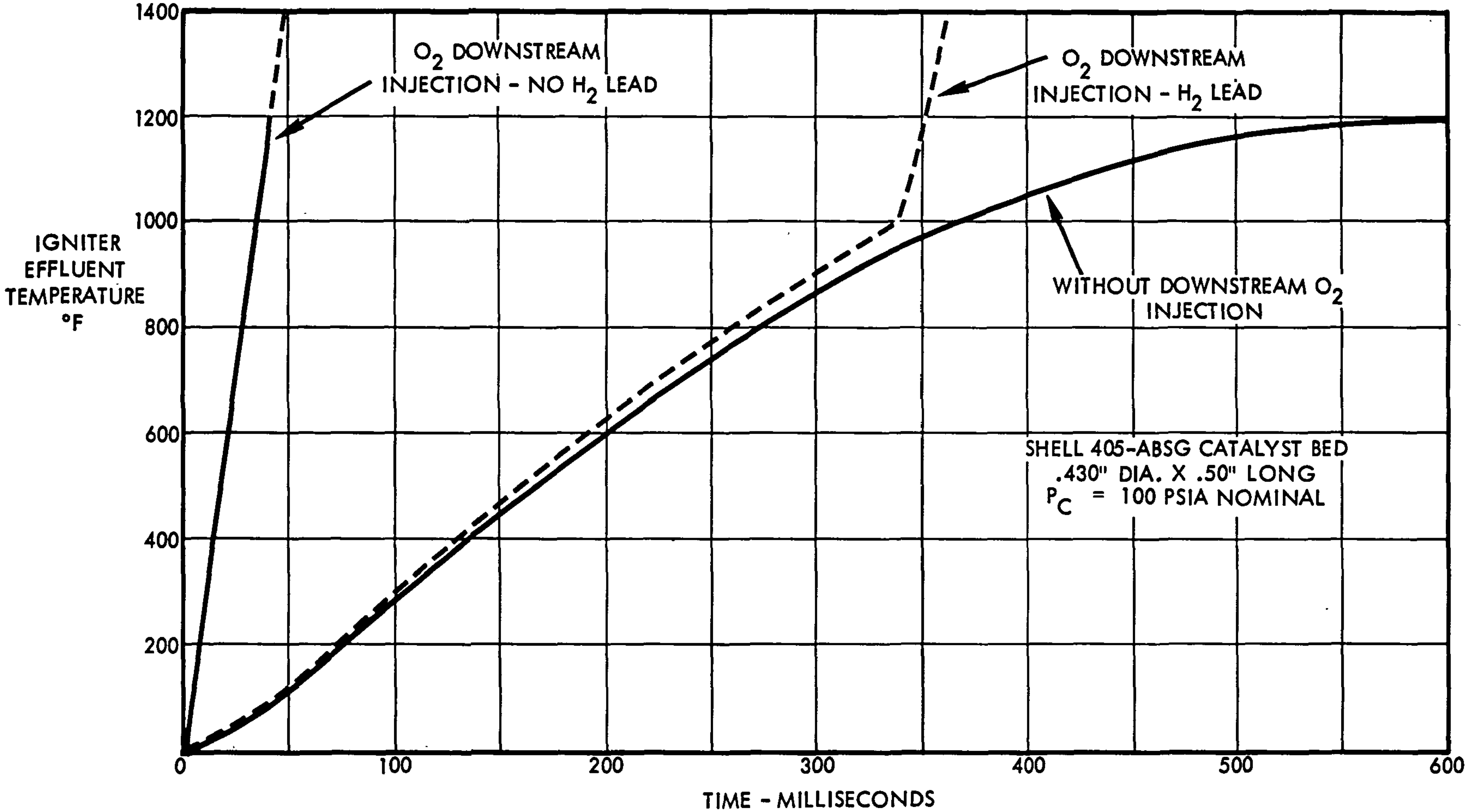
PURE O_2 INJECTED DOWNSTREAM

The most realistic definition of overall igniter response is the time required for the effluent gases to reach a temperature sufficient for ignition of the main thruster. This illustration compares the effluent temperature response with and without downstream oxygen injection for the 100 psia igniter.

Without oxygen downstream injection, reactor effluent temperatures reach 1200°F in 600 milliseconds after onset of propellant flows. Initial tests with downstream oxygen injection were conducted with a hydrogen lead through the catalyst bed, and resulted in no ignition of the oxygen until autoignition occurred at nearly 1000°F. Injection of downstream oxygen simultaneously with hydrogen flow achieved effluent temperatures of 1200°F in approximately 40 milliseconds, as shown.

The effects of ignition with oxygen-rich propellant mixtures on catalyst life will have to be evaluated, although the oxygen-rich ignition transient can be minimized in length to avoid sintering the downstream catalyst bed pellets.

IGNITER RESPONSE ENHANCEMENT DATA



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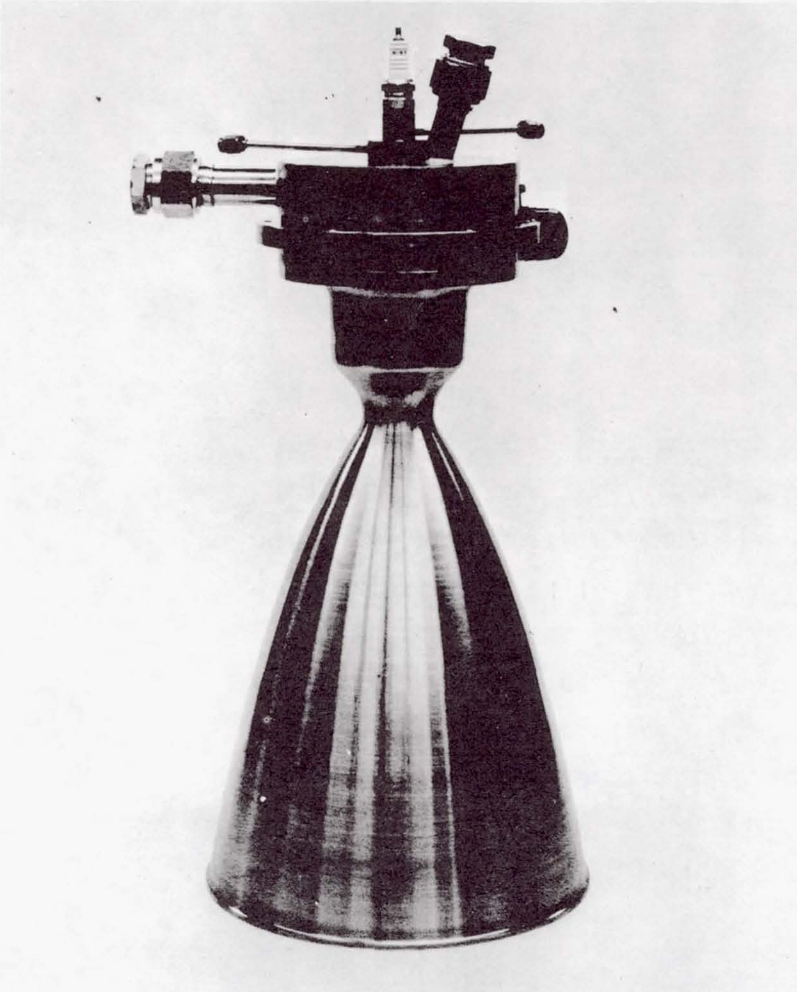
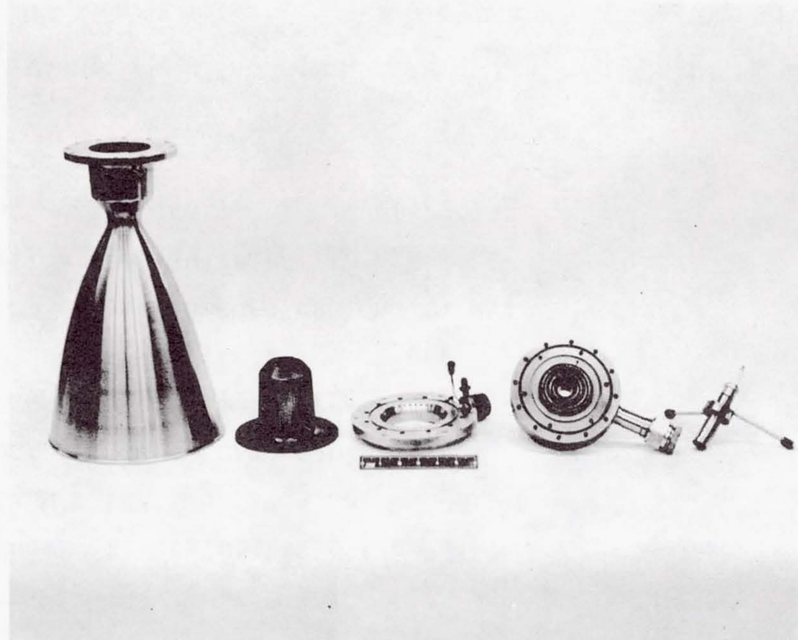
FLIGHTWEIGHT THRUSTER DEVELOPMENT

- DESIGN 30 LBF, 15 PSIA COOLED FLIGHTWEIGHT GO_2/GH_2 THRUSTER WITH CATALYTIC PILOT BED IGNITER.
- FABRICATE TWO FLIGHTWEIGHT THRUSTER ASSEMBLIES.
- CONDUCT INJECTOR SCREENING TESTS, ALSO EMPLOYING THE IMPINGING SHEET INJECTOR FROM NAS 3-11227.
- EVALUATE THRUSTER PERFORMANCE AND DURABILITY FOR PULSE MODE AND STEADY-STATE FIRING, INCLUDING A 1800 SECOND SINGLE START TEST.

Illustrated in this figure is the basic TRW Systems duct-cooled H_2/O_2 thruster design approach. The nominal design/operating conditions for the specific hardware shown are 900 lbf thrust at 300 psia chamber pressure. With this design approach, a portion of the H_2 is employed as the duct coolant fluid - with subsequent injection into the nozzle convergent region. The nozzle throat/divergent regions are film cooled by the duct effluent gases. This design approach permits a thin-walled (lightweight) chamber design which is truly adiabatic with all components operating within their material elastic regimes. The disassembled view illustrates the chamber shell, duct liner, coolant manifold, main propellant injector, and pilot spark igniter.



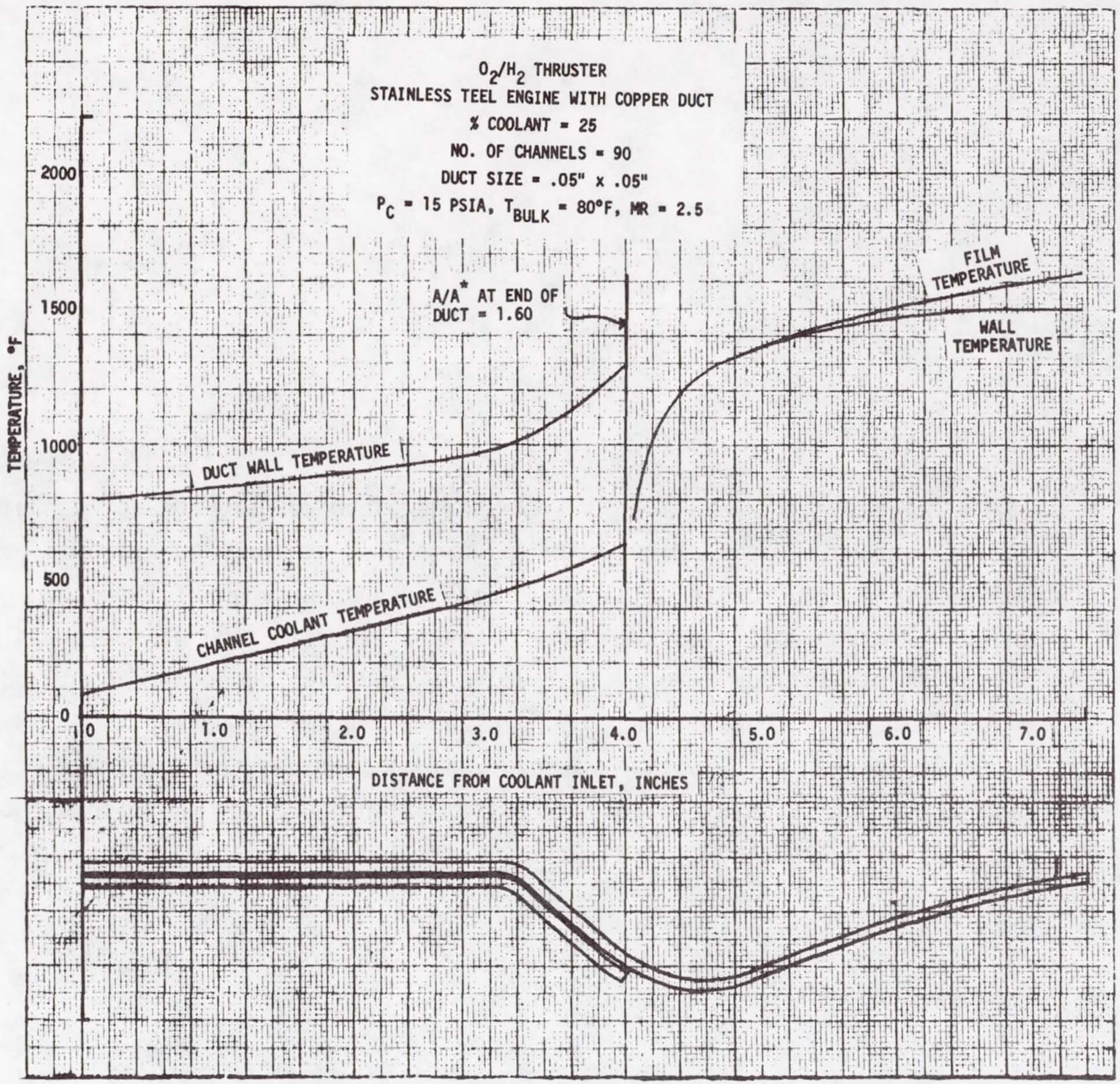
ASSEMBLY



DISASSEMBLY

TRW H_2/O_2 EXPERIMENTAL DUCT COOLED CHAMBER

Analytical investigations were conducted to evaluate the thermal operating characteristics of the subject 30 lbf/15 psia gaseous H_2/O_2 thruster - utilizing TRW Systems duct cooled chamber computer programs. Thrust chamber wall operating temperatures were computed for a range of coolant flow rates, duct geometries, and chamber wall configurations. This graph illustrates typical results for one set of design conditions: thin wall chamber shell, a duct coolant flowrate equivalent to 25% of the total thruster H_2 flowrate, and a duct exit located at a contraction ratio of 1.6:1.

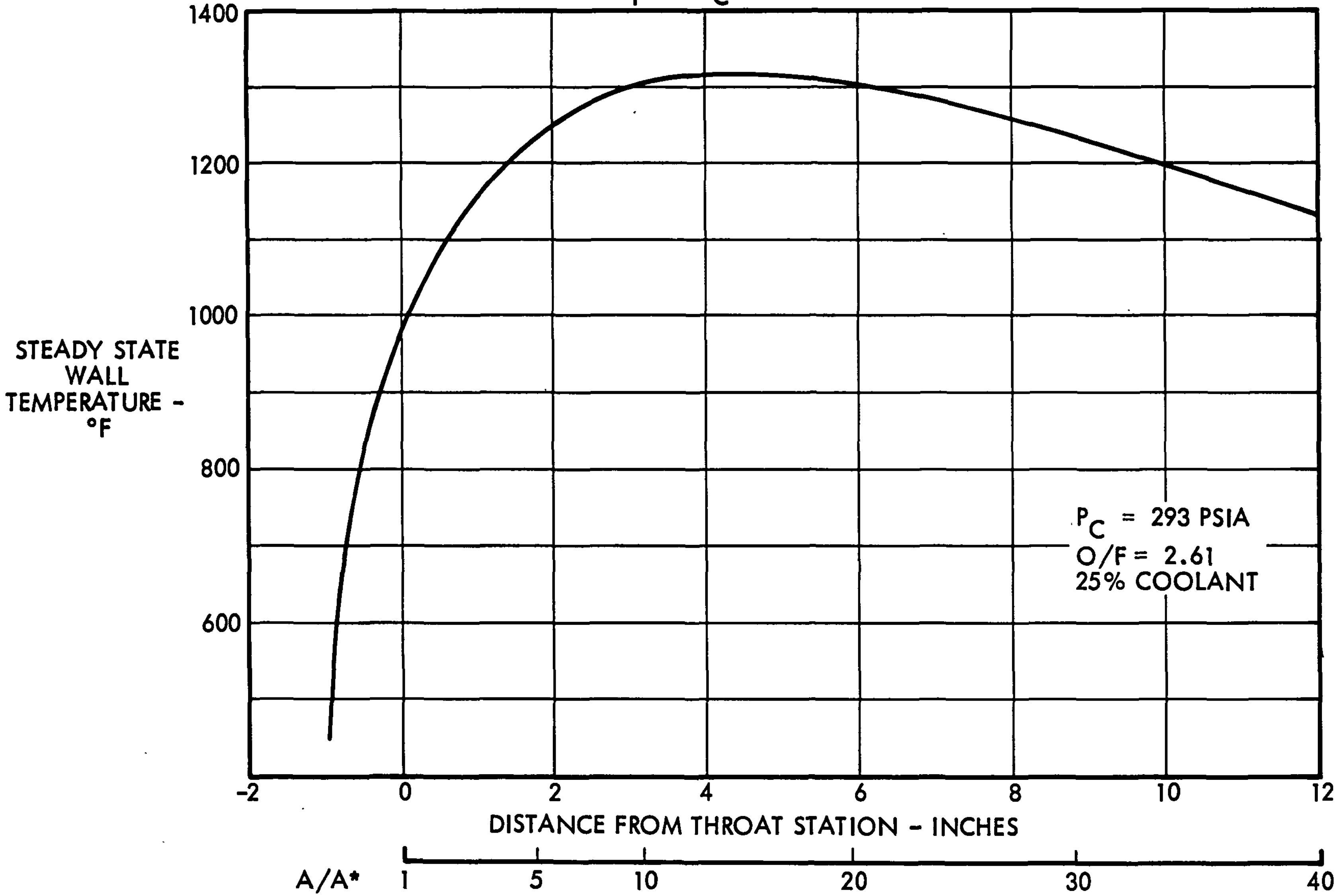


O_2/H_2 THRUSTER
STAINLESS TEEL ENGINE WITH COPPER DUCT
% COOLANT = 25
NO. OF CHANNELS = 90
DUCT SIZE = .05" x .05"
P_C = 15 PSIA, T_{BULK} = 80°F, MR = 2.5
A/A* AT END OF DUCT = 1.60
FILM TEMPERATURE
WALL TEMPERATURE
DUCT WALL TEMPERATURE
CHANNEL COOLANT TEMPERATURE
TEMPERATURE, °F
0
500
1000
1500
2000
0
1.0
2.0
3.0
4.0
5.0
6.0
7.0
DISTANCE FROM COOLANT INLET, INCHES

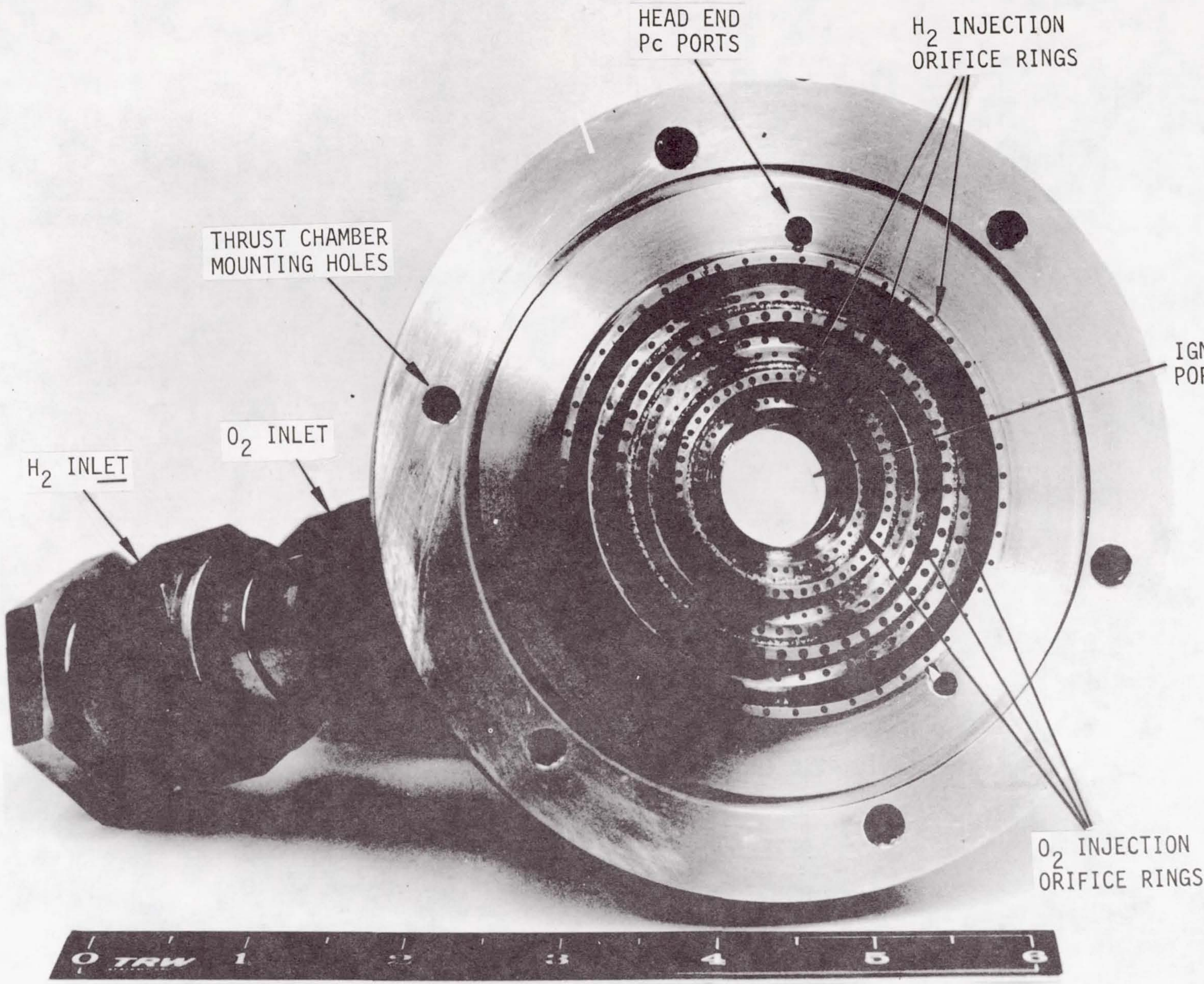
Measured thrust chamber (steady-state) wall temperature data from the high thrust (900 lbf @ 300 psia) tests are plotted in this figure for the TRW Systems thin-walled duct cooled thruster design. The maximum chamber temperature of about 1320°F was measured in the divergent nozzle region at a point corresponding to a nozzle expansion ratio of approximately 15:1. This particular test data was selected at a mixture ratio comparable to the 2.5 O/F specified for the subject low pressure thruster. Steady-state chamber wall temperature data has been obtained encompassing the mixture ratios of interest for the high pressure thruster.

EXPERIMENTAL DUCT-COOLED GASEOUS O_2/H_2 THRUSTER DATA

($F = 900\ LB_F$ AT $P_C = 300$ PSIA)

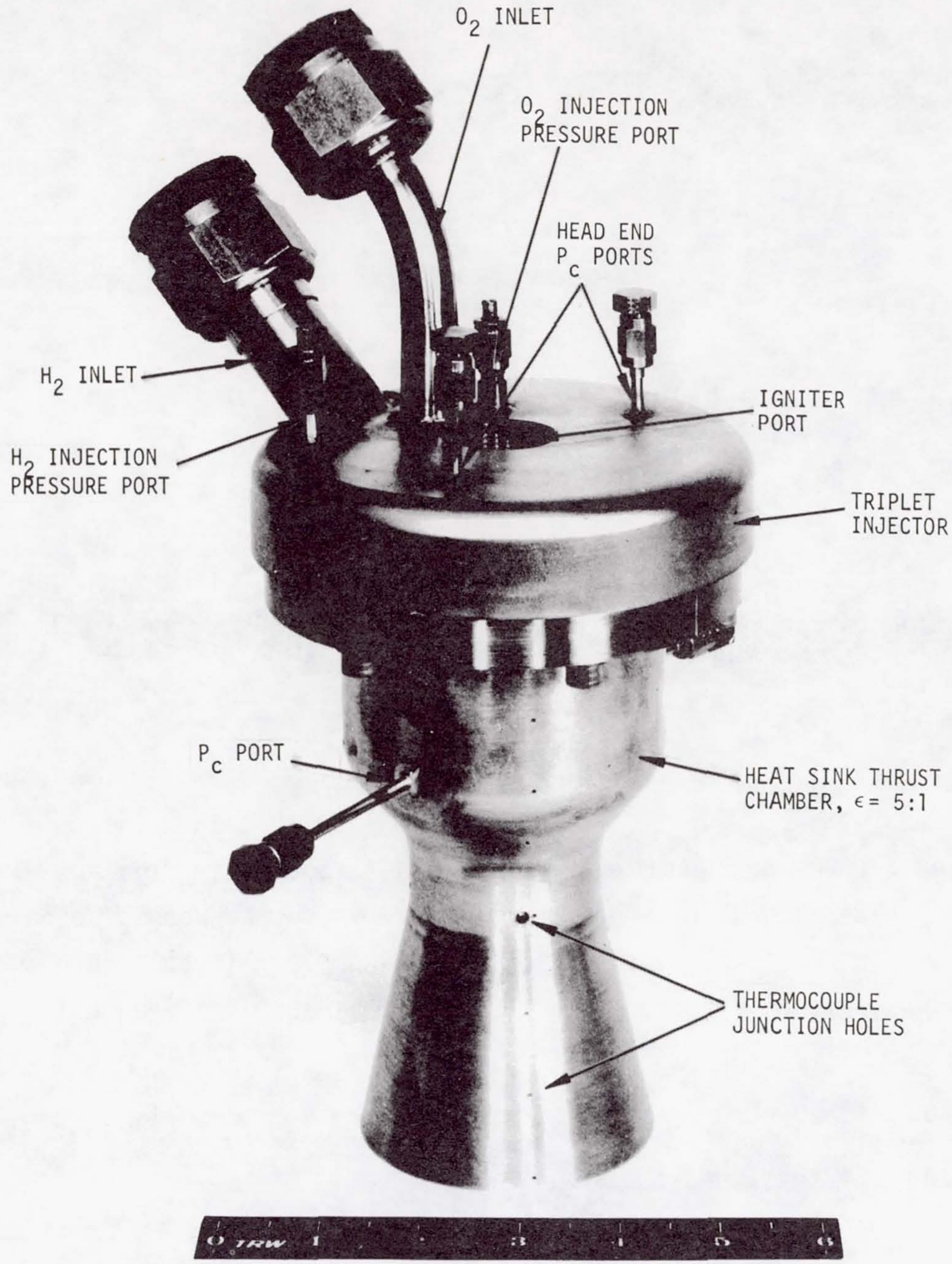


The thruster injector fabricated for this program is a raised post triplet design with a centrally located igniter port. This is the same basic injector design employed successfully for the 900 pound thrust, 300 psia chamber pressure thruster.



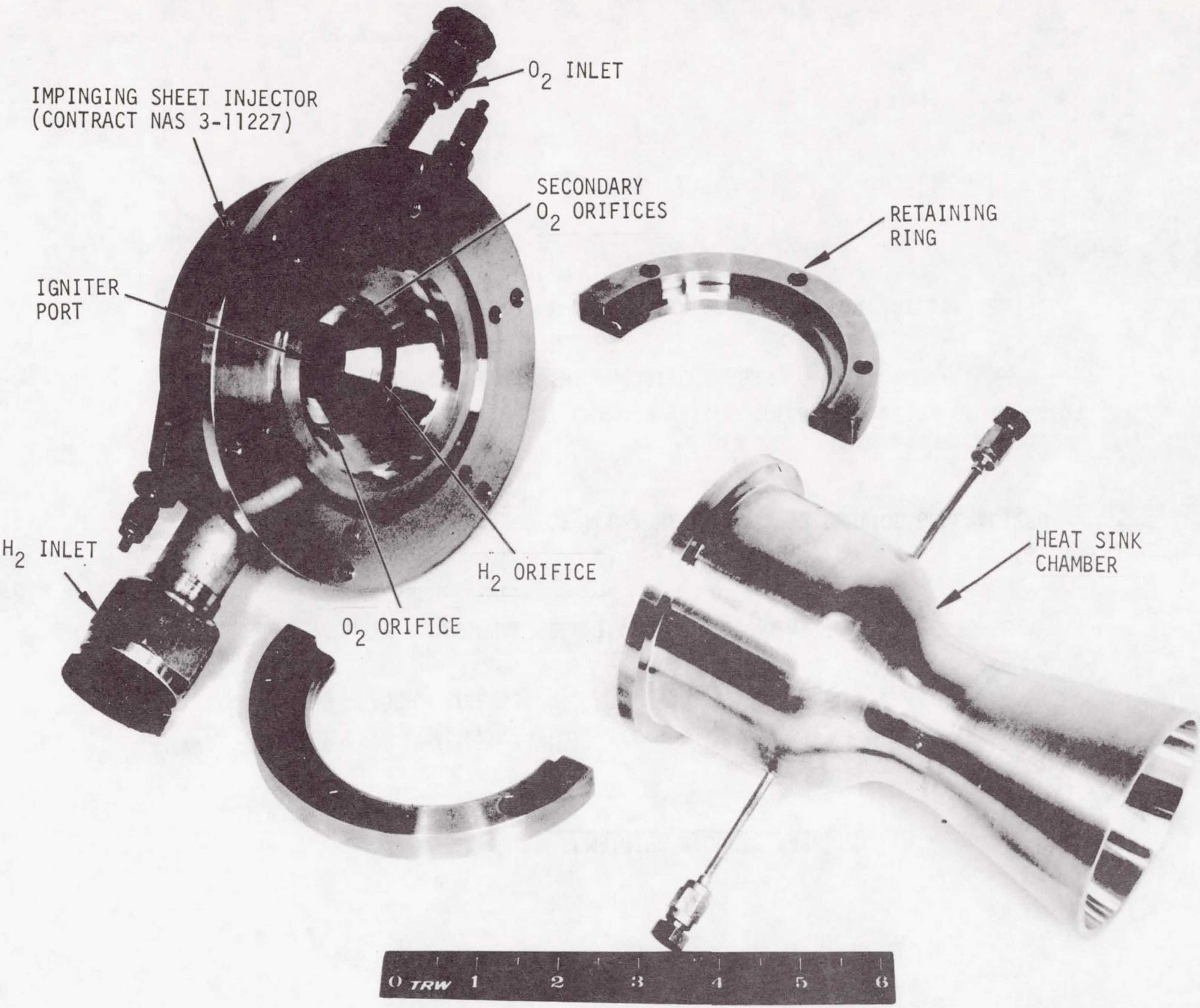
Flightweight Triplet Injector

This photograph shows the triplet injector assembled to the heat sink thrust chamber which will be used for injector screening tests.



Assembly of Flightweight Triplet Injector and Heat Sink Thrust Chamber

The heat sink thrust chamber is also compatible with the residual impinging sheet low chamber pressure injector from Contract NAS 3-11227. Injector screening tests will be conducted with this injector, as well as with the triplet injector, to obtain performance and chamber wall temperature measurements.



Impinging Sheet Injector (NAS 3-11227) and
Heat Sink Thrust Chamber, Disassembled

REMAINING PROGRAM TASKS

- COMPLETE LOW TEMPERATURE CYCLIC LIFE TESTS AT -250°F WITH THE SHELL 405 - ABSG CATALYST.
- EVALUATE ADDITIONAL TECHNIQUES OF ENHANCING IGNITER RESPONSE AT EACH PRESSURE LEVEL.
- COMPLETE CORRELATION OF ALL IGNITER DATA TO PROVIDE GENERALIZED DESIGN GUIDELINES.
- FINALIZE COOLED THRUST CHAMBER DESIGN AFTER COMPLETION OF INJECTOR TESTS, AND FABRICATE TWO THRUST CHAMBERS.
- CONDUCT FLIGHTWEIGHT THRUSTER PERFORMANCE AND DURABILITY TESTS.

"AUXILIARY PROPULSION SUBSYSTEM INVESTIGATIONS"

J. P. McCARTY

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AUXILIARY PROPULSION SUBSYSTEM INVESTIGATIONS

MARSHALL SPACE FLIGHT CENTER

J. P. McCarty

In House

AUXILIARY PROPULSION SUBSYSTEM INVESTIGATIONS

This paper presents some results of technology investigations to establish what efforts need to be undertaken to provide the data and experience necessary to set requirements and establish designs for the Space Shuttle Vehicle Auxiliary Propulsion Subsystem (APS). The APS performs the vehicle propulsion function from the completion of main engine boost through completion of re-entry when vehicle control is accomplished by aerodynamic forces. This includes attitude control of the booster and attitude control and orbital maneuvering of the orbiter. The results and efforts described emphasize the subsystem and storage, conditioning and distribution assembly requirements and depends, for completeness, on component technology, particularly in the storage and thrustor areas, being developed in parallel efforts.

MOTIVATION AND OBJECTIVE

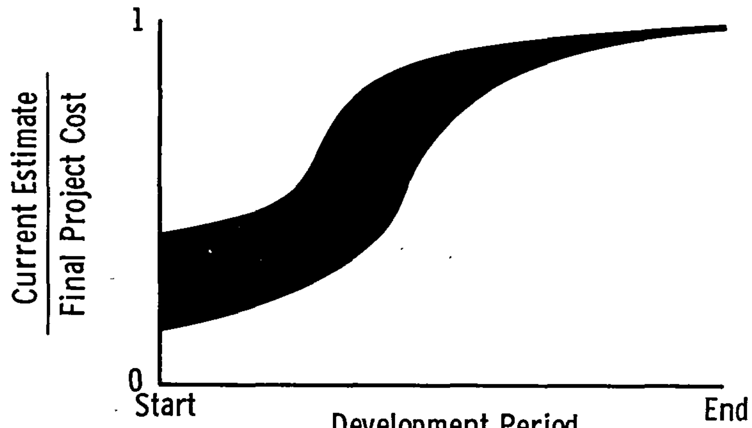
History shows first estimates of advanced systems development cost are low. The costs rise initially as detail design of subsystems and components to meet the established requirement is accomplished then more abruptly with the advent of experimental component and subsystem data. One cause of this rise, during the early years of the development period, is the higher frequency of "unpleasant discoveries" which require unplanned changes and expenditures to correct. As the project progresses the probability of "unpleasant discoveries" decreases thus increasing the likelihood of estimated costs truly reflecting the final project costs.

Recognition of these characteristics provides a strong motivation for technology and advanced development effort. This effort would establish directly applicable technical data for use in setting development requirements that truly reflect those needed to achieve the development objective and are realistically based on known hardware capabilities. Also this effort will provide knowledge of potential development problems and appropriate solutions reducing the probability of "surprises" or "unpleasant discoveries" in the project.

The development of the Shuttle Vehicle APS will most likely experience the same characteristics as past development programs unless efforts are expended early to establish technical knowledge upon which to base realistic requirements. Such an effort is described in the following charts and has the objective of establishing a technical base to support subsystem design and initiate component development.

MOTIVATION AND OBJECTIVE

LAB: S&E-ASTN-PP
NAME: J. P. McCarty
DATE: April 7, 1971

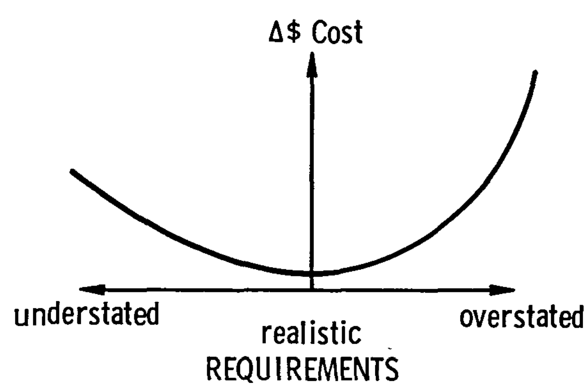


- High probability of "unpleasant discoveries" during early years
- Impact of "unpleasant discovery"
  - -Date
  - -Nature

Necessary Objective

Establish technical base to support subsystem design and initiate component development

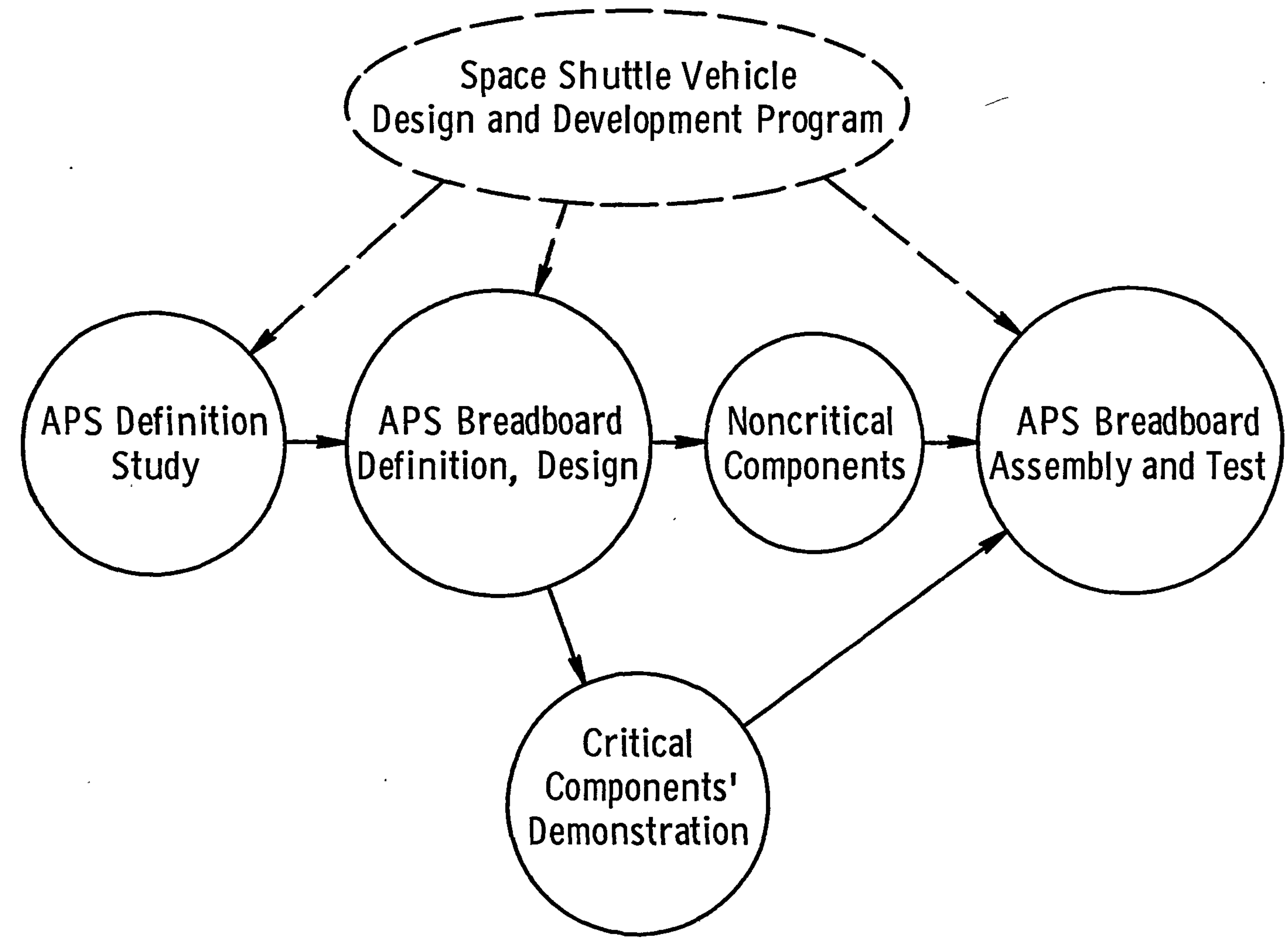
- Demonstrate subsystem performance and operation
- Validate component requirements
- Basis for mechanical and environmental design requirements
- Initial assessment of life, cost and maintenance



APS ADVANCED DEVELOPMENT LOGIC

The instrument for achieving the stated objectives is the assembly and test of a breadboard APS system. The breadboard system would simulate the anticipated flight system in operation and control although flight weight components would not be employed unless critical to proper system operation or data validity. The necessary technology efforts are guided by the overall Shuttle Vehicle design & requirements. Initially vehicle descriptions, groundrules and mission requirements are provided for APS Definition Studies. These studies perform the analysis and design required to screen the numerous potential systems and arrive at candidate flight system designs and identify the critical requirements and assumptions of the designs. The APS breadboard is defined from these preliminary flight system designs, requirements and assumptions. Design effort based upon Shuttle Vehicle configurational data, establishes and maintains compatible component performance requirements and interconnect hardware fabrication information. Components identified as critical, i.e. unique to the system and exceeding existing component capabilities but crucial to satisfactory system operation, are pursued through demonstration of the achievable operating capabilities. Non-critical components are purchased from existing stocks or built with existing capabilities. Both critical and non-critical components are then assembled into the breadboard system and tested to shuttle duty cycles and environment. The breadboard system configuration is such that basic component arrangements can be changed and examined for system interactions.

MARSHALL SPACE FLIGHT CENTER SCIENCE & ENGINEERING	APS ADVANCED DEVELOPMENT LOGIC	LAB: S&E-ASTN-PP NAME: J. P. McCarty DATE: April 7, 1971



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ORBITER
ORBITAL MANEUVERING ACCOMPLISHED BY MULTIPLE GAS/GAS ACP THRUSTERS

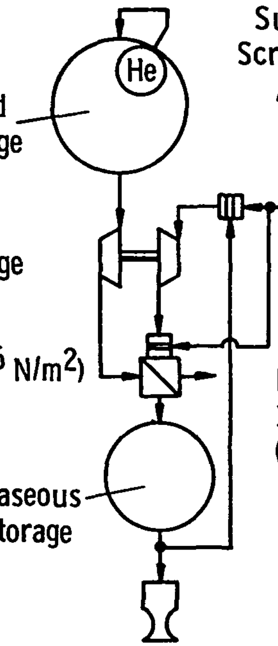
Two basic APS approaches are available. The first consists of employing a single system to perform all the APS propulsion requirements. For this case a high pressure, turbopump based system was selected and a basic system configuration established to meet the current shuttle groundrules and constraints. Several major decisions remain including hardware commonality between orbiter and booster and the degree of integration with other cryogen using subsystems.

MARSHALL SPACE FLIGHT CENTER SCIENCE & ENGINEERING	ORBITER Orbital Maneuvering Accomplished By Multiple Gas/Gas ACP Thrusters	LAB: S&E-ASTN-PP NAME: J. P. McCarty DATE: April 7, 1971

Note:

Total Impulse = 12-13 x 10^6 lb-sec (53-58 x 10^6 N-sec)
Mission Duration = 7 days

Turbopump Based System



Major Decisions Remaining

- Extent of commonality
- Integration with other subsystems
- Control system
- Malfunction detection and isolation

Mass
35900 lb
(16280 kg)

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CHART No. ______

ORBITER
ORBITAL MANEUVERING AND ATTITUDE CONTROL ACCOMPLISHED BY SEPARATE THRUST DEVICES

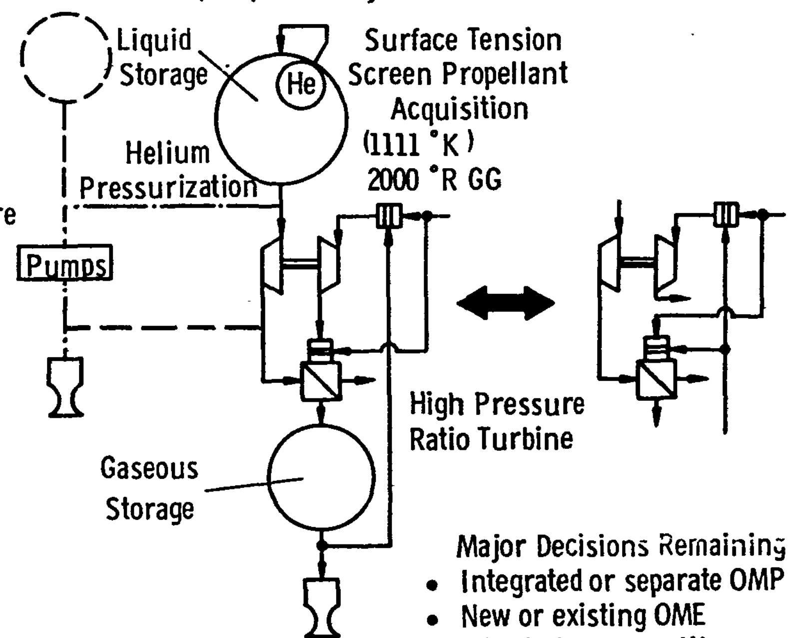
The second approach consists of employing one system, the attitude control propulsion subsystem (ACPS), to perform all attitude control and small orbital maneuver propulsion requirements ($\sim$3M lb-sec) and a second, the orbital maneuvering subsystem (OMS), for the major orbital maneuver requirements ($\sim$ 10 M lb-sec.). The configuration currently selected for the ACPS is again a high pressure turbopump based system while the OMS could be a liquid/liquid thruster fed from either the ACPS turbomachinery or from its own separate turbomachinery or an RL-10 derivative. This overall system approach has received less detailed investigation, thus there are more decisions remaining to be made.

MARSHALL SPACE FLIGHT CENTER SCIENCE & ENGINEERING	ORBITER Orbital Maneuvering And Attitude Control Accomplished By Separate Thrust Devices	LAB: S&E-ASTN-PP NAME: J. P. McCarty DATE: April 7, 1971

Note:
Total Impulse = 12-13 x 10^6 lb-sec (53-58 x 10^6 N-sec)
Mission Duration = 7 days

Turbopump Based System

Pump Discharge Pressure
1300-2600 psia
(8.9-17.9 x 10^6 N/m^2)



Mass
35000 to 36300 lb
(15870 to 16460 kg)

Major Decisions Remaining
- Integrated or separate OMP
- New or existing OME
- Extent of commonality
- Integration with other systems
- Series (reburn) or parallel Hx
- Thruster cooling (inlet pressure)
- Integral or separate ACP/OMP tankage
- Control System
- Malfunction detection and isolation

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ORBITER
AREA OF CONCERN

The identified APS configurations are systems balanced on weight, operational flexibility, system simplicity, and technology required. Each configuration has several critical requirements and assumptions which define areas of concern meriting further investigation to demonstrate the required capability. The areas of concern are identified together with alternatives or "fall back" positions in case the primary approach proves unworkable. Overall system penalties are shown for each alternative.

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SCIENCE & ENGINEERING

ORBITER

LAB: S&E-ASTN-PP

NAME: J. P. McCarty

DATE: April 7, 1971

Areas of Concern	Alternatives	Mass Penalty			
		(lb)*	(kg)*	(lb)**	(kg)**
Thruster Minimum Inlet Temperature	Increase Minimum Temperature	2200	998	1400 to 4300	635 to 1950
Reburn Heat Exchanger	Conventional Heat Exchanger	1800	816	500 to 3400	227 to 1542
Reusable HPI	Vacuum Jackets	800	363	800	363
Turbopump Life	Reduce Operating Requirements	700	317	700	317
Pressure Vessel Cycle Life	Increase Design Margin	500	227	500	227
Propellant Acquisition	Small Separate Tanks	400	181	400	181
Turbopump Response	Reduce Response Requirement	300	136	300	136
Thruster Life	Increase Coolant	300	136	90	41

Areas of Concern	Alternatives	Mass Sensitivity, Mass/Specific Impulse			
		(lbm^2/lbf-sec)*	(kg^2/N-sec)*	(lbm^2/lbf-sec)**	(kg^2/N-sec)**
OMP Thruster Performance	Reduce Requirement	100	4.6	70	3.2
ACP Thruster Performance	Reduce Requirement			30	1.4

*Orbital Maneuvering Accomplished by Multiple Gas/Gas ACP Thrusters

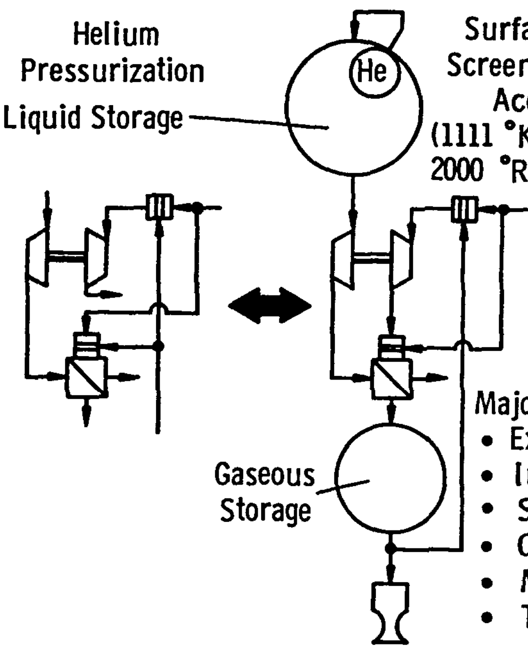
**Orbital Maneuvering and Attitude Control Accomplished by Separate Thrust Devices

BOOSTER

The configuration currently selected for the booster APS is also a high pressure turbopump based system. Several major decisions remain.

MARSHALL SPACE FLIGHT CENTER SCIENCE & ENGINEERING	BOOSTER	LAB: S&E-ASTN-PP NAME: J. P. McCarty DATE: April 7, 1971

Note:
Total Impulse = 0.8-1.0 x 10^6 lb-sec (3.5-4.5 x 10^6 N-sec)
Mission Duration = 6 min to 4 hr



Major Decisions Remaining
- Extent of Commonality
- Integration with other systems
- Series (Reburn) or parallel Hx
- Control system
- Malfunction detection and isolation
- Thruster cooling (inlet pressure)

BOOSTER
AREA OF CONCERN

The identified APS configurations again are systems balanced on weight, operational flexibility, system simplicity, and technology required, with several areas of concern meriting further investigation. The areas of concern, alternatives and system penalties are shown.

MARSHALL SPACE FLIGHT CENTER SCIENCE & ENGINEERING	BOOSTER	LAB: S&E-ASTN-PP NAME: J. P. McCarty DATE: April 7, 1971

Areas of Concern	Alternatives	Mass Penalty	
		(lb)	(kg)
Thruster Minimum Inlet Temperature	Increase Minimum Temperature	1,000 to 2,600	453 to 1,179
Reburn Heat Exchanger	Conventional Heat Exchanger	300 to 2,000	136 to 907
Turbopump Response	Reduce Response Requirement	300	136
Thruster Life	Increase Coolant	30	14

Areas of Concern	Alternatives	Mass Sensitivity	
		(lbm^2/lbf-sec)	(kg^2/N-sec)
ACP Thruster Performance	Reduce Requirement	11	0.5

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BREADBOARD DEFINITION, DESIGN & FABRICATION

The breadboard system design will be based on preliminary designs of flight systems defined in the APS Definition Studies. This effort would establish the detailed breadboard design and provide necessary configuration updating to assure proper modeling of important subsystem characteristics. Critical components would be defined. Experience from the critical component demonstration efforts would be incorporated into the breadboard design as it became available.

MARSHALL SPACE FLIGHT CENTER SCIENCE & ENGINEERING	BREADBOARD DEFINITION, DESIGN, AND FABRICATION	LAB: S&E-ASTN-PP NAME: J. P. McCarty DATE: April 7, 1971

- Establish breadboard design

  -Flow schematics, fluid and thermodynamic models, control logic

  -Component functional and performance requirements

  -Interconnect and support component design and fabrication

- Update configuration and design to ensure proper modeling of important subsystem characteristics

  -Ability to alter critical parameters

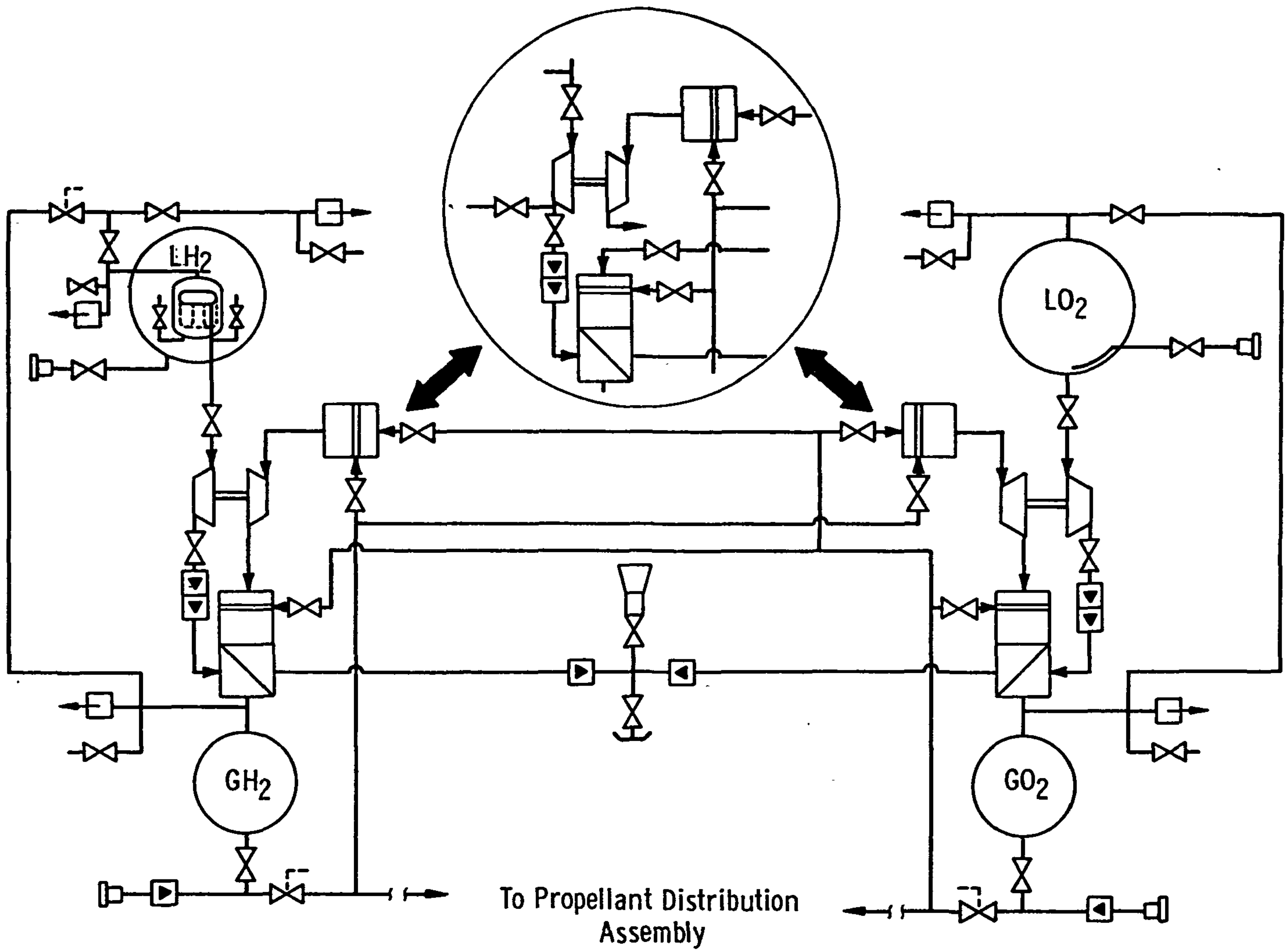
- Incorporate component experience from critical component demonstrations

PRELIMINARY

SPACE SHUTTLE APS BREADBOARD SCHEMATIC

The APS breadboard as presently envisioned is depicted in this schematic. The turbopump feed system employs either reburn heat exchangers or parallel gas generators (see inset) to drive the turbines and heat exchangers independently.

MARSHALL SPACE FLIGHT CENTER SCIENCE & ENGINEERING	SPACE SHUTTLE APS BREADBOARD SCHEMATIC	LAB: S&E-ASTN-PP NAME: J. P. McCarty DATE: April 7, 1971



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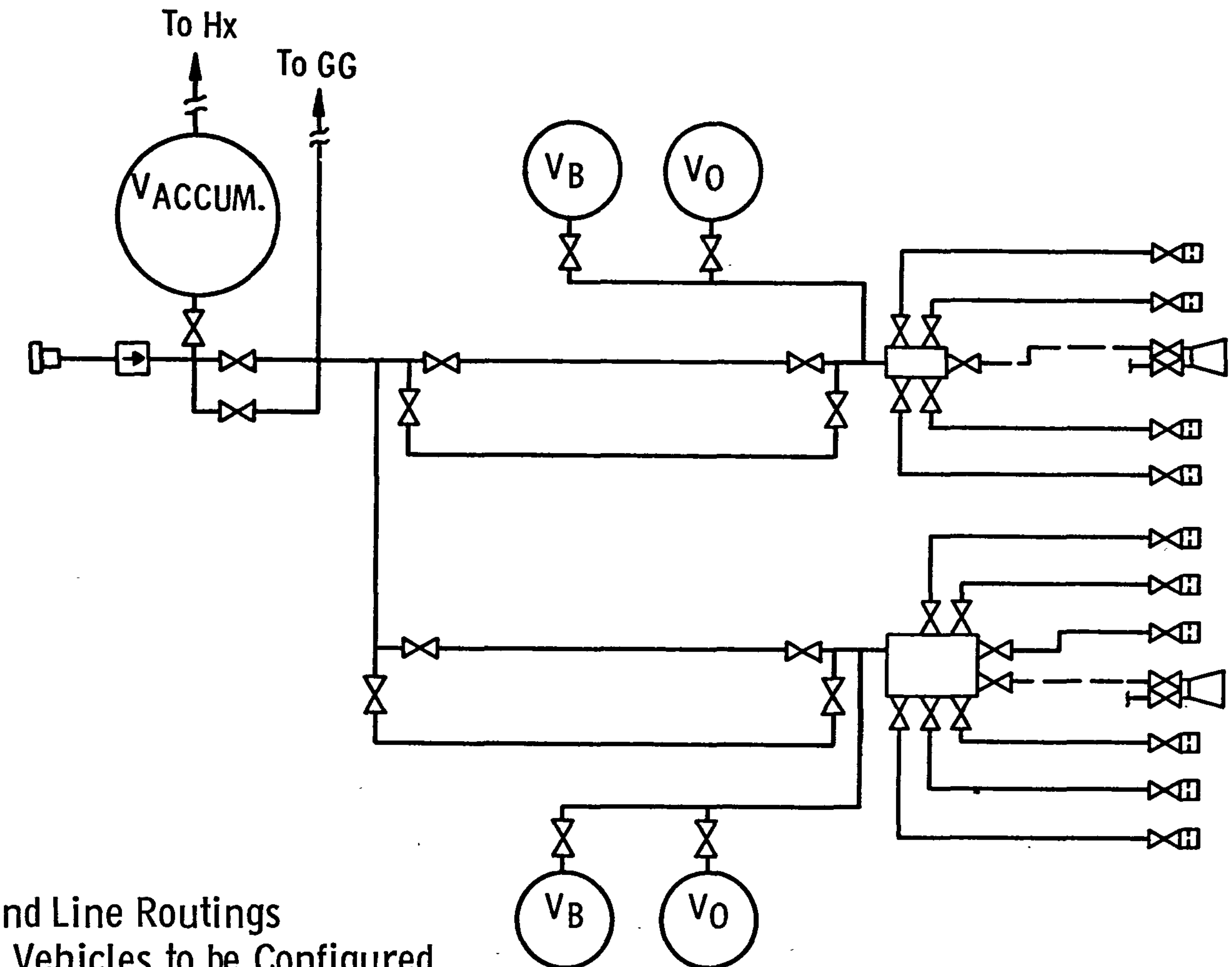
PRELIMINARY
SPACE SHUTTLE APS BREADBOARD
PROPELLANT DISTRIBUTION SCHEMATIC

The APS Breadboard propellant distribution system downstream of the gaseous propellant accumulators is shown. It includes two main branches simulating line routing to vehicle fore and aft thrusters. Each branch would contain at least one actual thruster assembly. Additional thrusters would be simulated by valves and flow control orifices. The capacity of other branches could be simulated by adding additional volumes (V_B, V_O) to the system.

MARSHALL SPACE FLIGHT CENTER SCIENCE & ENGINEERING	SPACE SHUTTLE APS BREADBOARD PROPELLANT DISTRIBUTION SCHEMATIC TYPICAL FOR BOTH GH_2 AND GO_2 SIDES	LAB: S&E-ASTN-PP NAME: J. P. McCarty DATE: April 7, 1971



Note:
Volumes and Line Routings
Depend on Vehicles to be Configured

CRITICAL COMPONENT DEMONSTRATIONS

The identified critical components would be pursued through design, fabrication, and test efforts to develop needed component technologies and to demonstrate component capabilities. These component assemblies would then be delivered to the breadboard system effort for assembly into the system.

MARSHALL SPACE FLIGHT CENTER SCIENCE & ENGINEERING	CRITICAL COMPONENT DEMONSTRATIONS	LAB: S&E-ASTN-PP NAME: J. P. McCarty DATE: April 7, 1971

- Analyze, design, fabricate, and test critical components that require new technology

  -Design for desired requirements

  -Determine achievable performance

  -Evaluate operation and life

- After test, deliver for breadboard

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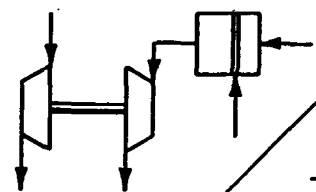
CRITICAL COMPONENTS

The critical components that have been identified are shown along with corresponding operating parameter values.

MARSHALL SPACE FLIGHT CENTER SCIENCE & ENGINEERING	CRITICAL COMPONENTS	LAB: S&E-ASTN-PP NAME: J. P. McCarty DATE: April 7, 1971

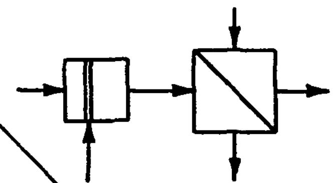
TURBOPUMP AND GG

Parameter	H_2	O_2
P_D, psia	1300-2600	1300-2600
W_P, lb/sec	3.8-7.4	12-25
P_C, psia	500	500
T_{TI}, °R	1860	1860

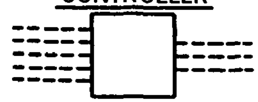


THERMO CONDITIONER

Parameter	H_2	O_2
P_G, psia	50-500	50-500
T_G, °R	1860-3500	1860
T_{HO}, °R	100-300	200-600

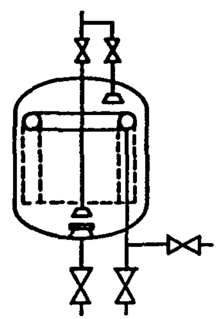


PCA CONTROLLER



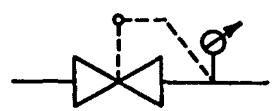
H_2 ACQUISITION

W_{MAX}, lb/sec	7.5
a, g's	-1
Q_{screen}, $\frac{Btu}{sec}$	→ 0



PRIMARY REGULATORS

Parameter	H_2	O_2
P_{IN}, psia	800-2800	800-2800
P_{OUT}, psia	595	595
W, lb/sec	4.0	15.0



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CRITICAL COMPONENTS

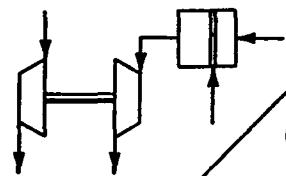
LAB: S&E-ASTN-PP

NAME: J. P. McCarty

DATE: April 7, 1971

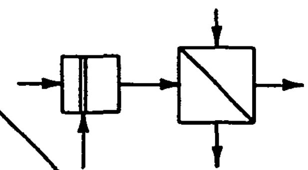
TURBOPUMP AND GG

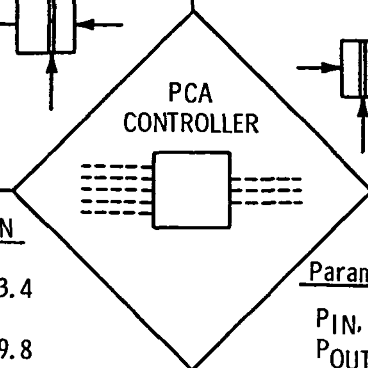
Parameter	H_2	O_2
P_D, N/m^2	$8.9\text{-}17.9 \times 10^6$	$8.9\text{-}17.9 \times 10^6$
W_P, kg/sec	1.7-3.3	5.4-11.3
P_C, N/m^2	3.4×10^6	3.4×10^6
T_{TI}, °K	1033	1033



THERMO CONDITIONER

Parameter	H_2	O_2
P_G, N/m^2	$0.34\text{-}3.4 \times 10^6$	$0.34\text{-}3.4 \times 10^6$
T_G, °K	1033-1944	1033
T_{HO}, °K	55-166	111-332



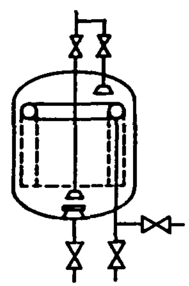


H_2 ACQUISITION

W_{MAX}, kg/sec 3.4

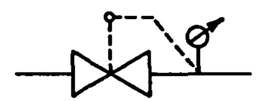
a, $\frac{m}{sec^2}$ -9.8

Qscreen, W → 0



PRIMARY REGULATORS

Parameter	H_2	O_2
P_{IN}, N/m^2	$5.5\text{-}19.3 \times 10^6$	$5.5\text{-}19.3 \times 10^6$
P_{OUT}, N/m^2	4.1×10^6	4.1×10^6
W, kg/sec	1.8	6.8



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BREADBOARD ASSEMBLY AND TEST

The entire APS breadboard effort culminates in the assembly and test of the breadboard system. The breadboard system should simulate the major assembly interfaces, environment (vacuum) for critical areas, and physical locations typical of the vehicles.

MARSHALL SPACE FLIGHT CENTER SCIENCE & ENGINEERING	BREADBOARD ASSEMBLY AND TEST	LAB: S&E-ASTN-PP NAME: J. P. McCarty DATE: April 7, 1971

- Major assembly interfaces
- Environmental simulation (vacuum) for critical areas
- Physical locations typical of vehicles

TECHNOLOGY AND ADVANCED DEVELOPMENT EFFORT

Through the APS breadboard effort early demonstration of the candidate system operation will be achieved including verification of concept feasibility and component design, confirmation of critical component capabilities, and identification of actual or potential problem areas. The base for component development specification requirements will have been established.

MARSHALL SPACE FLIGHT CENTER SCIENCE & ENGINEERING	TECHNOLOGY AND ADVANCED DEVELOPMENT EFFORT	LAB: S&E-ASTN-PP NAME: J. P. McCarty DATE: April 7, 1971

- Early verification of concept feasibility and component design decisions

- Early confirmation of critical component capabilities

- Early identification of problem areas

  -Controls
  -Component Interactions
  -Thermal control
  -Interface requirements

- Provide base for component development specification requirements

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"INJECTOR PERFORMANCE, HEAT FLUX AND FILM COOLING IN O_2/H_2 ENGINES"

R. K. WILLIAMS

J. E. BOUVIER

MANNED SPACECRAFT CENTER

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INJECTOR PERFORMANCE,

HEAT FLUX AND FILM COOLING

IN O_2/H_2 ENGINES

MANNED SPACECRAFT CENTER
THERMOCHEMICAL TEST AREA

ROBERT K. WILLIAMS JAMES E. BOUVIER

THE MANNED SPACECRAFT CENTER (MSC) AT HOUSTON, TEXAS, HAS BEEN CONDUCTING AN IN-HOUSE GASEOUS HYDROGEN/GASEOUS OXYGEN (GH_2/GO_2) ROCKET ENGINE TEST PROGRAM FOR THE PAST 2 YEARS.

THE INITIAL OBJECTIVE OF THE PROGRAM WAS TO GAIN EXPERIENCE WITH USING GH_2 AND GO_2 AS ROCKET ENGINE PROPELLANTS AND RE-ORIENT OUR FACILITIES FROM THE LIQUID, HYPERGOLIC ROCKET ENGINE AND ENGINE SYSTEMS. THE SPECIFIC OBJECTIVES ARE LISTED IN FIGURE 1.

PROGRAM OBJECTIVES

- DEFINE IMPORTANT INJECTOR PARAMETERS
- EVALUATE VARIOUS INJECTOR TECHNIQUES
- EVALUATE GEOMETRY EFFECTS OF COMBUSTION CHAMBER L^*, L' AND CR
- DETERMINE COMBUSTION CHAMBER HEATING RATES
- EVALUATE COMBUSTION CHAMBER COOLING WITH GASEOUS HYDROGEN

FIGURE 1

THREE BASIC TYPES OF INJECTION TECHNIQUES WERE STUDIED. THE FIRST TYPE WAS A BASIC SHOWERHEAD INJECTOR WITH A REMOVABLE INJECTOR PLATE TO ALLOW SEVERAL VARIATIONS OF THE BASIC SHOWERHEAD. FIGURE 2 SHOWS ONE OF THE VARIATIONS - AN IMPINGING STREAM INJECTOR. A HIGH PRESSURE DROP SHOWERHEAD, A LOW PRESSURE DROP SHOWERHEAD AND A DOUBLET ARRAY WERE TESTED. A SPARK IGNITION SYSTEM WAS USED FOR ALL THE ENGINES TESTED.

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IMPINGING STREAM INJECTOR

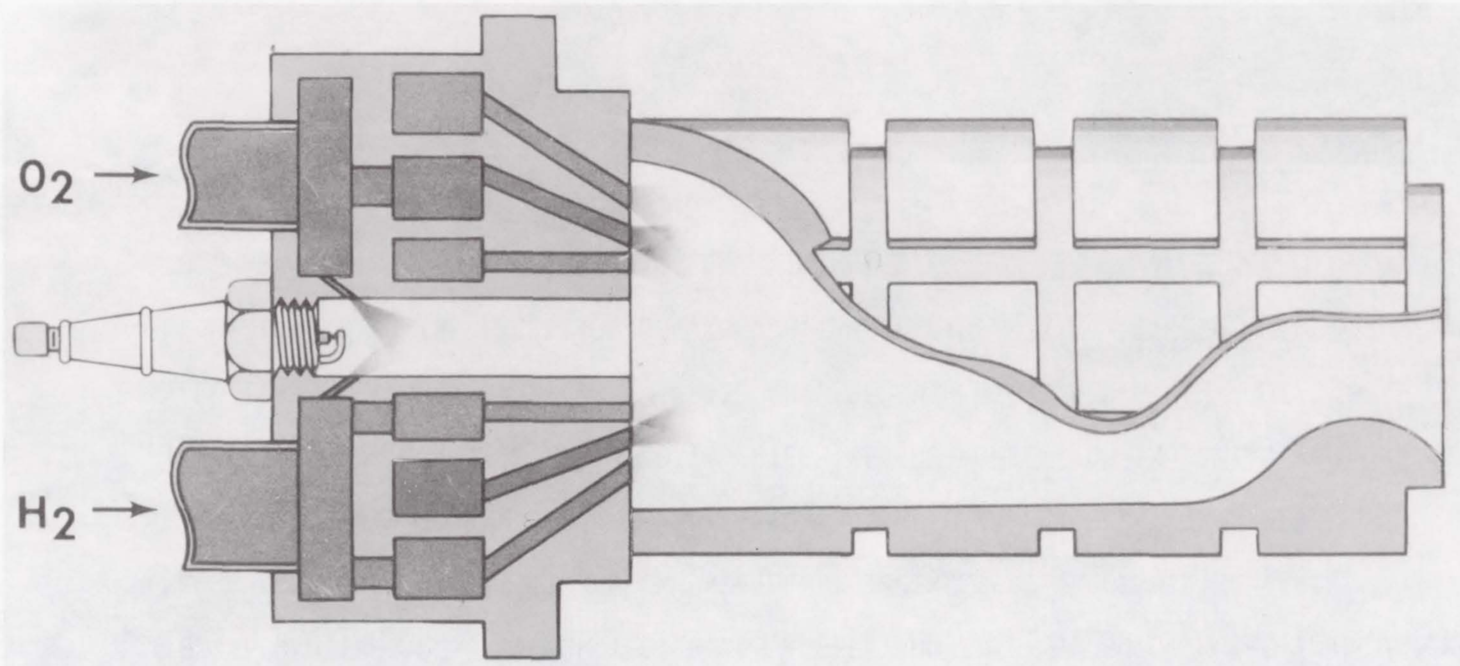

FIGURE 2

THE SECOND TYPE OF INJECTOR TESTED WAS A SHEET-ON-SHEET INJECTOR. THE INJECTOR HAD CHANGEABLE ELEMENTS TO ALLOW PROPELLANT IMPINGEMENT ANGLES TO BE VARIED BETWEEN 90° AND 0° (OR PARALLEL INJECTION). THE PARALLEL SHEET-ON-SHEET INJECTION CONFIGURATION WAS USED TO DETERMINE THE EFFECTS OF INJECTION VELOCITY AND INJECTION VELOCITY RATIO ON PERFORMANCE, AS ALL OTHER MODES OF MIXING, WITH THE EXCEPTION OF VISCOUS, SHEAT TYPE MIXING BETWEEN THE TWO PARALLEL SHEETS, ARE ESSENTIALLY ELIMINATED.

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ADJUSTABLE IMPINGING SHEET INJECTOR

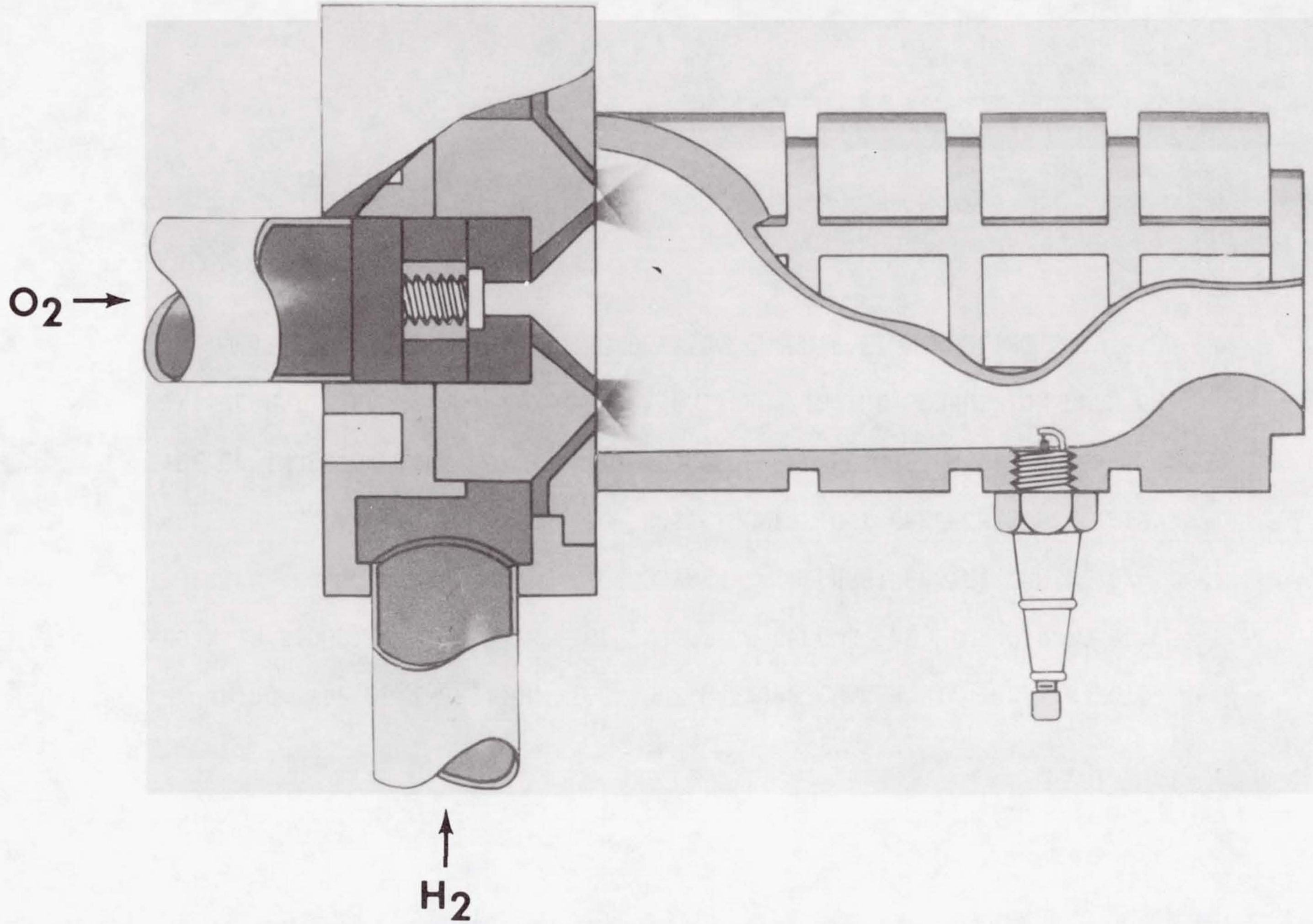

FIGURE 3

THE THIRD TYPE OF INJECTION TECHNIQUE WAS THE CONCENTRIC TUBE INJECTOR AS SHOWN IN FIGURE 4. THIS TYPE OF INJECTION ALLOWS THE HYDROGEN TO BE INJECTED AROUND A CENTRAL CORE OF OXYGEN. THE ENGINE SIZE CAN BE EXPANDED BY EXPANDING THE NUMBER OF BASIC CONETRIC TUBE ELEMENTS. THIS TYPE OF INJECTOR GAVE REASONABLY HIGH PERFORMANCE OVER A WIDE PROPELLANT MIXTURE RATIO RANGE AND WAS USED EXTENSIVELY IN THE COMBUSTION CHAMBER WALL HEATING RATE STUDIES AND COMBUSTION CHAMBER FILM COOLING STUDIES.

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CONCENTRIC TUBE INJECTOR

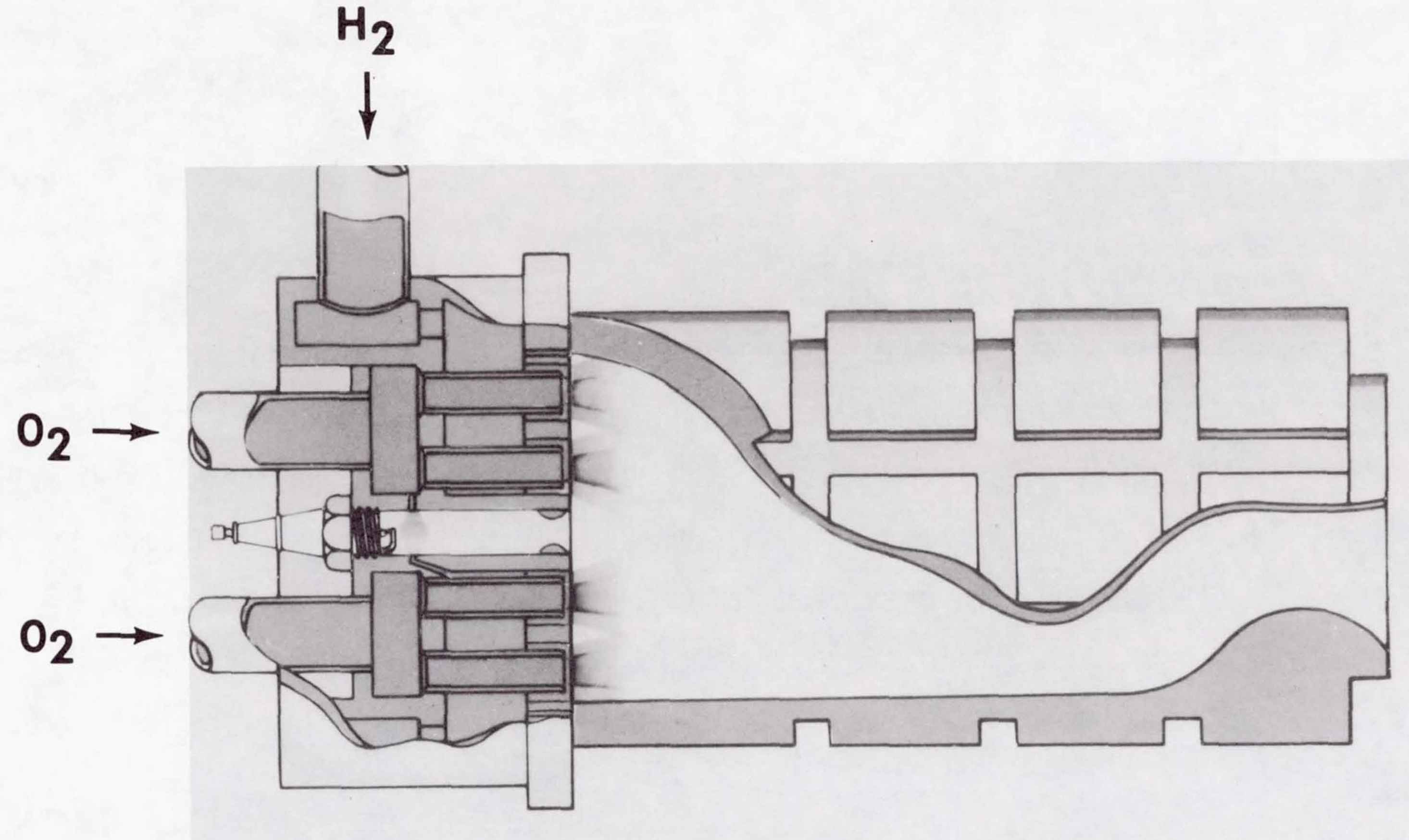

FIGURE 4

FIGURE 5 IS A PICTORIAL OF THE IN-HOUSE DESIGNED HIGH THRUST, LOW CHAMBER PRESSURE GH_2/GO_2 ENGINE. THIS ENGINE HAS OPERATED AT A THRUST LEVEL OF 450 POUNDS (SEA LEVEL) AND A CHAMBER PRESSURE OF 30 PSIA. IT IS ESSENTIALLY A CONCENTRIC TUBE INJECTOR WITH A COMBINATION FILM AND REGENERATIVELY COOLED COMBUSTION CHAMBER. THIS ENGINE HAS OPERATED SUCCESSFULLY FOR FIRINGS UP TO 60 SECONDS DURATION WITH NO DEGRADATION TO THE ALUMINUM THROAT SECTION OR TO THE STAINLESS STEEL COMBUSTION CHAMBER LINER.

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FILM COOLED O_2/H_2 ENGINE

COMBUSTION CHAMBER

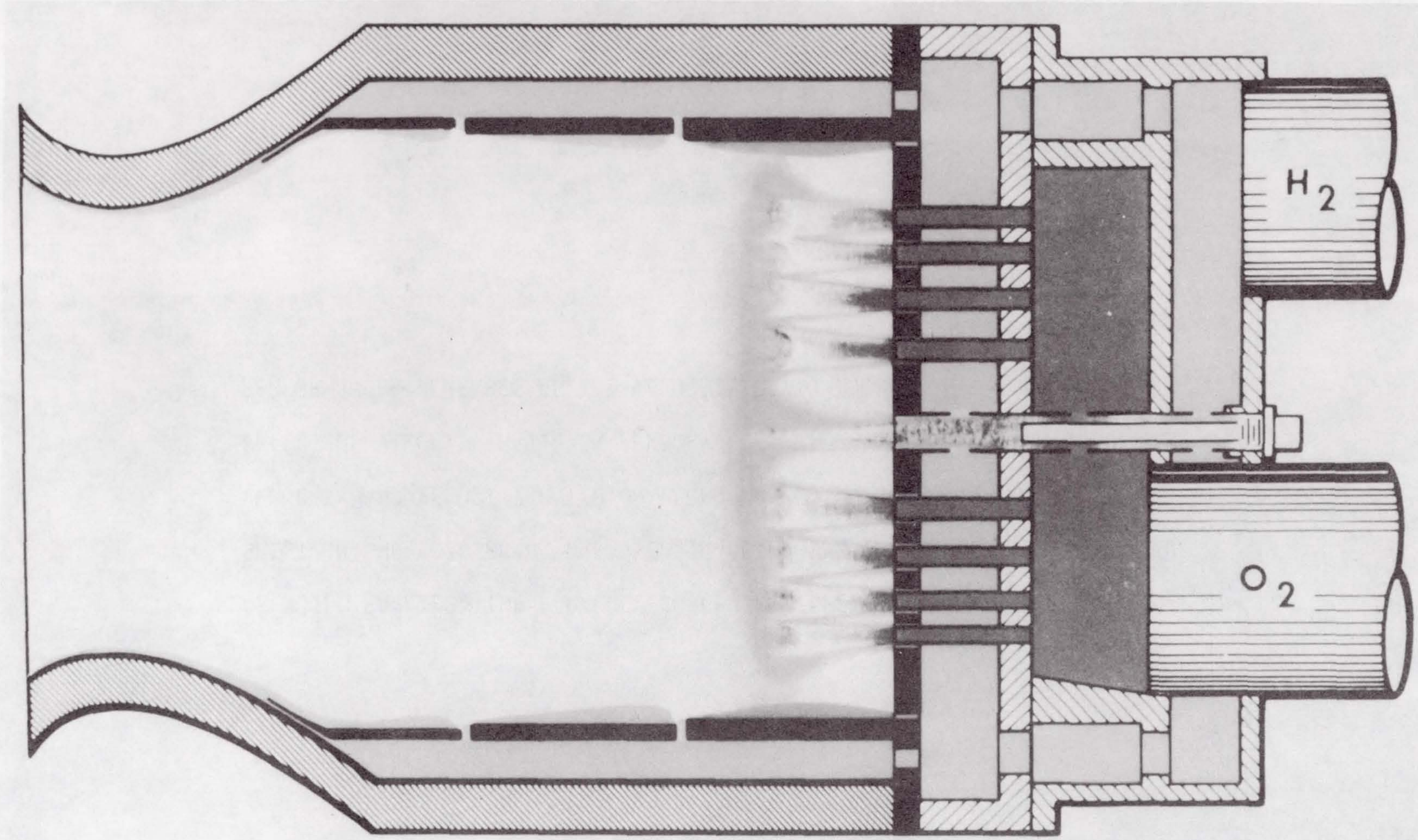

FIGURE 5

AS NOTED EARLIER, THE PARALLEL CONCENTRIC SHEET CONFIGURATION OF THE SHEET-ON-SHEET INJECTOR WAS USED TO DETERMINE THE EFFECTS OF INJECTION VELOCITY AND VELOCITY RATIO ON PERFORMANCE. FIGURE 6 ILLUSTRATES THE EFFECT OF VELOCITY RATIO ON PERFORMANCE. FOR ALL MIXTURE RATIOS, THE PERFORMANCE INCREASES AS THE VELOCITY RATIO INCREASES.

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PARALLEL CONCENTRIC SHEET INJECTOR EFFECTS OF VELOCITY RATIO

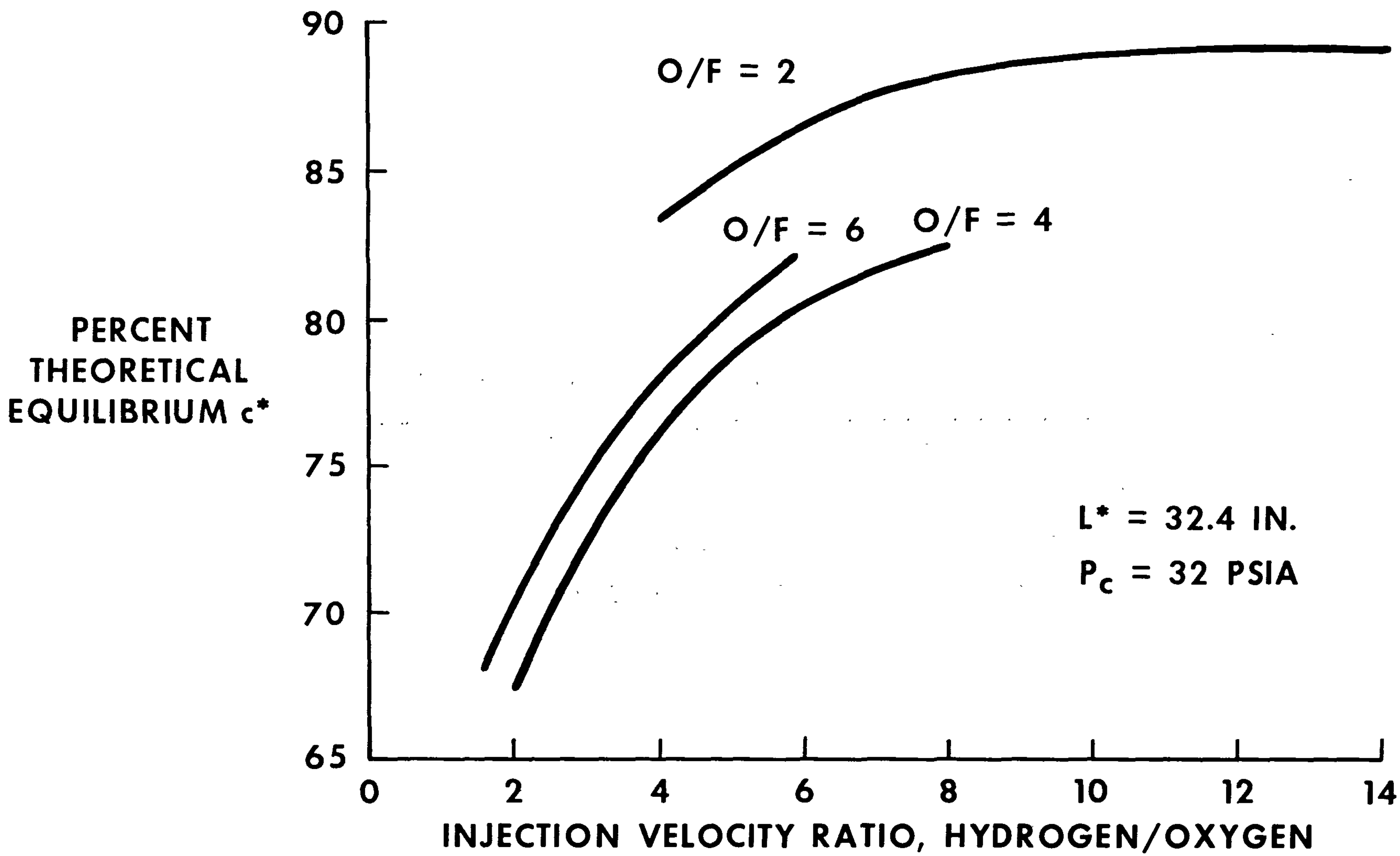

FIGURE 6

THE HIGH THRUST, LOW CHAMBER PRESSURE ENGINE (FIGURE 7) WAS USED TO STUDY THE EFFECTS OF VARYING PROPELLANT INJECTION TEMPERATURES ON PERFORMANCE. THIS ENGINE WAS DESIGNED AROUND A PROPELLANT MIXTURE RATIO OF 3. AT THE DESIGN POINT, THE PERFORMANCE LOSS WHEN INJECTION TEMPERATURES ARE DROPPED FROM ESSENTIALLY AMBIENT (50° F) TO -200° F IS APPROXIMATELY 5%.

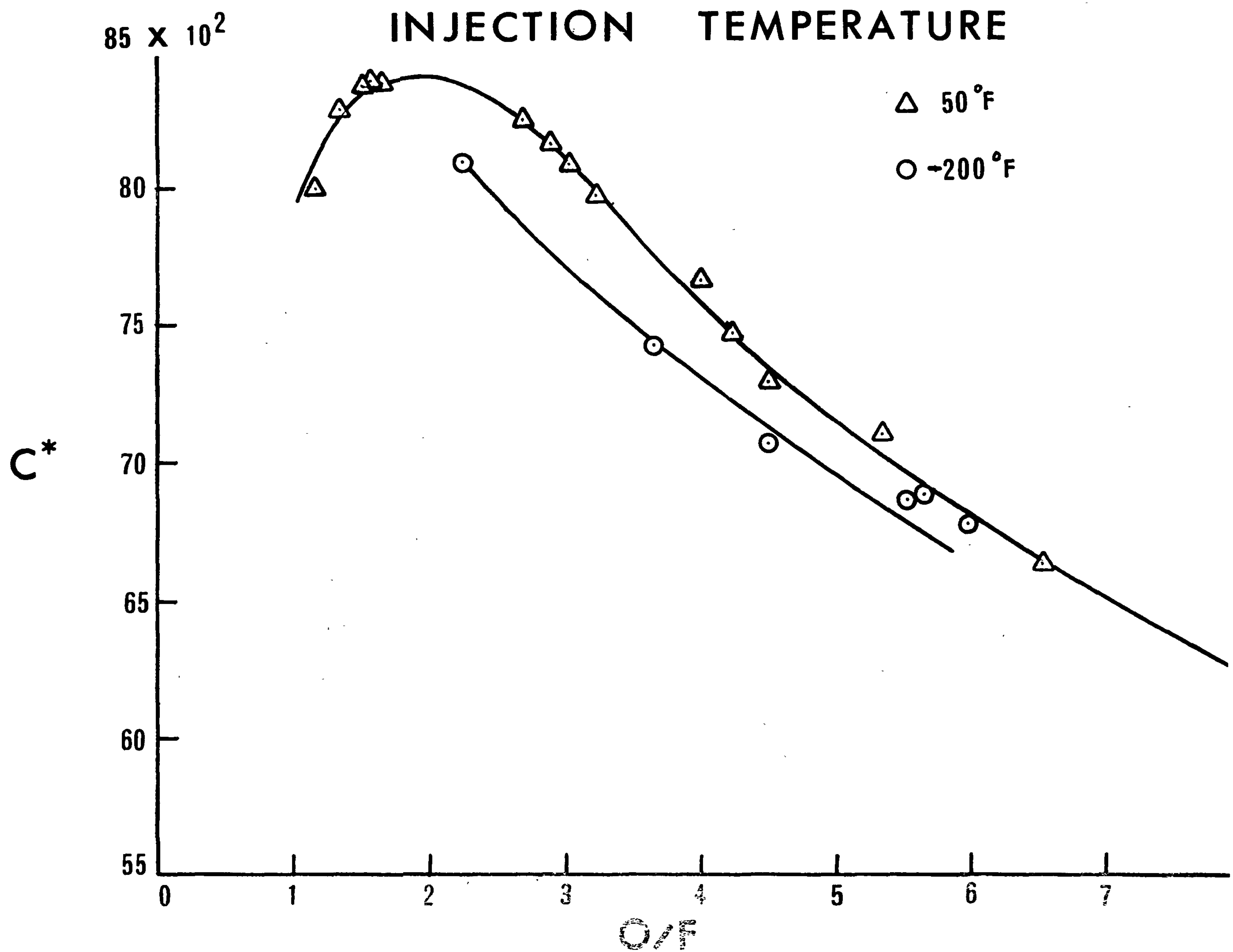
INJECTION TEMPERATURE
85 x 10^2
80
75
70
65
60
55
C*
0
1
2
3
4
5
6
7
O/F
△ 50 °F
○ -200 °F

FIGURE 7

THE EFFECTS ON PERFORMANCE OF COMBUSTION CHAMBER GEOMETRY WAS ALSO STUDIED DURING THE MSC IN-HOUSE PROGRAM. FIGURE 8 ILLUSTRATES THE PERFORMANCE EFFECTS OF VARYING COMBUSTION CHAMBER L* . NOTE THAT AS L* IS INCREASED FROM 10.8 INCHES TO 24.33 INCHES, THE PERFORMANCE OF THE ENGINE INCREASES. HOWEVER, AT THE ENGINE DESIGN MIXTURE RATIO OF 3, THE ENGINE PERFORMANCE DOES NOT CONTINUE TO INCREASE AFTER THE 17.6 INCH LEVEL.

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CONCENTRIC TUBE INJECTOR PERFORMANCE

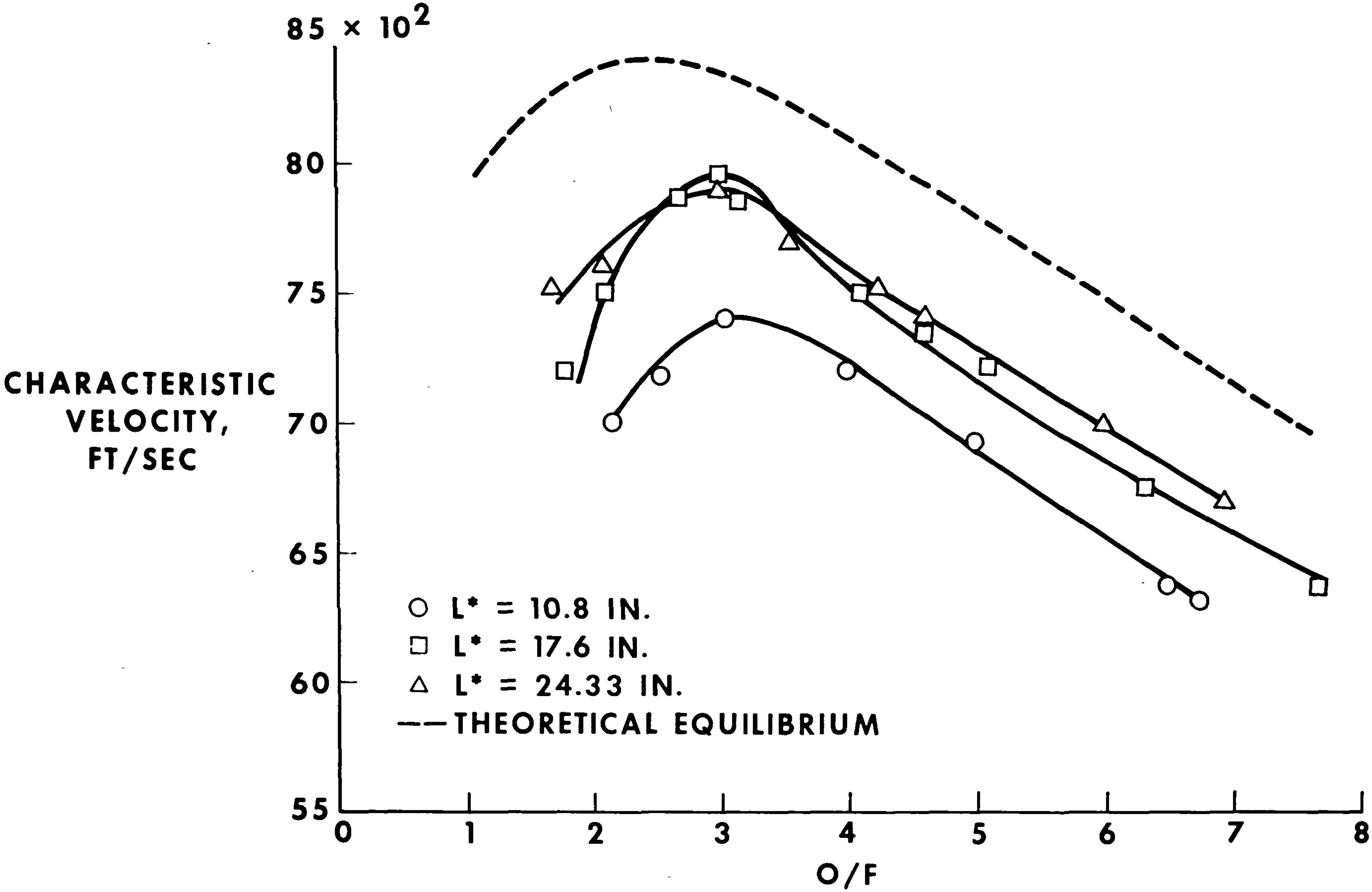

FIGURE 8

THE OVERALL INJECTOR PERFORMANCE STUDY IS SUMMARIZED IN FIGURE 9. THE HIGH ΔP (INJECTOR PRESSURE DROP) SHOWERHEAD INJECTOR YIELDED THE HIGHER PERFORMANCE OVER THE WIDEST MIXTURE RATIO RANGE. HOWEVER, PRESSURE BUDGET LIMITATIONS FOR THE LOW PRESSURE GH_2/GO_2 SYSTEM MAKE THIS INJECTOR TYPE NOT FEASIBLE. IF THE INJECTOR HOLES ARE MADE LARGER TO DECREASE THE PRESSURE DROP, THE PERFORMANCE, AS INDICATED BY THE LOW ΔP SHOWERHEAD, IS SUBSTANTIALLY REDUCED. THE SHEET-ON-SHEET INJECTORS GAVE INTERMEDIATE LEVELS OF PERFORMANCE WITH THE HIGHEST PERFORMANCE OCCURRING FOR THE HIGHEST INCLUDED ANGLE BETWEEN THE PROPELLANT STREAMS. THE MULTIPLE CO-AXIAL OR MULTIPLE CONCENTRIC TUBE INJECTOR YIELDED REASONABLY HIGH PERFORMANCE OVER A WIDE MIXTURE RATIO RANGE, AND WHEN THE VARIOUS COOLING TECHNIQUES WERE INTEGRATED INTO HIGH THRUST, LOW CHAMBER PRESSURE ENGINE, THERE WAS VERY LITTLE PERFORMANCE PENALTY AT THE DESIGN MIXTURE RATIO OF 3.

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INJECTOR PERFORMANCE

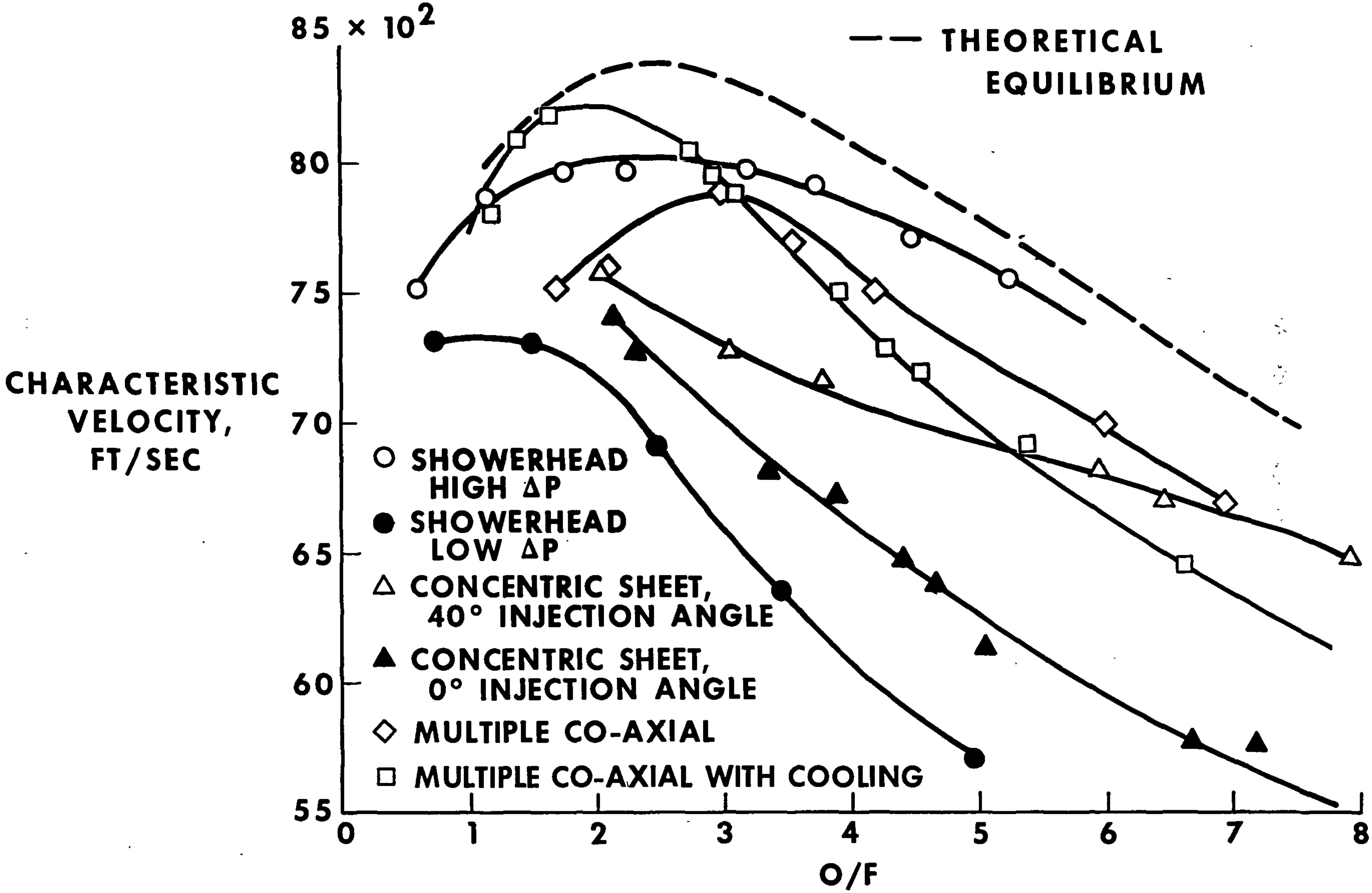

FIGURE 9

FIGURE 10 VERY BRIEFLY SUMMARIZES THE EXTENT OF THE MSC IN-HOUSE TEST EFFORT. NOTE THAT EACH OF THE 2000 SEA-LEVEL FIRINGS WAS AN INDIVIDUAL EFFORT, i.e., NO PULSE MODE, DUTY CYCLE WORK WAS CONDUCTED.

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PERFORMANCE SUMMARY

- APPROXIMATELY 2000 SEA LEVEL FIRINGS AND 3500 SECONDS ACCUMULATED FIRING DURATION
- THRUST RANGE FROM 15 TO 600 POUNDS
- MIXTURE RATIOS FROM 0.5 TO 10.0
- L* FROM 6 TO 70 INCHES
- SEVERAL CONTINUOUS FIRINGS OF 60 SECOND DURATION EACH
- INJECTION TEMPERATURES FROM -200 TO +200 °F

FIGURE 10

FIGURE 11 IS A PHOTOGRAPH OF THE HIGH THRUST, LOW CHAMBER PRESSURE ENGINE FIRING IN THE SEA-LEVEL TEST STAND IN THE THERMOCHEMICAL TEST AREA AT THE MANNED SPACECRAFT CENTER IN HOUSTON. INITIALLY THE COMBUSTION CHAMBER AND THROAT WERE WATER-COOLED. HOWEVER, AS CONFIDENCE GREW IN THE HYDROGEN FILM COOLING TECHNIQUES, THE WATER WAS TURNED OFF FOR RUN DURATIONS OF 60 SECONDS WITH NO HARDWARE DEGRADATION OBSERVED.

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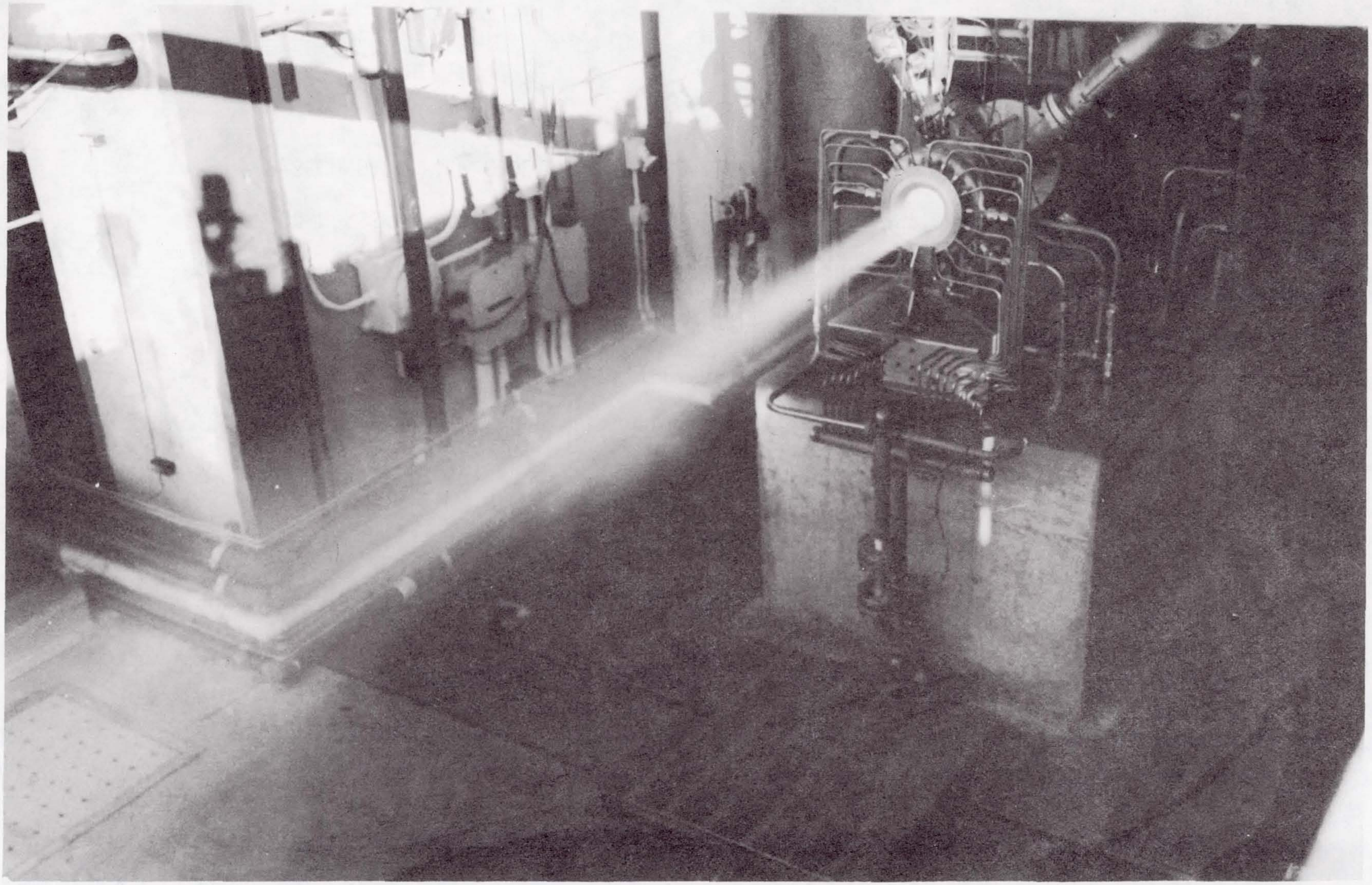

FIGURE 11

THE OBJECTIVE OF THE MSC IN-HOUSE COMBUSTION CHAMBER COOLING STUDY WAS TO DETERMINE COMBUSTION CHAMBER HEATING RATES FOR LOW AND HIGH CHAMBER PRESSURE ENGINES AND TO DETERMINE THE EFFECTIVENESS OF USING GASEOUS HYDROGEN FOR COMBUSTION CHAMBER COOLING. ALL WORK WAS DONE WITH AMBIENT TEMPERATURE GASES. THERE WERE TWO PRINCIPAL TYPES OF COOLING STUDIED, FILM COOLING AND REGENERATIVE COOLING.

FIGURE 12 IS A PICTURE OF THE COMBUSTION CHAMBER ASSEMBLY USED IN THE LAST PHASE OF THE LOW CHAMBER PRESSURE, FILM COOLED ENGINE STUDY. THIS PICTURE ILLUSTRATES THE INTEGRATION OF THE VARIOUS COOLING MECHANISMS INTO A SINGLE COMBUSTION CHAMBER DESIGN. SHOWN ARE THE PRIMARY FILM COOLING PORTS AT THE INJECTOR, THE ANNULUS AROUND THE COMBUSTION CHAMBER USED FOR REGENERATIVE OR DOWNPASS DUMP COOLING AND THE SLOTS IN THE COMBUSTION CHAMBER LINER USED FOR SECONDARY FILM COOLING AND THROAT COOLING.

FILM COOLED O_2/H_2 ENGINE

COMBUSTION CHAMBER

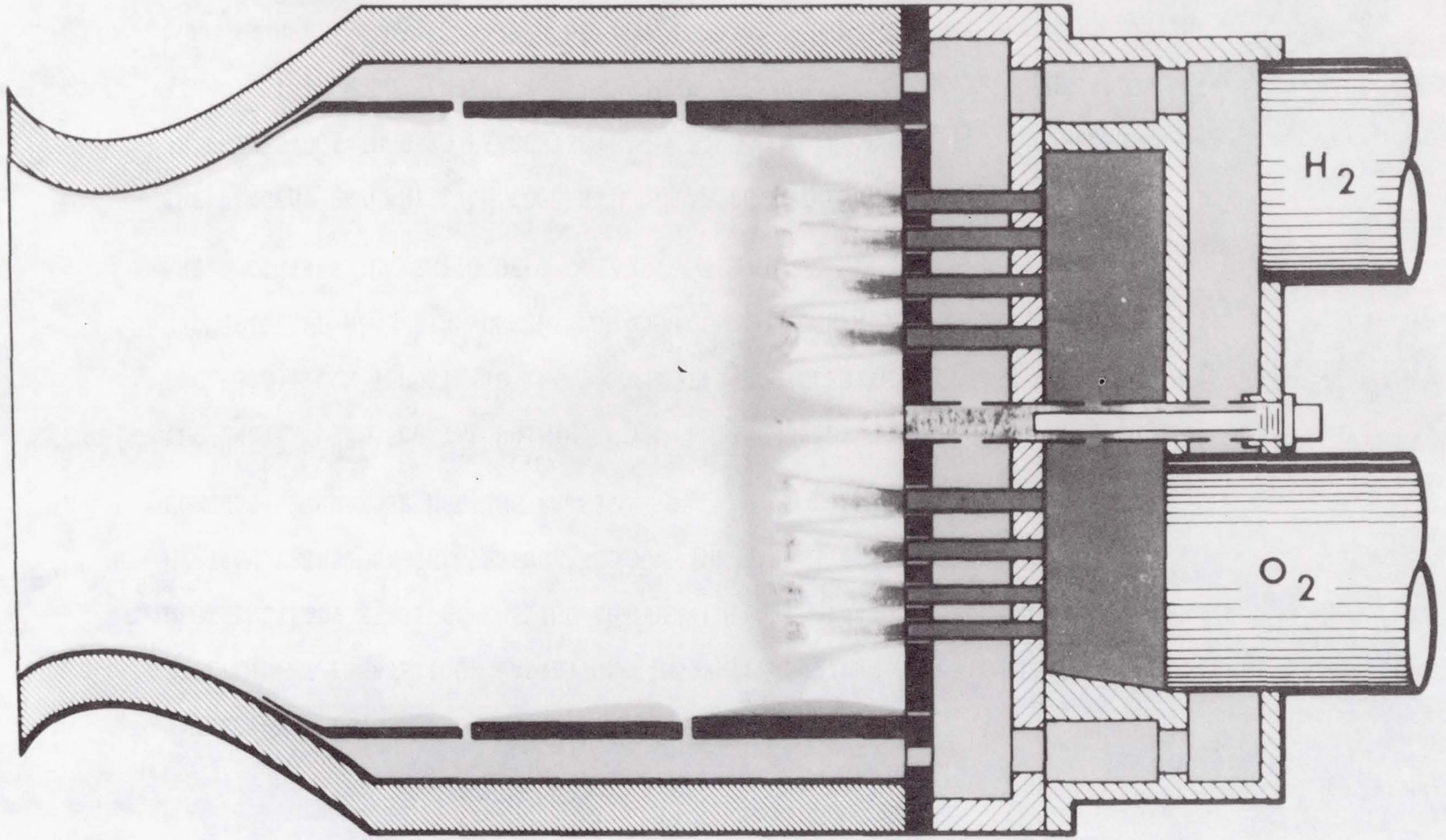

FIGURE 12

INITIALLY, TESTS WERE CONDUCTED WITH PRIMARY FILM COOLING ONLY IN ORDER TO DETERMINE THE EFFECT OF VARIATIONS IN PERCENT PRIMARY FILM COOLING ON PROJECTED STEADY STATE COMBUSTION CHAMBER WALL TEMPERATURES, COMBUSTION CHAMBER HEATING RATES, ENGINE PERFORMANCE AND THE EFFECT OF CHAMBER PRESSURE ON COMBUSTION CHAMBER HEATING RATES.

THE ENGINE USED FOR THE PRIMARY FILM COOLING STUDY WAS A LOW CHAMBER PRESSURE (30 PSIA), LOW THRUST (18 LB AT SEALEVEL) ENGINE WITH A CONCENTRIC TUBE INJECTOR. THE HEAT SINK COMBUSTION CHAMBER CONFIGURATION HAD AN L^* OF 17.6 AND A CONTRACTION RATIO OF 3.

THE PERCENT PRIMARY FILM COOLING WAS VARIED BETWEEN 0 AND 33 PERCENT. FIGURE 13 POINTS OUT THAT 20 PERCENT PRIMARY FILM COOLING EFFECTIVELY LOWERED THE STEADY STATE WALL TEMPERATURES APPROXIMATELY 40 PERCENT. THE COMBUSTION CHAMBER HEATING RATES VARIED BETWEEN 0.3 AND 3 BTU/IN^2- SEC DEPENDING ON THE AXIAL LOCATION DOWN THE COMBUSTION CHAMBER WALL, THE MIXTURE RATIO AND PERCENT PRIMARY FILM COOLING.

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FILM COOLING EFFECTIVENESS

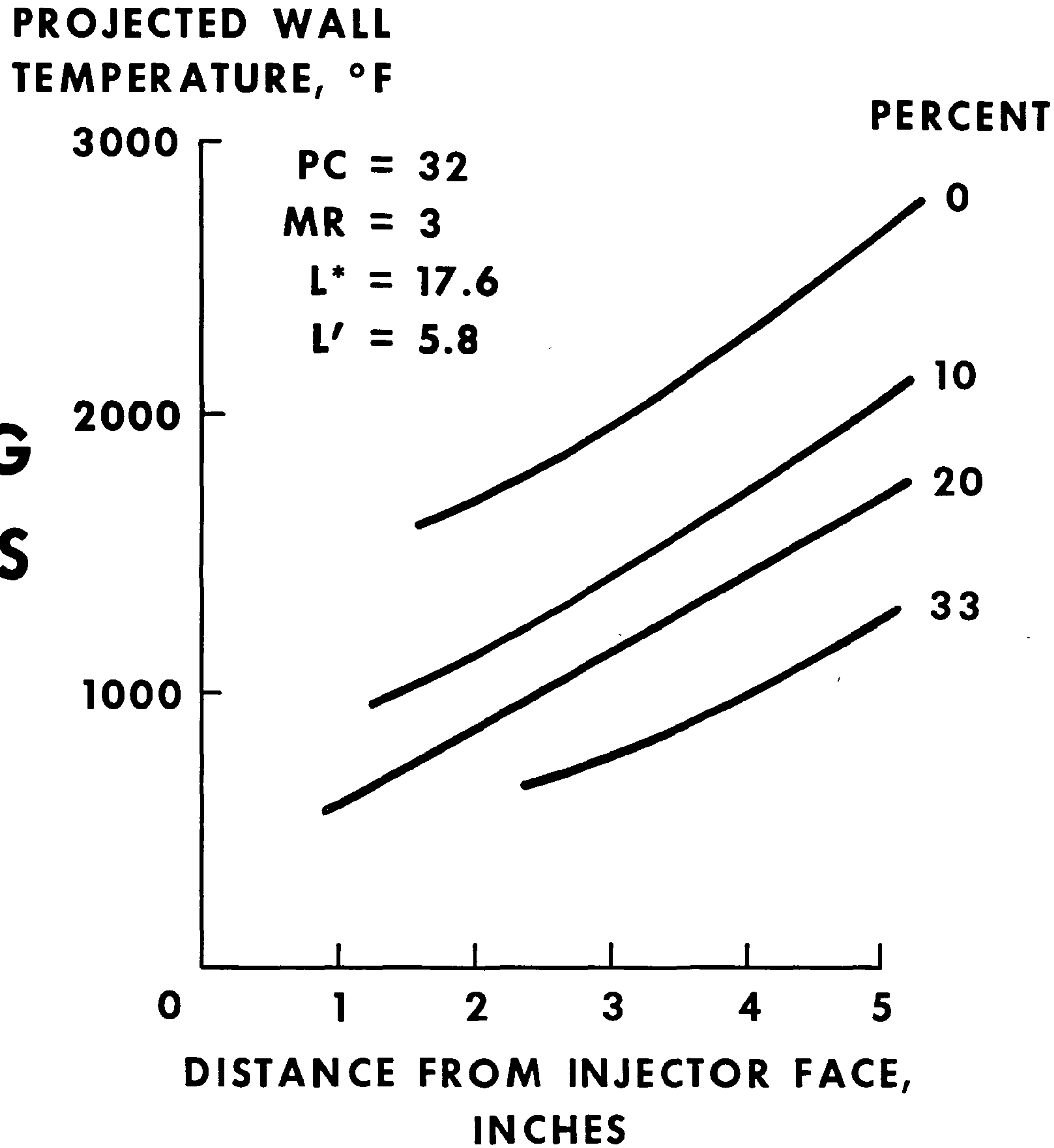

FIGURE 13

FIGURE 14 ILLUSTRATES THE EFFECT OF VARIATIONS IN PERCENT PRIMARY FILM COOLING ON ENGINE PERFORMANCE BASED ON C^*. THE PERFORMANCE LEVEL FOR MIXTURE RATIOS BETWEEN 3 AND 5 AND PERCENT PRIMARY FILM COOLING BETWEEN 0 AND 20 WAS 94 TO 96 PERCENT. TESTS WERE CONDUCTED WITH 20 PERCENT FILM COOLING WITH A MEASURED PERFORMANCE PENALTY OF LESS THAN 2 PERCENT BETWEEN MIXTURE RATIOS OF 3 AND 5. IN FACT, 10 PERCENT FILM COOLING WAS USED WITH NO MEASURED PERFORMANCE PENALTY.

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O/F VS C*

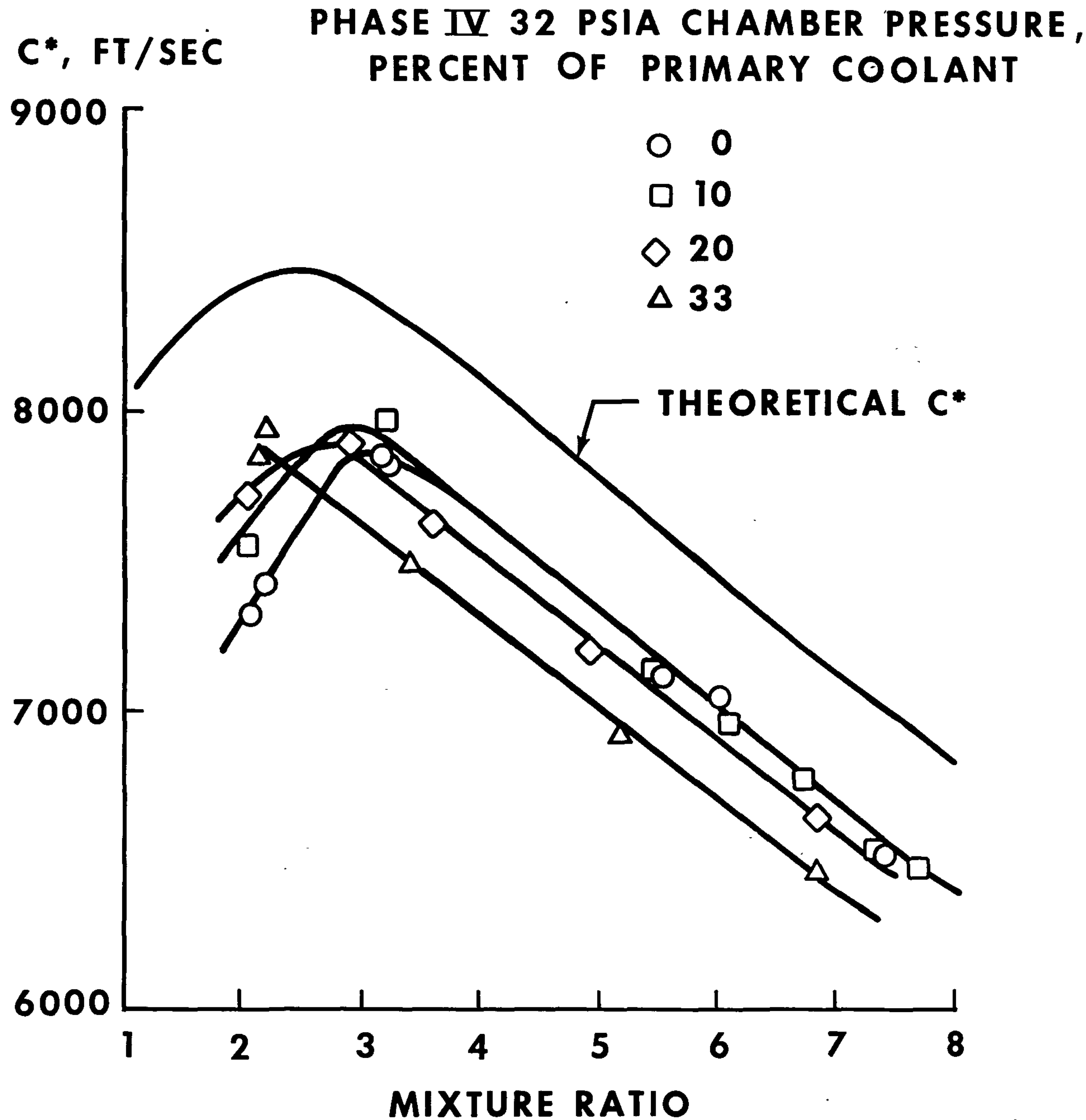

FIGURE 14

THE SAME ENGINE CONFIGURATION USED FOR LOW CHAMBER PRESSURE DATA WAS USED TO OBTAIN HIGH CHAMBER PRESSURE ENGINE DATA. FIGURE 15 ILLUSTRATES THE EFFECT OF VARIATIONS IN PERCENT PRIMARY FILM COOLING ON COMBUSTION CHAMBER HEATING RATES. TWENTY PERCENT FILM COOLING LOWERED THE COMBUSTION CHAMBER HEATING RATES TO THE APPROXIMATE LEVEL OF THE HEATING RATES FOR 30 PSIA CHAMBER PRESSURE INDICATED BY THE DOTTED LINE. AN ADDITIONAL 13 PERCENT COOLING DID NOT SIGNIFICANTLY LOWER THE HEATING RATES BELOW THAT FOR 20 PERCENT COOLING. THIS DATA WAS TAKEN APPROXIMATELY 4.5 INCHES FROM THE INJECTOR FACE.

$\frac{Q}{A}$ VS T_W

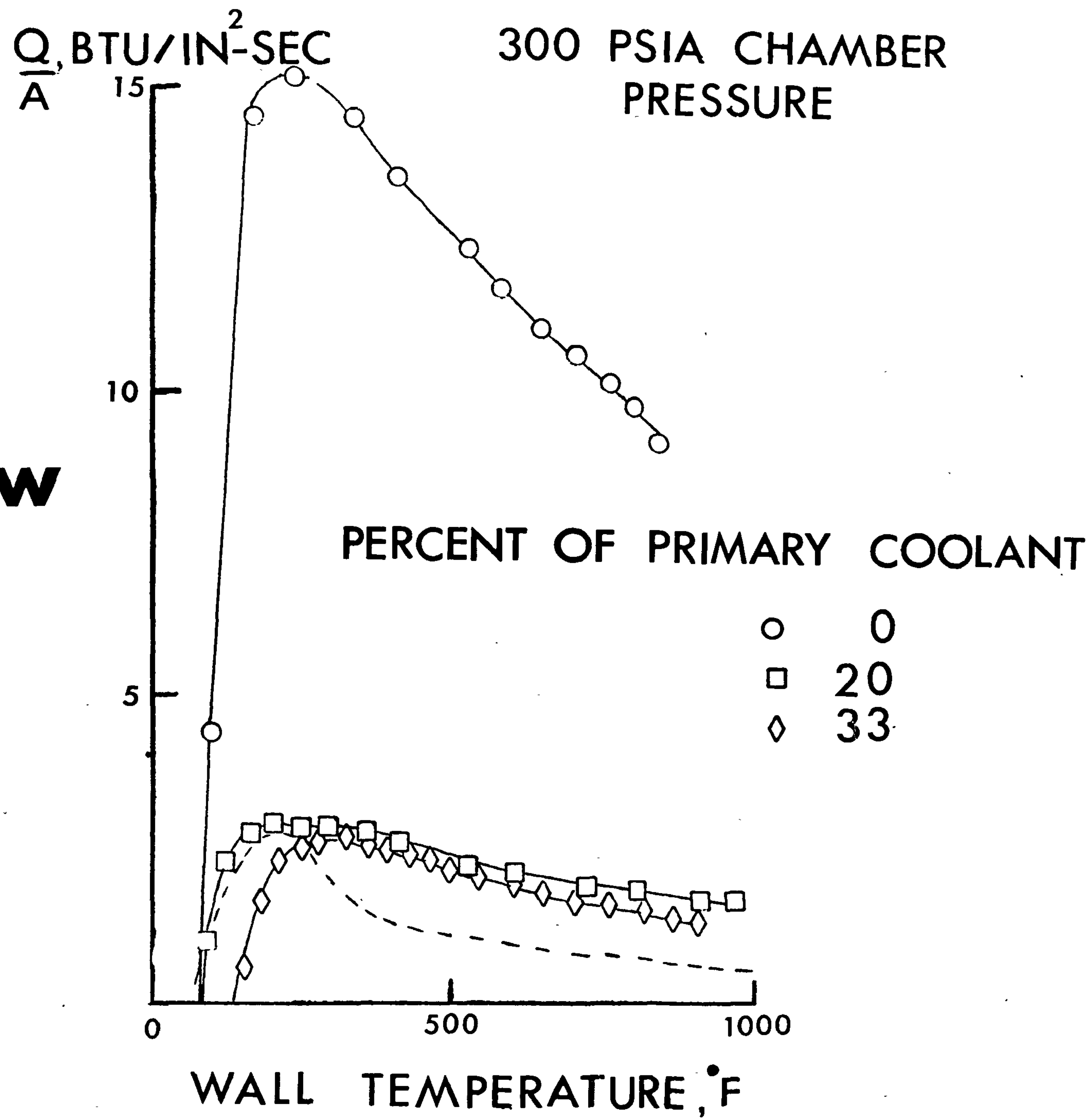

FIGURE 15

FIGURE 16 SHOWS THE EFFECT OF VARIATIONS IN PERCENT PRIMARY FILM COOLING ON ENGINE PERFORMANCE BASED ON C^* AT A CHAMBER PRESSURE OF 300 PSIA. TWENTY PERCENT FILM COOLING WAS USED WITH A MEASURED PERFORMANCE PENALTY OF LESS THAN 2 PERCENT FOR MIXTURE RATIOS BETWEEN 3 AND 5. IN FACT, FOR THE DESIGN MIXTURE RATIO OF 3, THERE WAS NO MEASURED PERFORMANCE PENALTY. THE PERFORMANCE LEVEL FOR 20 PERCENT FILM COOLING BETWEEN MIXTURE RATIOS OF 3 AND 5 WAS 95 to 97 PERCENT.

O/F VS C*

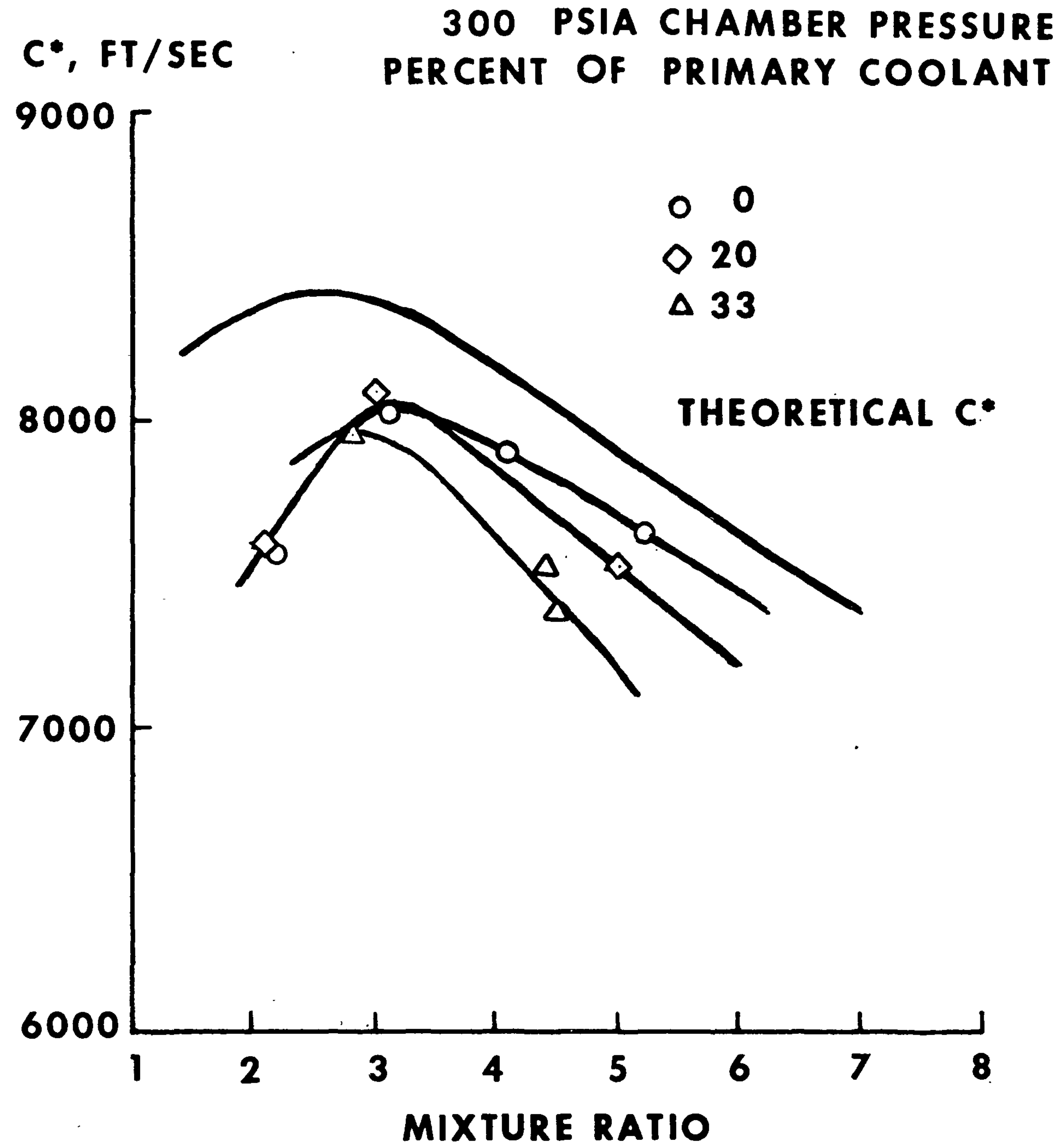

FIGURE 16

AT THE CONCLUSION OF THE PRIMARY FILM COOLING STUDY, IN ORDER TO REDUCE THE COMBUSTION CHAMBER WALL TEMPERATURES, THE COMBUSTION CHAMBER ASSEMBLY USED IN THE PRIMARY FILM COOLING STUDY WAS MODIFIED TO ADAPT REGENERATIVE OR DOWNPASS DUMP COOLING. A COPPER COMBUSTION CHAMBER WITH A STAINLESS STEEL LINER HAVING THE SAME L^* AND CONTRACTION RATIO AS THAT USED IN THE PRIMARY FILM COOLING STUDY WAS USED. THE DATA ON FIGURE 17 IS FROM A 20 SECOND STEADY STATE FIRING. TWENTY THREE PERCENT REGENERATIVE COOLING WAS USED WITH 23 PERCENT PRIMARY FILM COOLING. A PERFORMANCE LEVEL OF 95 PERCENT WAS MAINTAINED WITH A MAXIMUM COMBUSTION CHAMBER LINER TEMPERATURE OF ONLY 840^o F.

REGENERATIVE COOLED ENGINE

LOW PRESSURE • LOW THRUST

CHAMBER PRESSURE	30 PSIA
THRUST	18 LBF
REGENERATIVE COOLING	23%
PERFORMANCE	95%
MAXIMUM COMBUSTION CHAMBER LINER TEMPERATURE	840°F

FIGURE 17

AFTER IT WAS DEMONSTRATED THAT THE LOW THRUST, LOW CHAMBER PRESSURE ENGINE COULD BE COOLED WITH STEADY STATE OPERATION AND MAINTAIN A HIGH PERFORMANCE LEVEL, THE COMBUSTION CHAMBER HARDWARE WAS MODIFIED TO ADAPT A COMBINATION OF PRIMARY FILM COOLING, SECONDARY FILM COOLING AND REGENERATIVE COOLING. ALTHOUGH IT WAS DESIRABLE TO INTRODUCE THE HYDROGEN FOR SECONDARY FILM COOLING INTO THE COMBUSTION CHAMBER AT A SHALLOW ANGLE, ANGLES OF LESS THAN 15 DEGREES WERE DIFFICULT TO MACHINE IN THE COMBUSTION CHAMBER LINER. A COMPROMISING ANGLE OF 15 DEGREES WAS USED INITIALLY AS INDICATED IN FIGURE 18. THIS CREATED AN UNFAVORABLE FLOW SITUATION. FOR COOLANT FLOW RATES HIGHER THAN 17 PERCENT A SUBSTANTIAL AMOUNT OF THE INJECTED HYDROGEN WAS SEPARATING FROM THE COMBUSTION CHAMBER WALL AFFORDING LITTLE OR NO THERMAL PROTECTION TO THE WALL. IN FACT, AS THE COOLANT MASS FLOW WAS REDUCED AND SEPARATION WAS SUPPRESSED SOMEWHAT, THE WALL TEMPERATURE DECREASED DUE TO THE ADDITIONAL HYDROGEN ATTACHING TO THE WALL. SEPARATION WAS MINIMIZED AT 17 PERCENT. USING THE SAME INJECTION ANGLE, LARGER INJECTION PORTS WERE USED TO LOWER THE COOLANT INJECTION VELOCITIES (DOTTED LINES). THIS IMPROVED THE SITUATION BUT DID NOT CORRECT IT. THE DATA INDICATED THAT IF SECONDARY FILM COOLING WAS TO BE USED EFFECTIVELY, INJECTION PARALLEL TO THE COMBUSTION CHAMBER WALL WAS NECESSARY.

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FILM COOLING EFFECTIVENESS
15 DEGREE INJECTION ANGLE

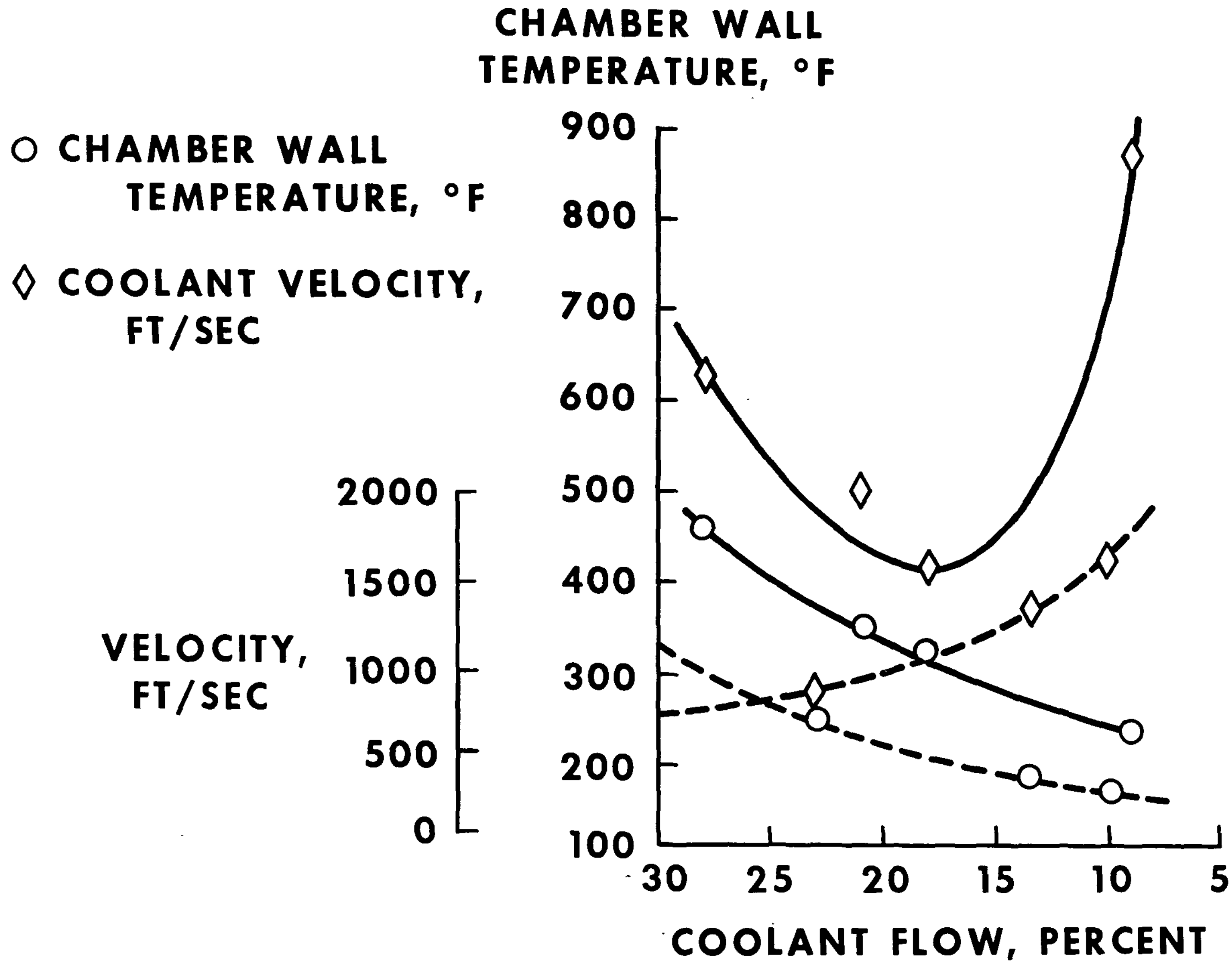

FIGURE 18

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THIS CONCEPT WAS IMPLEMENTED INTO THE DESIGN OF THE COMBUSTION CHAMBER FOR THE FILM COOLED LOW CHAMBER PRESSURE - HIGH THRUST ENGINE (FIGURE 12). FIGURE 19 SHOWS THAT IN THIS 20 SECOND STEADY STATE FIRING, WHERE APPROXIMATELY 5 PERCENT OF THE HYDROGEN WAS USED FOR PRIMARY FILM COOLING AND 20 PERCENT WAS USED FOR REGENERATIVE AND SECONDARY FILM COOLING, A PERFORMANCE LEVEL OF 95 PERCENT WAS MAINTAINED, YET THE MAXIMUM COMBUSTION CHAMBER LINER TEMPERATURES WAS ONLY 325° F. THIS DATA INDICATES THAT THE AMOUNT OF HYDROGEN USED FOR COOLING COULD BE REDUCED SUBSTANTIALLY AND THUS FURTHER INCREASE ENGINE PERFORMANCE.

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LOW CHAMBER PRESSURE - HIGH THRUST FILM COOLED ENGINE

CHAMBER PRESSURE	30 PSIA
THRUST LEVEL	450 LBF
STEADY STATE	20 SECONDS
PERFORMANCE	95 PERCENT
MAXIMUM COMBUSTION CHAMBER LINER TEMPERATURE	325 °F

FIGURE 19

IN SUMMARY, THE SUCCESSFUL STEADY STATE FIRING OF THE FILM COOLED LOW PRESSURE - HIGH THRUST ENGINE, AND THE DEMONSTRATION THAT THE HEATING RATES AT A CHAMBER PRESSURE OF 300 PSIA COULD BE REDUCED TO THE LEVEL FOR 30 PSIA WITH LITTLE OR NO PERFORMANCE PENALTY, INDICATES THAT GASEOUS HYDROGEN CAN BE USED EFFECTIVELY TO COOL THE COMBUSTION CHAMBER FOR HIGH PRESSURE ENGINES AS WELL AS LOW PRESSURE ENGINES.

HEAT TRANSFER SUMMARY

- FILM COOLING FEASIBLE WITHOUT SIGNIFICANT PERFORMANCE LOSS

- STEADY STATE FIRING OF 450 POUND THRUST ENGINE WITH FILM COOLING

FIGURE 20

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"AUXILIARY PROPULSION SYSTEM DEFINITION STUDY"

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SPACE SHUTTLE HIGH AND LOW PRESSURE AUXILIARY PROPULSION SUBSYSTEM DEFINITION

PREPARED BY:

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CONTRACTS NAS 8-26249
NAS 9-11013

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OBJECTIVES

THE OBJECTIVES OF THE HIGH AND LOW PRESSURE AUXILIARY PROPULSION SUBSYSTEM (APS) STUDIES WERE TO DEFINE FOR BOTH BOOSTER AND ORBITER ELEMENTS THOSE O_2/H_2 APS CONFIGURATIONS CAPABLE OF MEETING THE SPACE SHUTTLE REQUIREMENTS. BOTH PROGRAMS WERE DIVIDED INTO TWO SUBTASKS. IN SUBTASK A THE EFFORTS WERE DIRECTED TO CONFIGURE AND EVALUATE CONCEPTUAL APS CANDIDATES FOR BOTH BOOSTER AND ORBITER ELEMENTS AS SPECIFIED IN THE NASA AUXILIARY PROPULSION SUBSYSTEM SSVDRD OF 1 JULY 1970 WHEREIN THE APS WOULD BE CAPABLE OF THE FOLLOWING:

- BASELINE ANALYSIS - (1) APS ORBITER PERFORMS ALL VEHICLE ANGLES
  - (2) ALL TRANSLATION MANEUVERS, X Y Z
  - (3) ALL TRANSLATION MANEUVERS EXCEPT PLUS X TRANSLATION BURNS $>$ 50 FPS
  - (4) ALL TRANSLATION MANEUVERS EXCEPT PLUS X TRANSLATION $>$ 10 FPS

THE SELECTED SYSTEMS WERE TO ALSO BE EVALUATED IN TERMS OF:

- MINIMUM NEW TECHNOLOGY
- SIMPLICITY
- RELIABILITY
- SYSTEM PERFORMANCE

IN SUBTASK B SELECTED SYSTEMS FROM SUBTASK A WERE TO BE EVALUATED IN DETAIL TO ESTABLISH APS DESIGN REQUIREMENTS. THE EFFORTS WERE CULMINATED WITH AN IDENTIFICATION OF THE CRITICAL TECHNOLOGY AREAS FOR THE SELECTED CYCLES.

CONTRACT TASKS

THE CONTRACT SUMMARY TASKS WERE AS FOLLOWS:

SUBTASK A

CONFIGURE AND EVALUATE CONCEPTUAL AUXILIARY PROPULSION SUBSYSTEM CANDIDATES FOR BOTH BOOSTER AND ORBITER ELEMENTS, SSVDRD, 1 JULY 1970.

- SUBSYSTEM SCHEMATICS
- THERMODYNAMIC BALANCES
- COMPONENT PARAMETRICS
- PRELIMINARY OPERATING POINTS
- IDENTIFICATION OF CRITICAL TECHNOLOGY
- RECOMMEND BASELINE SYSTEMS

SUBTASK B

CONDUCT PRELIMINARY DESIGN AND ANALYSIS SELECTED SYSTEMS WITH UPDATED SSVDRD REQUIREMENTS OF 15 OCTOBER 1970.

- SUBSYSTEM DEFINITION
- THERMODYNAMIC BALANCES
- APS CYCLE DYNAMICS/ CONTROL
- RELIABILITY/MAINTAINABILITY
- VEHICLE INTEGRATION
- IDENTIFICATION OF CRITICAL TECHNOLOGY

THE EFFORTS WERE CONDUCTED BETWEEN 1 JULY 1970 AND 1 FEBRUARY 1971.

HIGH PRESSURE AUXILIARY PROPULSION SUBSYSTEM SUMMARY

The Space Shuttle High Pressure Auxiliary Propulsion Subsystem (APS) Definition Program was an analytical effort to investigate, for both booster and orbiter baseline vehicles, candidate oxygen-hydrogen auxiliary propulsion subsystems capable of meeting the Space Shuttle APS requirements. The high pressure APS concepts included only those wherein the propellant pressures were raised to levels higher than the propellant storage pressures in the shuttle vehicle main tanks.

The studies indicate that a pumped and heated liquid cycle can satisfy the common needs of both the booster and the orbiter vehicle. The stored propellant pressure is increased from 50 psi to 1100 psi using a turbopump driven by separate gas generators and the liquid is vaporized and heated using a heat exchanger with energy supplied from a separate gas generation source. Conditioned propellants are fed to a propellant distribution system consisting of feedline accumulators, propellant distribution manifold and pressure regulators. Both the booster and the orbiter use a common 1840 lb thruster. The booster vehicle requires 26 common thrusters and the orbiter 32 at a mixture ratio of 4 to 6 and a chamber pressure of 300 psia.

The selected APS was analyzed regarding its capability to perform both a high velocity and a low velocity increment orbiter mission. Performance is of prime importance for the high velocity increment missions because of the 27,952 pounds of propellants involved in the total system weight of 35,021 pounds. For the low velocity increment mission with only 3921 pounds of propellants in the total system weight of 8543 pounds, performance is a secondary consideration and simplicity and high reliability become paramount.

The same single propellant conditioning unit was used for both vehicles with propellant manifolding running to the ends of the vehicle as required to supply propellant for the thrusters. Regulators were placed at each end of the vehicle so that a long manifold at high pressure was available to provide additional accumulator volume required to supply propellant.

The selected APS concept does not require a great deal of new technology before it can be implemented for use on the space shuttle. Two areas, however, do require additional investigation. They are: propellant acquisition and instrumentation particularly for isolation detection that can be done in a highly reliable manner.

LOW PRESSURE AUXILIARY PROPULSION SUBSYSTEM SUMMARY

The Space Shuttle Low Pressure Auxiliary Propulsion Subsystem (APS) Definition Program was a six and one-half month technical effort to investigate the possibility of utilizing a complete low pressure propellant supply system for accomplishing the Space Shuttle Vehicle axuiliary propulsion system requirements. Candidate systems were considered only where they could function on a pressure schedule compatible with the vehicle main tank ullage vent pressures and provide gaseous propellant flow to the thrusters. Specifically, the effort was directed to establish a preliminary auxiliary propulsion subsystem design and define critical technology areas for the various shuttle vehicle concepts.

The selected booster cycle is a blowdown system with only two moving conditioner parts: a main tank APS on-off valve operated once during the mission, and a separator regulator used to vent liquid from a vapor/liquid separator which provides gaseous propellant flows to the thrusters. No supplementary propellants of any type are required. The selected thruster operating point is at a chamber pressure of 11.5 psia, a mixture ratio of 2.5 and an ε of 3:1. Forty 965 lbf thrusters are used in a sequenced maneuvering mode to minimize the size of the distribution system. The MR of 2.5 results in a minimum propellant withdrawal from the ascent tanks. The predicted total booster APS weight is 6,865 lbs. Its total impulse capability is directly dependent on the main tank residuals.

The orbiter APS requirements and main tank thermodynamics indicate a need for the use of supplementary propellants. The main ascent tanks are insufficiently insulated to allow efficient utilization of the maximum possible residuals in its longer mission, as could be done in the booster in its short 6 minute mission. The selected cycle utilizes a mixing carburetor wherein ascent tank gas is mixed with supplementary fluids which have been selectively conditioned. Both active (gas generator) and passive (vehicle sensible heat) heat exchangers are possible in this cycle. The passive system will perform adequately, if its area and mass are large enough.

The orbiter APS has forty-four 1032 lbf thrusters operating at a chamber pressure of 25 psia, an area ratio of 5:1 and a mixture ratio of 3.1. The cycle utilizes helium pressuri-ation for simplicity in the supplementary tanks. The weights are 17,990 lbs for the active system and 17,569 lbs for the passive system.

BASIC CONSTRAINTS

THE BASIC CONSTRAINTS PLACED ON THE TWO EFFORTS WERE AS FOLLOWS:

LOW PRESSURE

- MUST OPERATE WITH A PRESSURE SCHEDULE DICTATED BY THE MAIN VEHICLE TANK ULLAGE PRESSURES
- MAXIMIZE USE OF THE MAIN TANK RESIDUALS
- DELIVER GASEOUS PROPELLANT TO THRUSTERS
- INCORPORATE FAIL OPERATIONAL/FAIL SAFE PRINCIPLES IN DESIGN TO MEET SSVDRD ACCELERATION REQUIREMENTS

HIGH PRESSURE

- EMPHASIZE OVERALL SYSTEM PERFORMANCE
- DETERMINE BEST USE OF ON-BOARD PROPELLANTS
- DELIVER GASEOUS PROPELLANTS TO THRUSTERS
- INCORPORATE FAIL OPERATIONAL/FAIL SAFE PRINCIPLES IN DESIGN

BOTH SYSTEMS

- NO OTHER ON-BOARD INTEGRATION

SUBTASK A ACTIVITY

THE VEHICLES SPECIFIED IN THE 1 JULY 1970 SSVDRD HAD SUFFICIENT DIFFERENCES THAT A WIDE RANGE OF CONCEPTS WERE POSSIBLE TO BE CONSIDERED FOR BOTH THE LOW AND HIGH PRESSURE SYSTEMS. THE REQUIRED NOMINAL MISSION, APS FLOW RATES WERE ALSO QUITE WIDE AND BECAUSE OF THESE VARIATIONS AND EXPECTED VEHICLE CHANGES THE SUBTASK A EFFORTS WERE DEVOTED LARGELY TO THE GENERATION OF COMPLETE PARAMETRIC DATA FOR THE VARIOUS COMPONENTS OF THE CONCEPTUAL CYCLES. THESE DATA WERE THEN USED TO ANALYTICALLY DETERMINE THE PERFORMANCE OF THE VARIOUS CYCLES OVER A WIDE RANGE OF FLOW RATES, AS A FUNCTION OF PRESSURE, TEMPERATURE AND THRUSTER MIXTURE RATIO.

THE CONCEPTS EVALUATED IN THE SUBTASK A EFFORTS FOR BOTH EFFORTS AND THEIR APPLICABILITY ARE SUMMARIZED IN TABLES I AND II. THE FACTORS OF MINIMUM TECHNOLOGY REQUIREMENTS, SIMPLICITY, AND RELIABILITY WERE ALSO APPLIED TO THE VARIOUS CONCEPTS IN ORDER TO SELECT CONCEPTS FOR SUBTASK B EVALUATION.

TABLE I. HIGH PRESSURE CYCLES AND VEHICLE APPLICABILITY

	Cycle Number	Basic Cycle	Variations of Basic Cycle	Applicability: Bipropellant Booster	Bipropellant Orbiter L	Bipropellant Orbiter I	Bipropellant Orbiter H	Monopropellant Booster	Monopropellant Orbiter L	Monopropellant Orbiter I	Monopropellant Orbiter H
(1) Pumped Liquid	1	High Pressure Gas Generator for Propellant Heating	Indirect Heating	X	X	X	X				
	2			X	X	X	X				
	3	Low Pressure Gas Generator for Propellant Heating	Direct Heating	X	X	X	X				
Gas Compression	4		No Heating	X	X						
	5		No Heating					X			
	6	High Pressure Gas Generator for Propellant Heating	Direct Heating	X	X						
	7	Low Pressure Gas Generator for Propellant Heating	Direct Heating	X	X						
	8		Direct Heating					X	(2) X		
Stored Gas	9	Low Pressure Gas Generator for Propellant Heating	No Heating	X							
	10		Direct Heating	X	X						
(1) Supercritical	11	High Pressure Gas Generator for Propellant Heating	Direct Heating	X	X						
	12	Low Pressure Gas Generator for Propellant Heating	Direct Heating	X	X	X					

(1) Pumped liquid and supercritical cycles can also use small propellant storage tanks refillable from the orbit maneuvering engine turbopumps for L (low) and I(intermediate) orbiter velocity increment.

(2) Separate makeup tanks using the structure as a passive heat exchanger can extend the range still further. Most applicable as a monopropellant oxygen system.

TABLE 2. LOW PRESSURE CYCLES AND VEHICLE APPLICABILITY

	BOOSTER			ORBITER								
				HIGH DELTA-V			INTERMEDIATE DELTA-V			LOW DELTA-V		
	Propellant Storage	Conditioning		Propellant Storage	Conditioning		Propellant Storage	Conditioning		Propellant Storage	Conditioning	
		Active*	Passive**		Active*	Passive**		Active*	Passive**		Active*	Propellant**
GOX MONOPROPELLANT	X	X	X									
GH_2 MONOPROPELLANT	X	X	X									
DUAL MONOPROPELLANT	X	X	X									
BIPROPELLANT		X	X	X	X		X	X	X	X	X	X

*USING GAS GENERATOR AND HEAT EXCHANGER
**USING ASCENT TANKS OR VEHICLE STRUCTURE

APS OPERATING POINT SELECTION

THE FOLLOWING FACTORS WERE CONSIDERED TO DETERMINE THE APS OPERATING POINT:

- PERFORMANCE - OPTIMIZE THRUSTER CHAMBER PRESSURE WITH COOLANT REQUIREMENTS
- THRUSTERS - MIXTURE RATIO SHIFTS DUE TO TEMPERATURE EXCURSIONS
- PROPULSIVE VENT - USE HIGH AS PRESSURE AS POSSIBLE
- PHYCHROMETRIC - REACTION PRODUCTS CONDENSATION AND FREEZING, VENTS, GAS GENERATORS, THRUSTERS AND HEAT EXCHANGERS
- PROPELLANT PROPERTIES - OXYGEN COMPRESSIBILITY AND TWO-PHASE FLOW
- IGNITION - TWO-PHASE FLOW EFFECTS
- DISTRIBUTION SYSTEM CONSTRAINTS - PROPELLANT CONDENSATION
- ACCUMULATOR BLOWDOWN RATIOS - RAPID DRAWOFF, ISOTHERMAL DRAWOFF, RECOMPRESSION
- ENVIRONMENTAL - REACTOR SOAKOUT, ENVIRONMENTAL SOAKOUT
- PROPELLANT USE MR VARIABILITY - SIZING AND PLACEMENT OF STORAGE BOTTLES
- CYCLE PERFORMANCE REQUIREMENTS - HIGH DELTA-V VERSUS LOW DELTA-V

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HIGH PRESSURE APS SUMMARY OPERATING POINT FINDINGS

THE FOLLOWING OPERATING POINTS WERE SELECTED FOR THE PUMPED LIQUID CYCLE:

MIXTURE RATIO

- LOW DELTA-V   4 ◄— MIXTURE RATIO —► 6   HIGH DELTA-V

THRUSTER CHAMBER PRESSURE

- 300 PSIA

REGULATOR PRESSURE

- EXIT - 400 PSIA
- MINIMUM INLET - 600 PSIA

GAS GENERATOR

- MIXTURE RATIO ≈ 1:1.

CONDITIONER

- HIGH DELTA-V: P_{COND} = 1140 PSIA, T_H = 265°R, T_O = 315°R
- LOW DELTA-V: P_{COND} = 1140 PSIA, T_H = 400-500°R, T_O = 400-500°R

DETAILED PERFORMANCE TRADES REQUIRED TO FINALIZE SELECTIONS

LOW PRESSURE APS SUMMARY OPERATING POINT FINDINGS

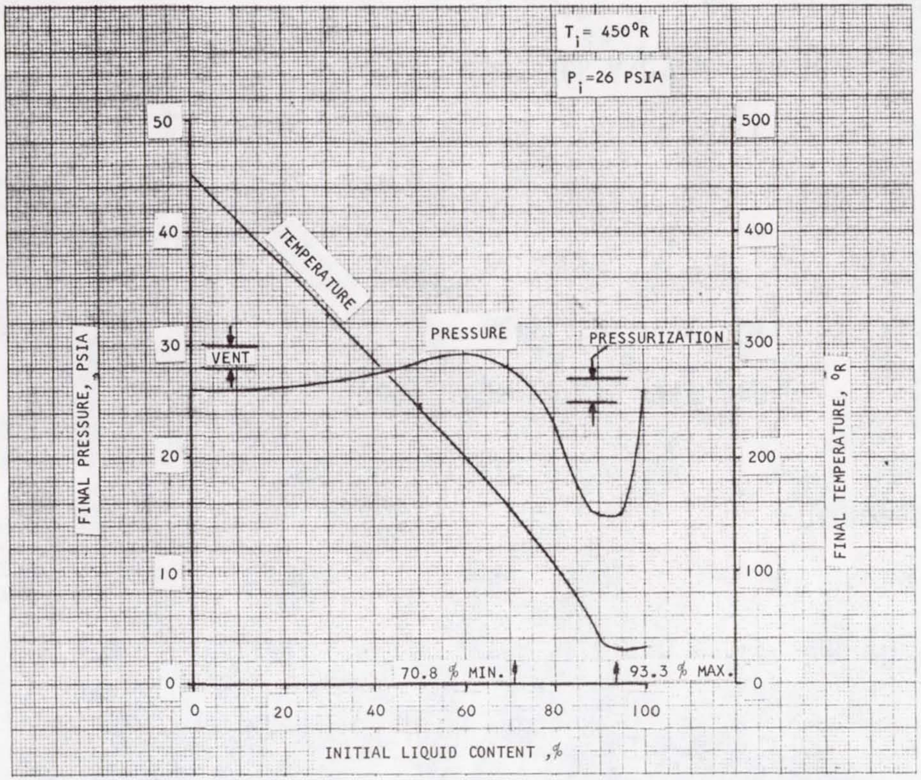
THE FOLLOWING OPERATING POINTS WERE SELECTED:

	BOOSTER	ORBITER
MIXTURE RATIO	2.5	6.0
THRUSTER CHAMBER PRESSURE, PSIA	11.5	25.0
MINIMUM ASCENT TANK PRESSURE, PSIA*	15.0	29.0
CONDITIONER EXIT TEMPERATURE, °R	37/163	200/200
MINIMUM CONDITIONER EXIT PRESSURE, PSIA	13.5	28.5

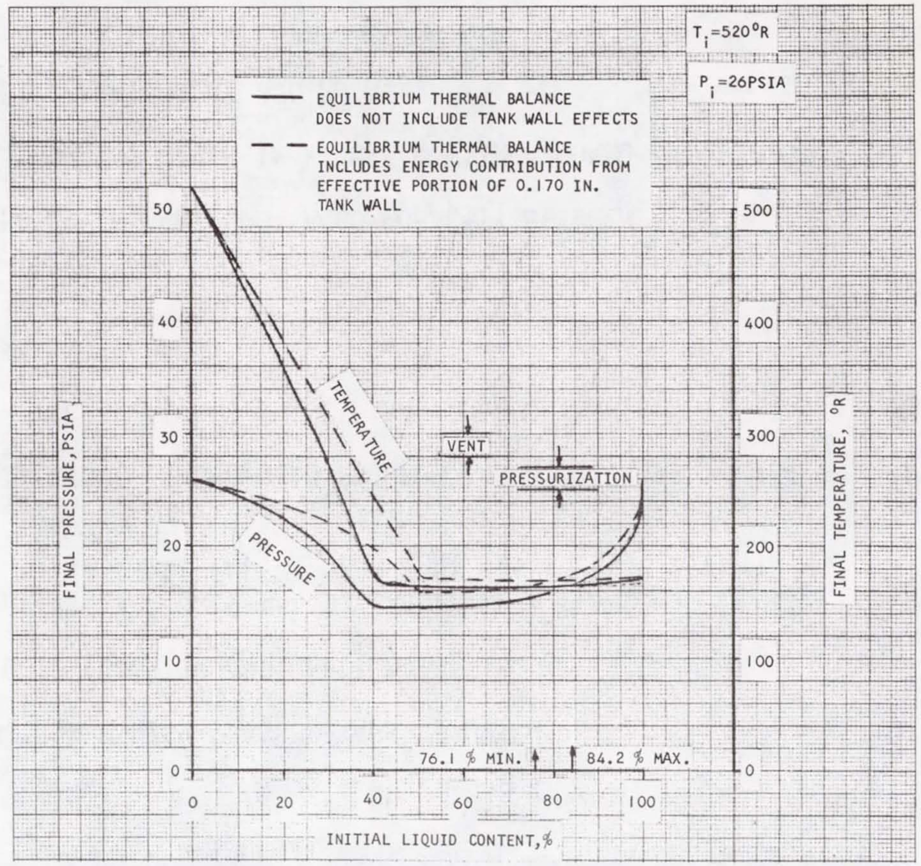
*BASED ON ASCENT TANK EQUILIBRATION ANALYSIS SHOWN IN FIGURE 1.

DETAILED PERFORMANCE TRADES REQUIRED TO FINALIZE SELECTIONS.

TANK EQUILIBRATION RESULTS



BOOSTER EQUILIBRIUM OXYGEN TANK



BOOSTER EQUILIBRIUM HYDROGEN TANK

FIGURE 1

SUBTASK B

SELECTED CYCLES

HIGH PRESSURE

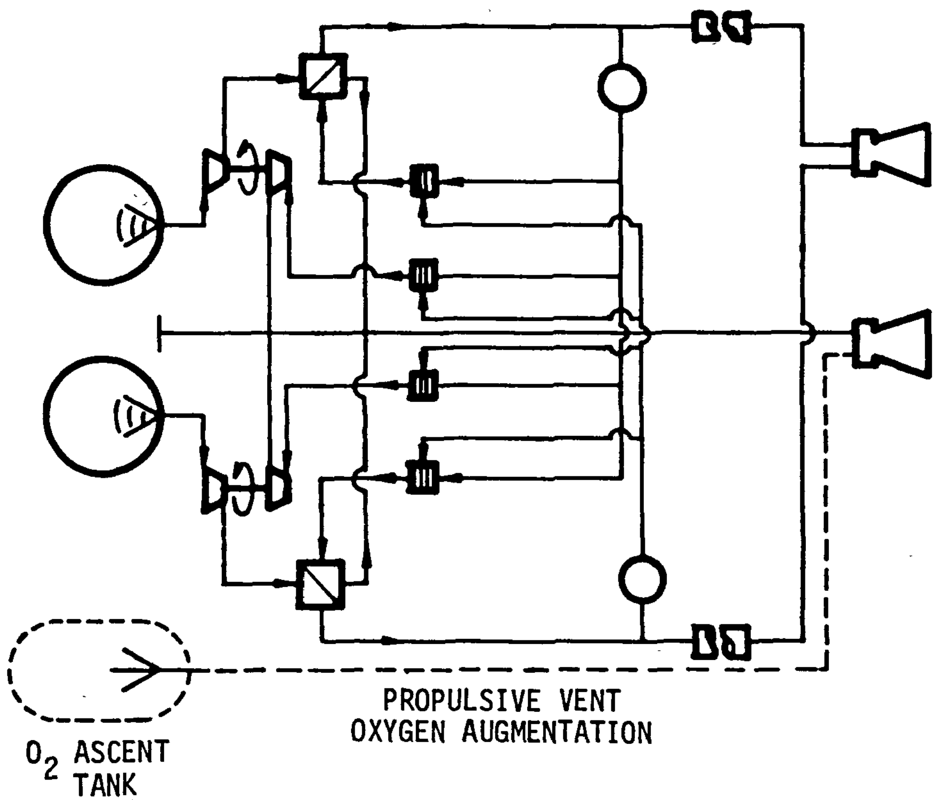
THE SELECTED CYCLES FOR THE SUBTASK B HIGH PRESSURE APS EFFORT WERE:

- PUMPED LIQUID (BOOSTER AND ORBITER)
- COMPRESSOR (BOOSTER ONLY AND DISCARDED WHEN FOUND HIGHLY DEPENDENT UPON TANK GAS EQUILIBRATION)

THE BASIC SELECTED PUMPED LIQUID CYCLE IS SHOWN IN FIGURE 2 AS A DUAL GAS GENERATOR CYCLE WITH ITS MAJOR CYCLE FEATURES. THE FINAL THERMODYNAMIC BALANCES FOR THE CYCLE ARE SHOWN IN FIGURE 3.

THE SELECTED CYCLE SIZING WAS BASED IN PART UPON A COMMONALITY BETWEEN BOOSTER AND ORBITER TO MEET THE DEMANDS GIVEN IN FIGURE 4 WITH A COMMON THRUSTER SIZE OF 1840 LBF THRUST OPERATING AT 300 PSIA.

SELECTED CYCLE



CYCLE FEATURES

- SEQUENCE CONTROLS TO MATCH EQUIPMENT DEMANDS
- OFF-LINE START ACCUMULATORS TO OPTIMIZE STARTUP, PROVIDE SAFING STARTUP
- ABSORBS THRUSTER CYCLING MR VARIABILITY
- ADAPTABLE TO SEVERAL CONTROL LOGICS
  (1) HIGH ΔV - PERFORMANCE
  (2) LOW ΔV - SIMPLICITY
- PRESSURIZATION
  (1) LOW ΔV - HE
  (2) HIGH ΔV - HE, O_2
  AUTO, H_2

FIGURE 2.

BASIC PUMPING CYCLE FLOW DIAGRAM

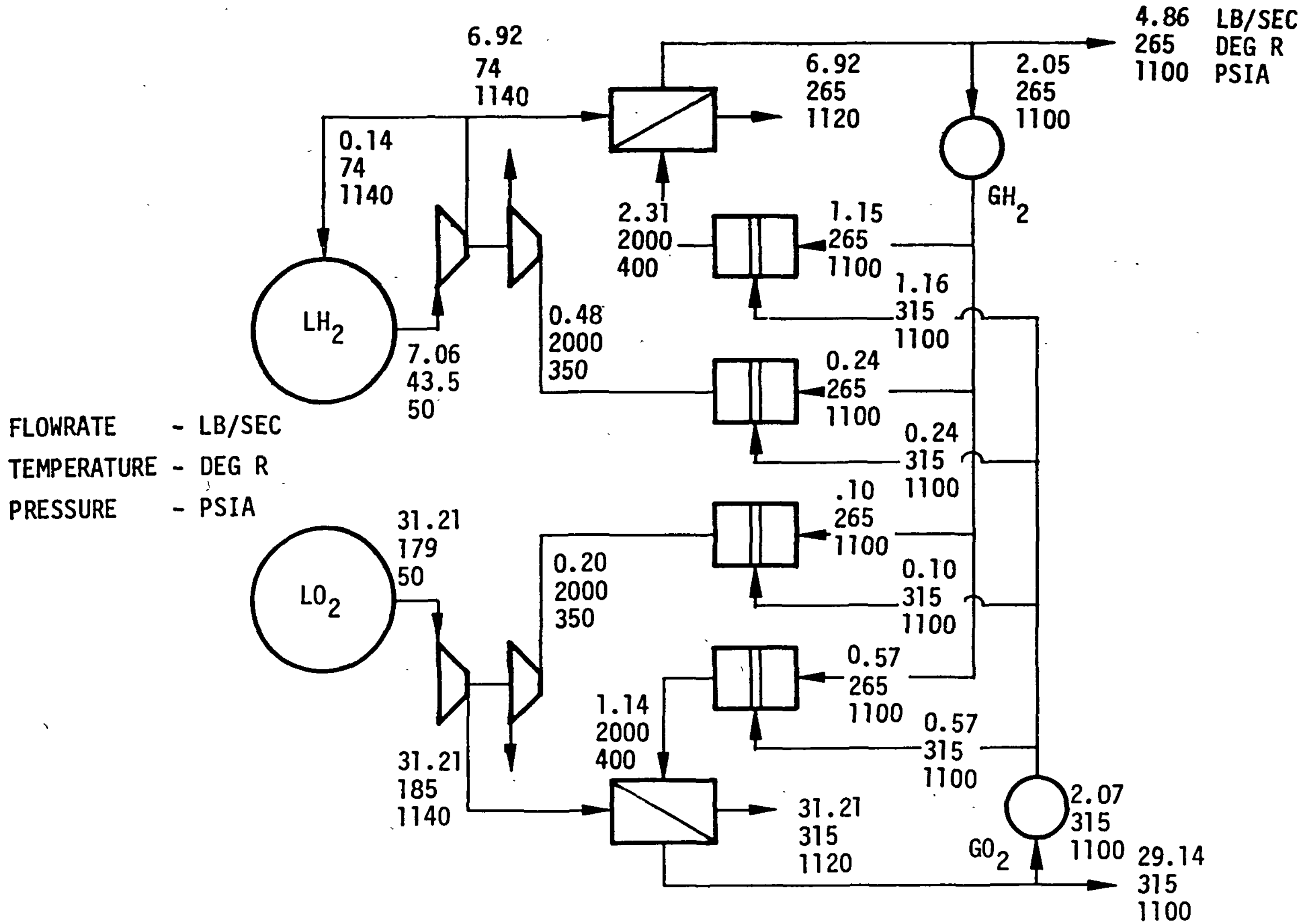
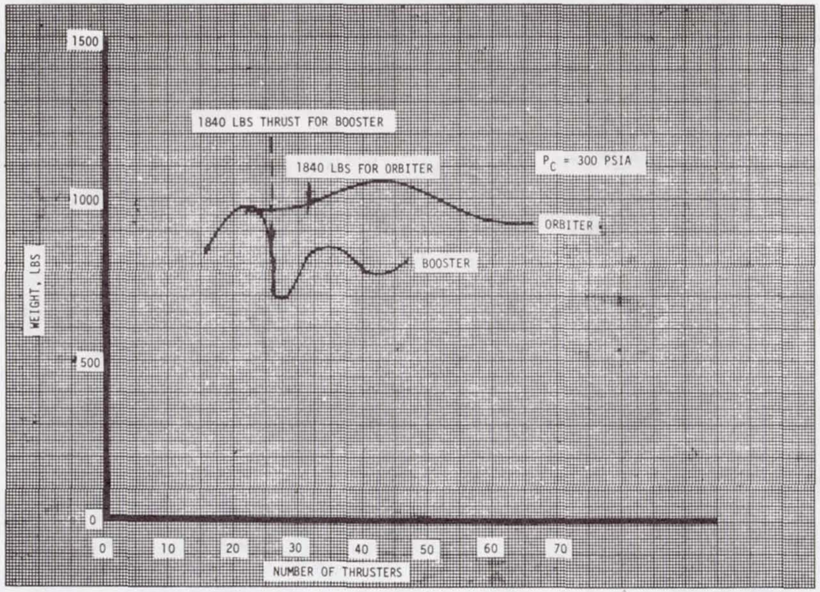


FIGURE 3

SELECTION OF COMMON THRUST LEVEL



Vehicle	Booster			Orbiter			
Maneuver	Yaw	Pitch	Roll	Yaw	Pitch	Roll	X Translation
Number of Thrusters Per Axis For Maneuvering in One Direction	8	4	2	8	4	4	4*
Thruster Flow Rates	34.5	17.25	8.67	34.5	17.25	17.25	17.25

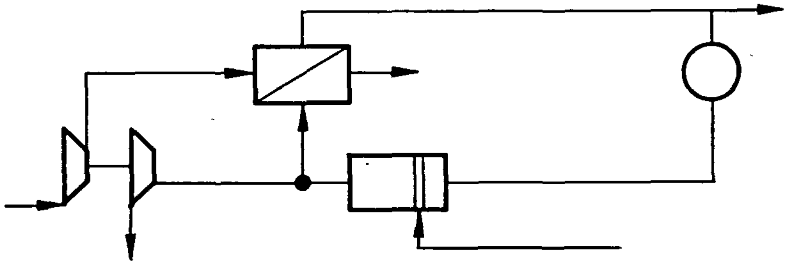
32 THRUSTERS ORBITER
26 THRUSTERS BOOSTER

FIGURE 4

FROM THE BASIC CYCLE SEVERAL CYCLES WERE ALSO EVALUATED, FEATURING DIFFERENT ARRANGEMENTS OF THE TURBINE/HEAT EXCHANGER ARRANGEMENT, AS SHOWN IN FIGURES 5 THROUGH 7, WHERE THE FEATURES OF EACH ARE INDICATED. THE THERMODYNAMICS OF THE CYCLES INDICATE ONLY MINOR PREFERENCE FROM ONE TO THE OTHER: THEREFORE SELECTIONS HAVE TO ULTIMATELY BE MADE ON OTHER BASES SUCH AS CONTROL AND WIDE FLOW RANGE CAPABILITY.

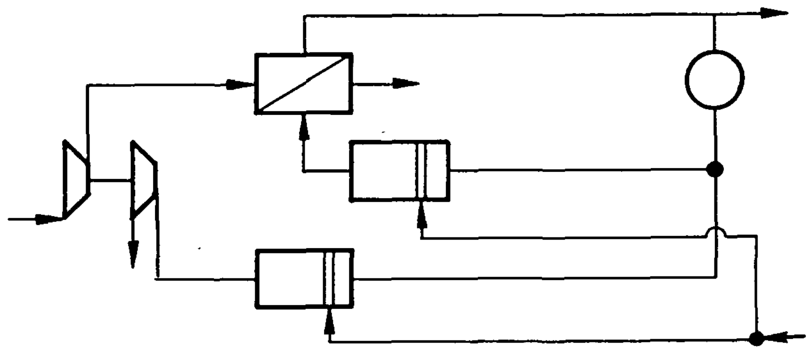
THE LOW PRESSURE GAS GENERATOR CYCLES ARE OF SOME INTEREST BECAUSE OF THEIR LOW CONDITIONING PENALTY. HOWEVER, THEY ARE SUBJECT TO CYCLE CONTROL AND HARDWARE COMPLEXITY DUE TO ASCENT TANK VAPOR DRAWOFF VARIABILITY AND THEY ARE EXPECTED TO BE VEHICLE INTEGRATION SENSITIVIE. ALTHOUGH THE WEIGHT TRADEOFFS WERE FAVORABLE THEY WERE NOT PURSUED BECAUSE OF THE CONTROLLABILITY QUESTION.

PARALLEL TURBINE - HEAT EXCHANGER ARRANGEMENTS



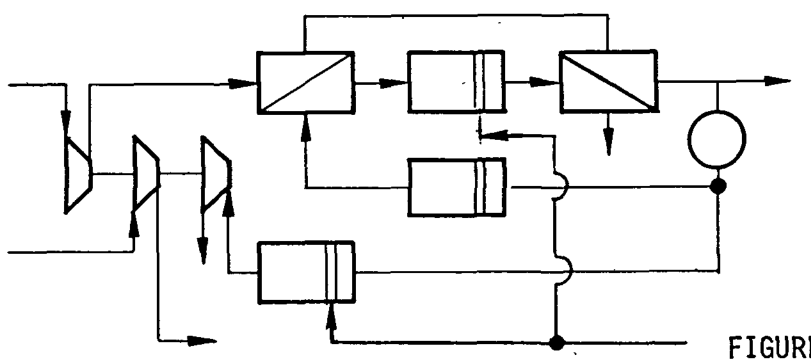
SINGLE GAS GENERATOR PER SIDE

- SIMPLE GAS GENERATOR CONTROL
- GAS GENERATOR OUTLET TEMPERATURE LIMIT 2000°R (O/F ~1)
- NARROW FLOW RANGE CAPABILITY



INDIVIDUAL GAS GENERATORS FOR TURBOPUMPS AND HEAT EXCHANGERS

- CAN SELECT OPTIMUM TURBINE AND HEAT EXCHANGER INLET TEMPERATURES (GROWTH POTENTIAL)
- WIDE FLOW RANGE CAPABILITY
- FINE OUTLET TEMPERATURE CONTROL POSSIBLE

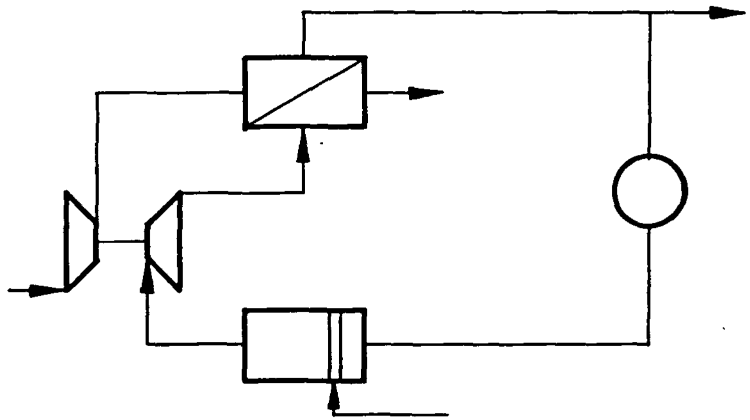


MULTIPLE STAGE COMBUSTION WITH COMBINED TURBOPUMP

- HIGH O/F OPERATION WITH MINIMUM NUMBER OF GAS GENERATORS (PARALLEL)
- CAN SELECT OPTIMUM TURBINE AND HEAT EXCHANGER INLET TEMPERATURES
- WIDE FLOW RANGE CAPABILITY
- LIMITED FLEXIBILITY FOR ONE-SIDE OPERATION

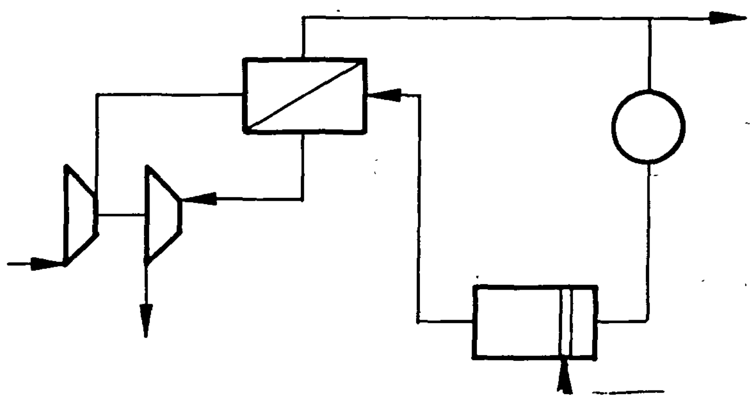
FIGURE 5

IN-LINE TURBINE - HEAT EXCHANGER ARRANGEMENTS



<u>SINGLE STAGE COMBUSTION WITH TURBINE UPSTREAM OF HEAT EXCHANGER</u>

- FAST STARTUP CAPABILITY
- GAS GENERATOR TEMPERATURE LIMIT 2260°R (HIGHER O/F THAN PARALLEL)
- TURBINE RUNS HOTTER THAN IN PARALLEL ARRANGEMENT (COOLING COMPLEXITY)
- LOWER HEAT EXCHANGER TEMPERATURE DIFFERENTIAL THAN PARALLEL ARRANGEMENT
- LOW PRESSURE HEAT EXCHANGER
- COMPLICATED OFF-DESIGN CONTROL

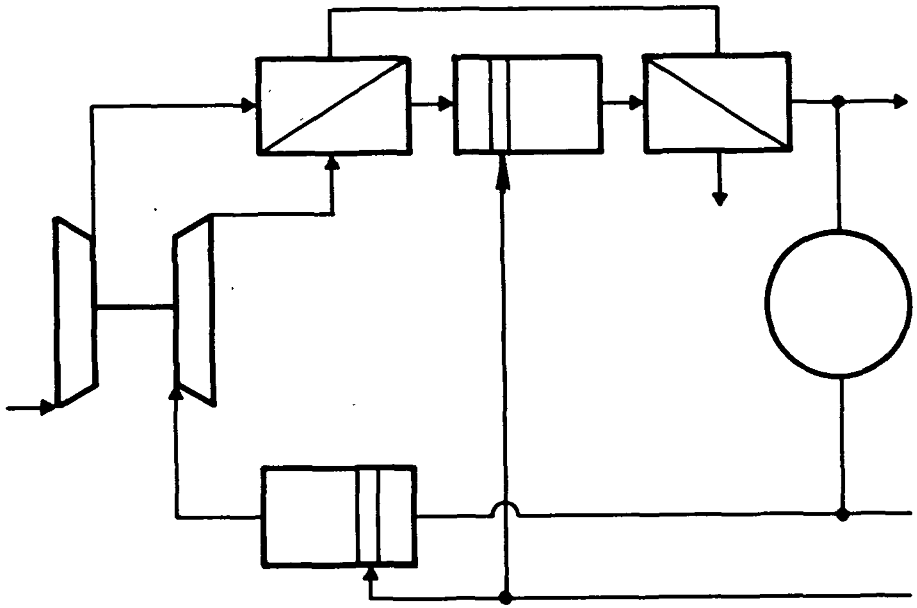


<u>SINGLE STAGE COMBUSTION WITH TURBINE DOWNSTREAM OF HEAT EXCHANGER</u>

- LOW TEMPERATURE TURBINE DESIGN POSSIBLE
- BETTER DYNAMIC COUPLING OF TURBINE AND HEAT EXCHANGER
- REDUCED START CAPABILITY
- POSSIBLE MOISTURE CARRYOVER INTO TURBINE - EROSION/CORROSION

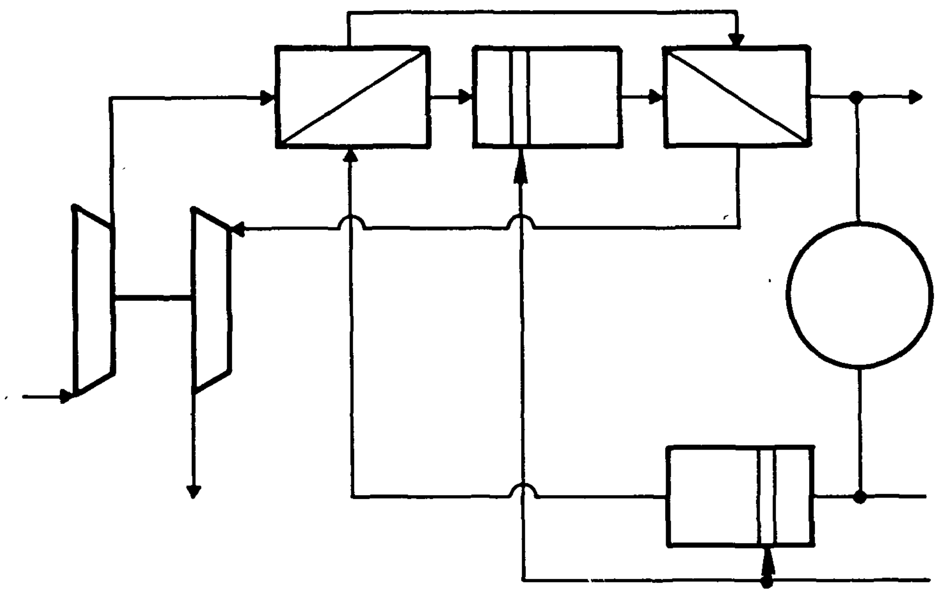
FIGURE 6

IN-LINE TURBINE - HEAT EXCHANGER ARRANGEMENTS



MULTIPLE STAGE COMBUSTION WITH UPSTREAM TURBINE

- FAST STARTUP CAPABILITY
- HIGH O/F POSSIBLE IN SECOND HEAT EXCHANGER
- LOW PRESSURE HEAT EXCHANGERS
- HIGH TEMPERATURE TURBINE
- OFF-DESIGN COMPLEXITY



MULTIPLE STAGE COMBUSTION WITH DOWNSTREAM TURBINE

- REDUCED STARTUP CAPABILITY
- HEAT EXCHANGER TEMPERATURE LIMIT 2000°R
- LOW TEMPERATURE TURBINE
- POSSIBLE MOISTURE CARRY-OVER PROBLEMS

FIGURE 7

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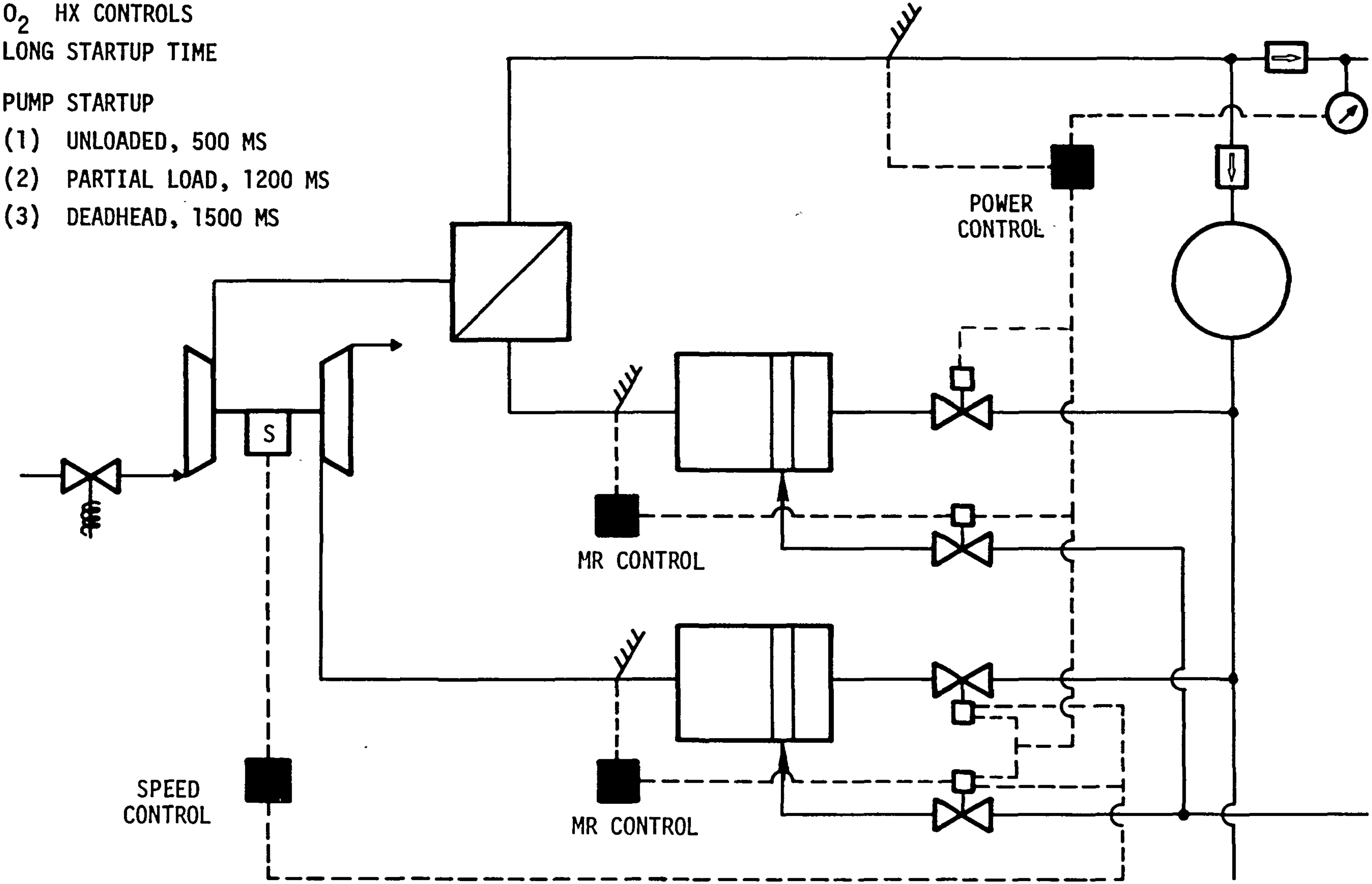
DUAL GAS GENERATOR CYCLE CONTROL SCHEMATIC

- O_2 HX CONTROLS
  LONG STARTUP TIME
- PUMP STARTUP
  (1) UNLOADED, 500 MS
  (2) PARTIAL LOAD, 1200 MS
  (3) DEADHEAD, 1500 MS



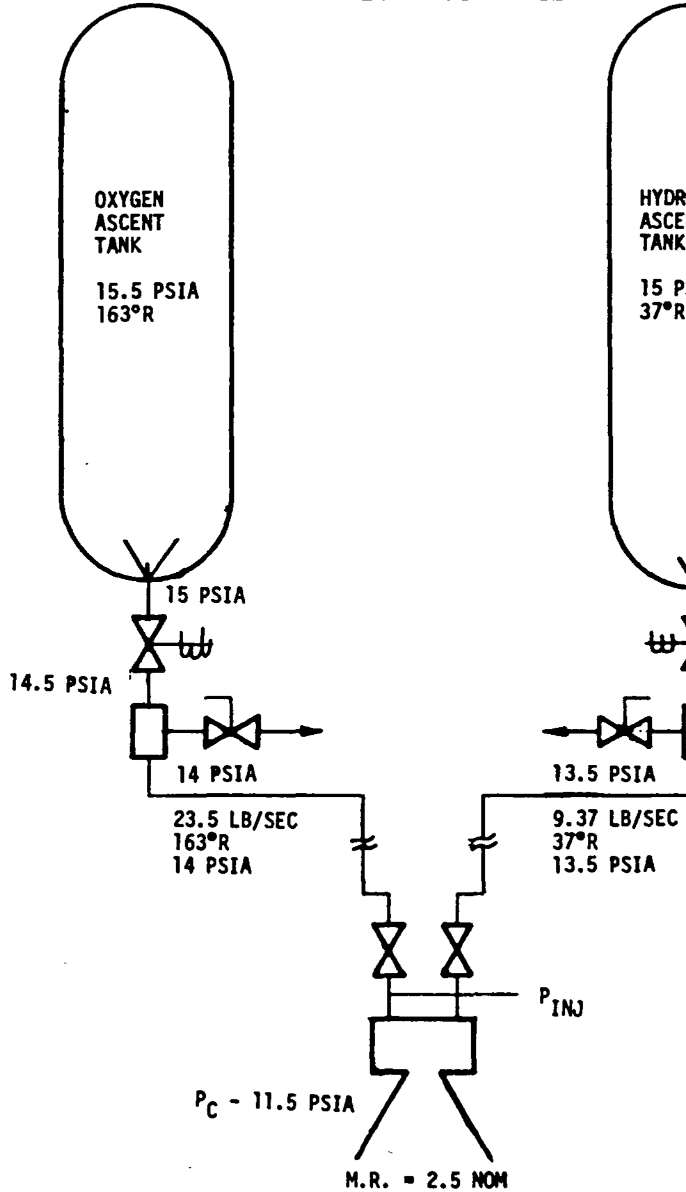
LOW PRESSURE

THE SELECTED CYCLES FOR THE SUBTASK B LOW PRESSURE APS EFFORT WERE:

- BLOWDOWN (BOOSTER)
- MIXING/CARBURETOR (ORBITER)
  ACTIVE CONDITIONING
  PASSIVE CONDITIONING

THE SELECTED BLOWDOWN CYCLE IS SHOWN IN FIGURE 8 ALONG WITH FINAL THERMODYNAMIC BALANCES. THE SELECTED ACTIVE ORBITER CYCLES ARE SHOWN IN FIGURES 9 AND 10.

BOOSTER LOW PRESSURE SCHEMATIC SHOWING THERMODYNAMIC BALANCES



- SIMPLE
- RELIABLE
- MINIMUM TECHNOLOGY

FIGURE 8.

ORBITER LOW PRESSURE SCHEMATIC SHOWING THERMODYNAMIC BALANCES

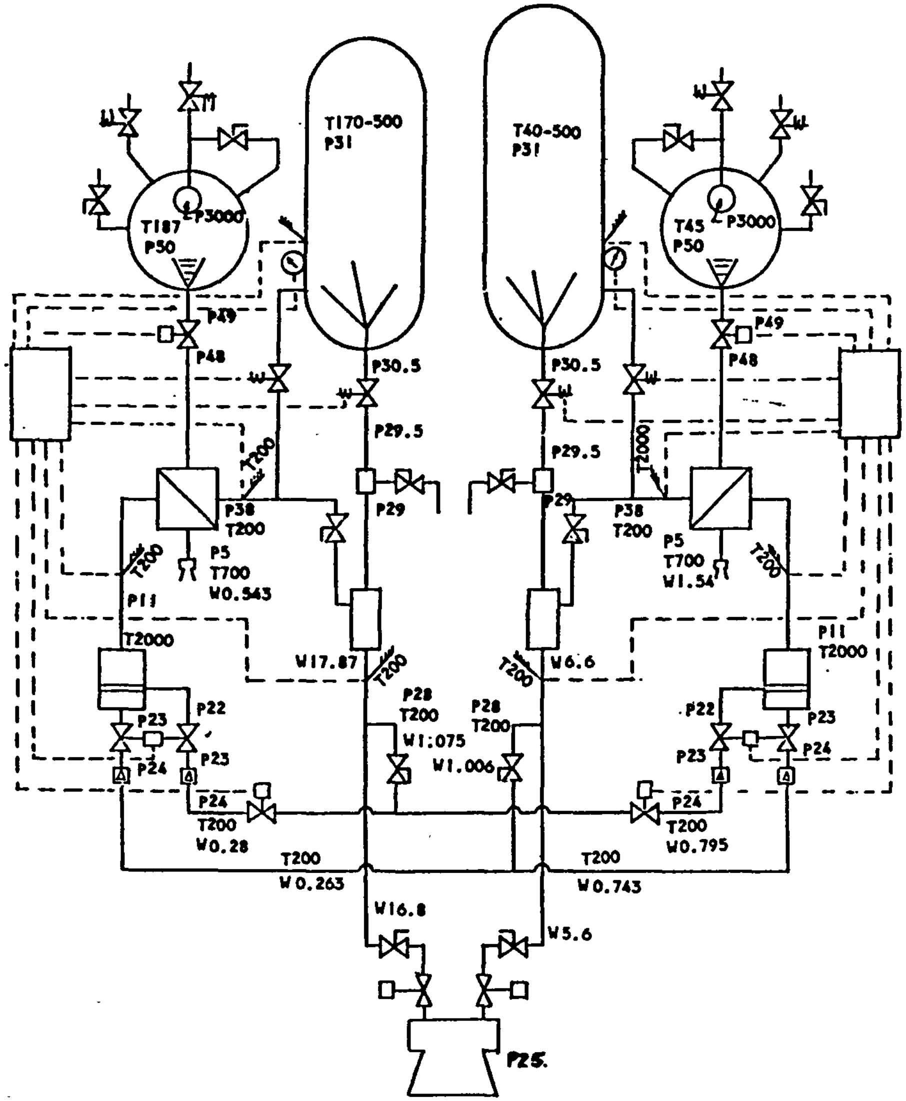
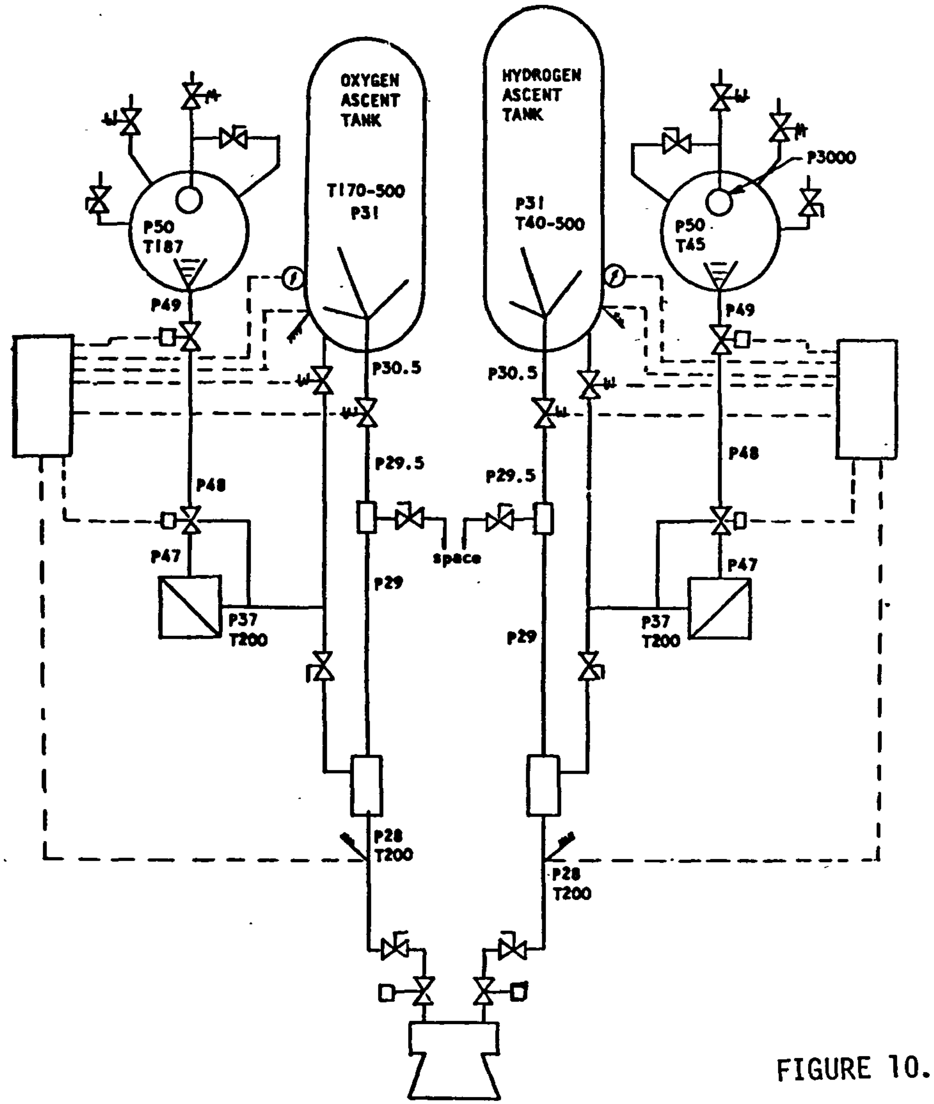


FIGURE 9.

ORBITER SCHEMATIC WITH PASSIVE CONDITIONING



- UTILIZES VEHICLE SENSIBLE HEAT WHERE POSSIBLE

FIGURE 10.

LOW PRESSURE ORBITER APS CONTROL

- DISTRIBUTION NETWORK FILLED WITH GASEOUS PROPELLANTS TO PROVIDE "ON-DEMAND" CAPABILITY.
- PROPELLANT ACQUISITION INITIATED BY OPENING ASCENT TANK SHUTOFF VALVES.
- MIXER HEATING/COOLING MODE DETERMINED BY ASCENT TANK PROPELLANT TEMPERATURE.
- PROPELLANT TEMPERATURE CONTROLLED BY MIXER INLET VALVE USING CLOSED TEMPERATURE LOOP.
- HEATING GAS TEMPERATURE CONTROLLED WITH GAS GENERATOR VALVES AND CLOSED TEMPERATURE LOOP.
- GAS GENERATOR EFFICIENCY MAINTAINED BY TRIMMING FUEL VALVE THROUGH CLOSED TEMPERATURE LOOP.
- ASCENT TANK PRESSURE MAINTAINED BY ON/OFF VALVE AND PRESSURE LOOP.
- THRUST PROVIDED BY ACTUATING THRUSTER VALVE AND IGNITER PACKAGE.

LOW PRESSURE BOOSTER CYCLE OPERATION

- PROPELLANT PLACED IN DISTRIBUTION NETWORK TO PROVIDE INITIAL DEMAND OPERATION
- MAIN VALVES OPEN
- SEPARATOR - VORTEX GENERATOR PROVIDE GASEOUS PROPELLANTS
- THRUSTERS OPERATED BY ACTUATING THRUSTER VALVES

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LOW PRESSURE SYSTEM UNKNOWN

BOOSTER

- PROPELLANT CONDENSATION IN DISTRIBUTION NETWORK
- IGNITION OF TWO-PHASE FLOW IN IGNITER
- ASCENT TANK EQUILIBRATION(S)
- NONE OF COMPONENTS HAVE BEEN DEVELOPED

ORBITER

- CONTROL OF MIXING CYCLE
- NONE OF COMPONENTS HAVE BEEN DEVELOPED
    - LIQUID ACQUISITION DEVICE REQUIRED ADDITIONAL TECHNOLOGY
    - MIXER REQUIRES VERIFICATION OF DESIGN CONCEPT
    - LIQUID VENT FOR SEPARATOR
- THERMAL CYCLING OF HEAT EXCHANGERS AND THRUSTERS AND MIXERS
- MECHANICAL CYCLING OF VALVES AND REGULATORS
- BREADBOARDING REQUIRED

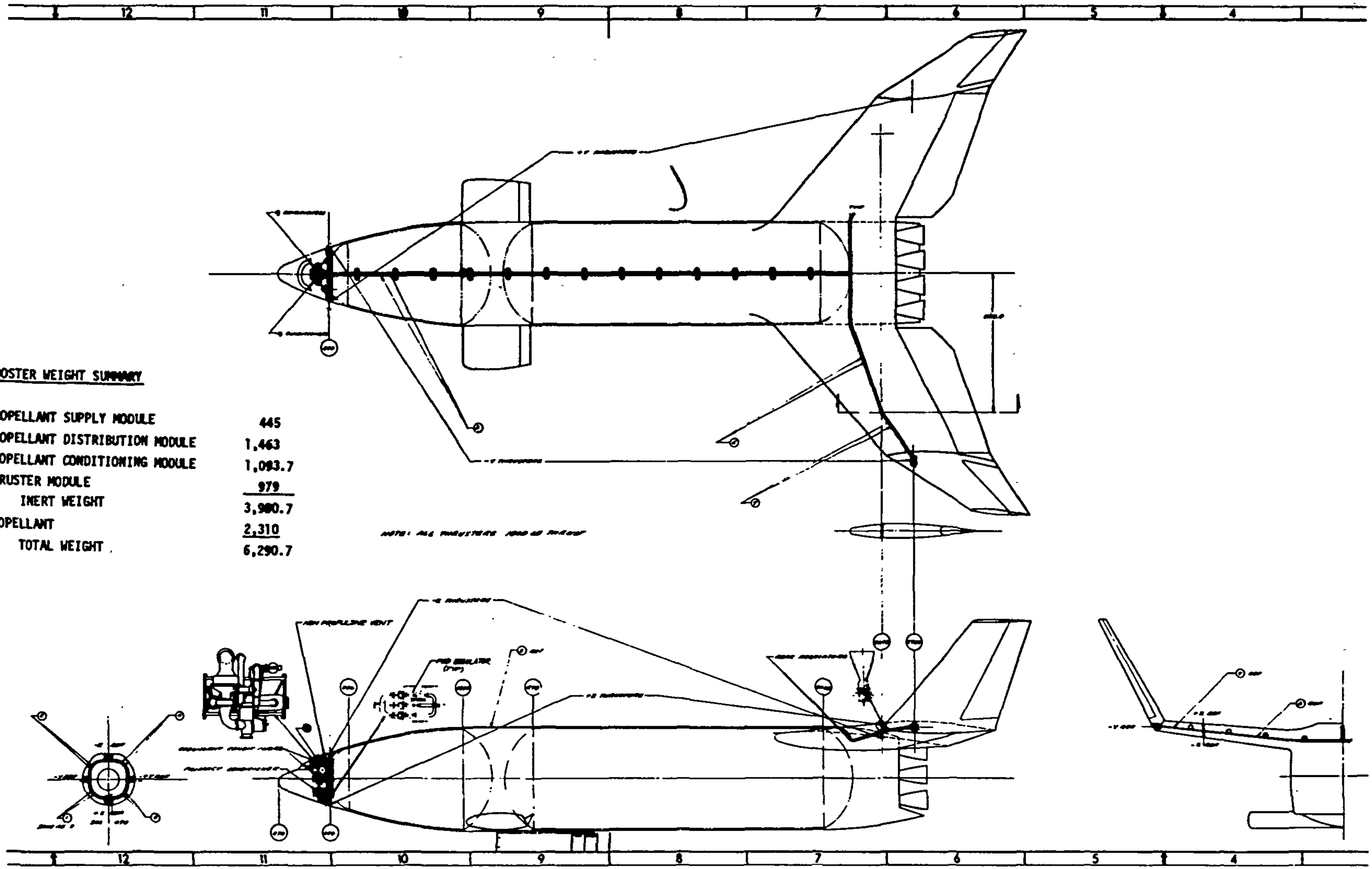
VEHICLE INSTALLATIONS

HIGH PRESSURE APS

THE APS INSTALLATION IN THE BOOSTER AND ORBITER ELEMENTS ARE SHOWN IN FIGURES 11 AND 12. IN BOTH SYSTEMS MAXIMUM ADVANTAGE IS TAKEN OF THE DISTRIBUTION LINES AS PART OF THE STORAGE VOLUMES. THE SUMMARY WEIGHTS ARE AS INDICATED TO BE COMPATIBLE WITH THE NOMINAL SSVDRD MISSION. FOR OTHER THAN THE NOMINAL MISSION THE ACCUMULATOR WEIGHTS IN FIGURE 13 SHOULD BE UTILIZED AS A FUNCTION OF APS CYCLES. A TYPICAL CONDITIONER MODULE IS INDICATED IN FIGURE 14.

LOW PRESSURE APS

THE LOW PRESSURE APS BOOSTER AND ORBITER INSTALLATIONS ARE SHOWN IN FIGURES 15 AND 16, ALONG WITH THEIR WEIGHTS.



BOOSTER WEIGHT SUMMARY

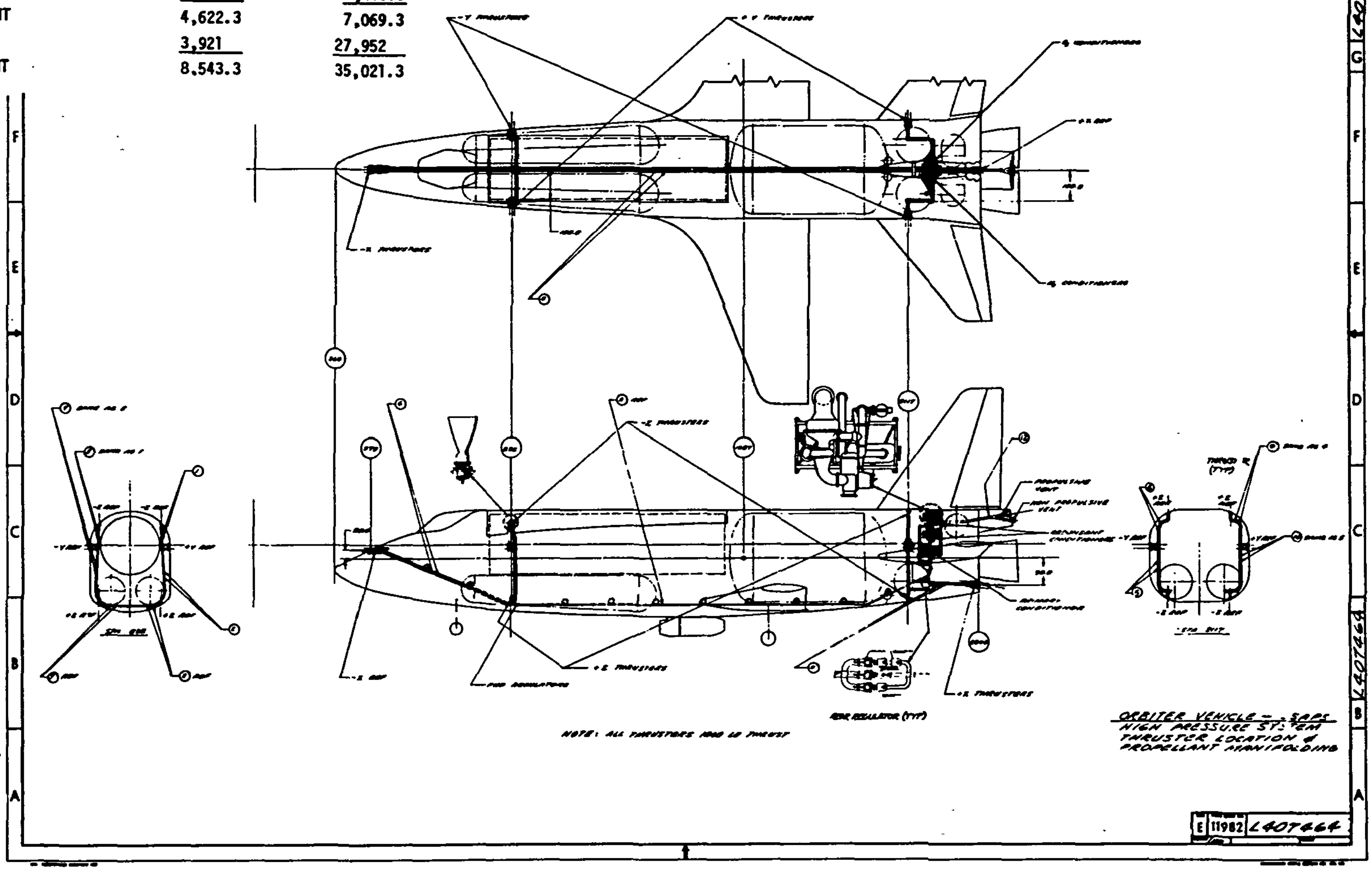
PROPELLANT SUPPLY MODULE	445
PROPELLANT DISTRIBUTION MODULE	1,463
PROPELLANT CONDITIONING MODULE	1,093.7
THRUSTER MODULE	979
INERT WEIGHT	3,980.7
PROPELLANT	2,310
TOTAL WEIGHT	6,290.7

BOOSTER PROPELLANT DISTRIBUTION AND COMPONENT LOCATION

FIGURE 11.

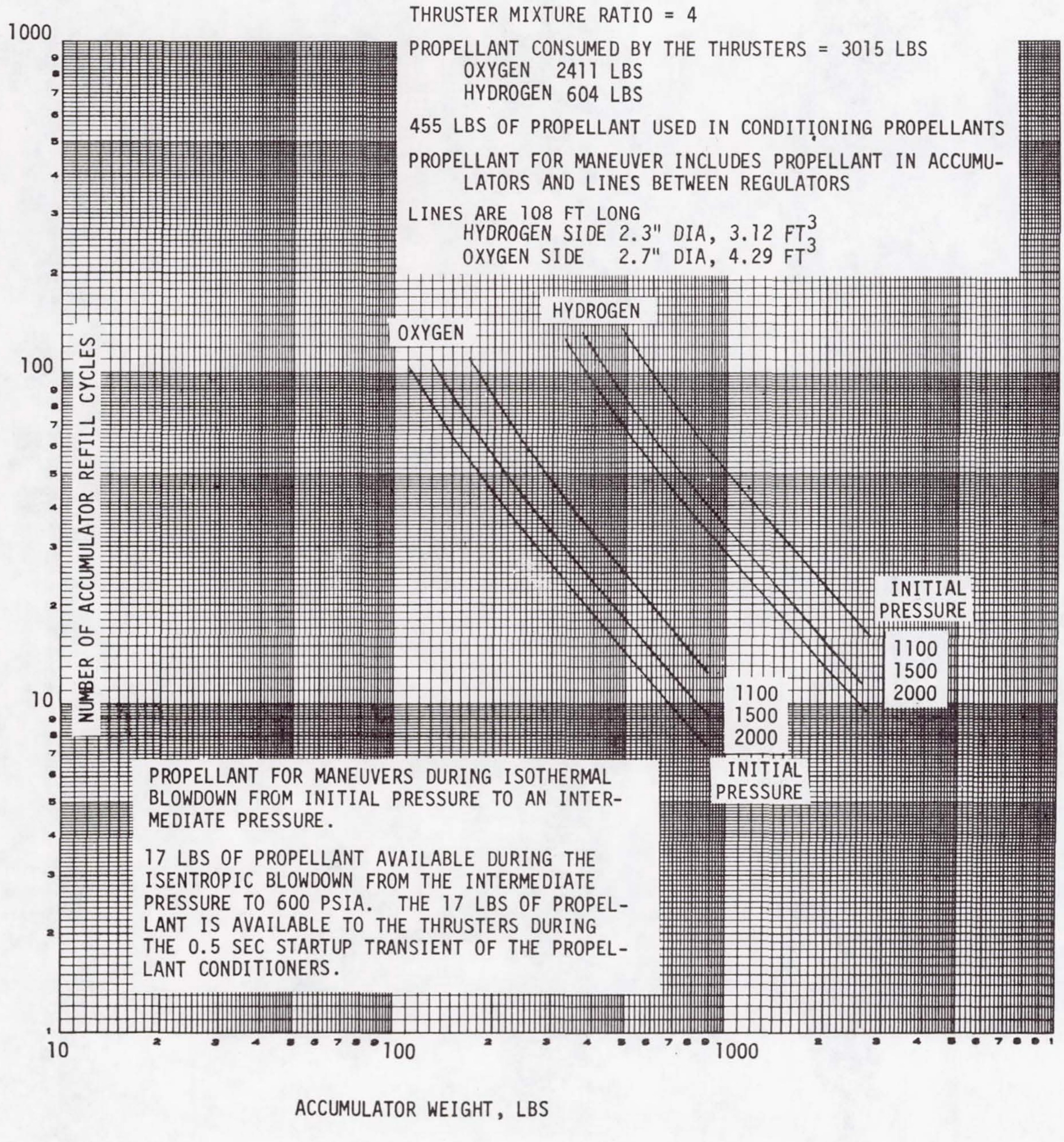
ORBITER WEIGHT SUMMARY

	LOW DELTA-V	ALL DELTA-V
PROPELLANT SUPPLY MODULE	675	3,122
PROPELLANT DISTRIBUTION MODULE	1,438	1,438
PROPELLANT CONDITIONING MODULE	1,093.7	1,093.7
THRUSTER MODULE	1,415.6	1,415.6
INERT WEIGHT	4,622.3	7,069.3
PROPELLANT	3,921	27,952
TOTAL WEIGHT	8,543.3	35,021.3



ORBITER PROPELLANT DISTRIBUTION AND COMPONENT LOCATION

FIGURE 12.



EFFECT ON ACCUMULATOR WEIGHT FOR ALL STABLE LIMIT CYCLE MISSION FOR MIXTURE RATIO = 4

FIGURE 13.

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ARRANGEMENT OF PROPELLANT CONDITIONER COMPONENTS

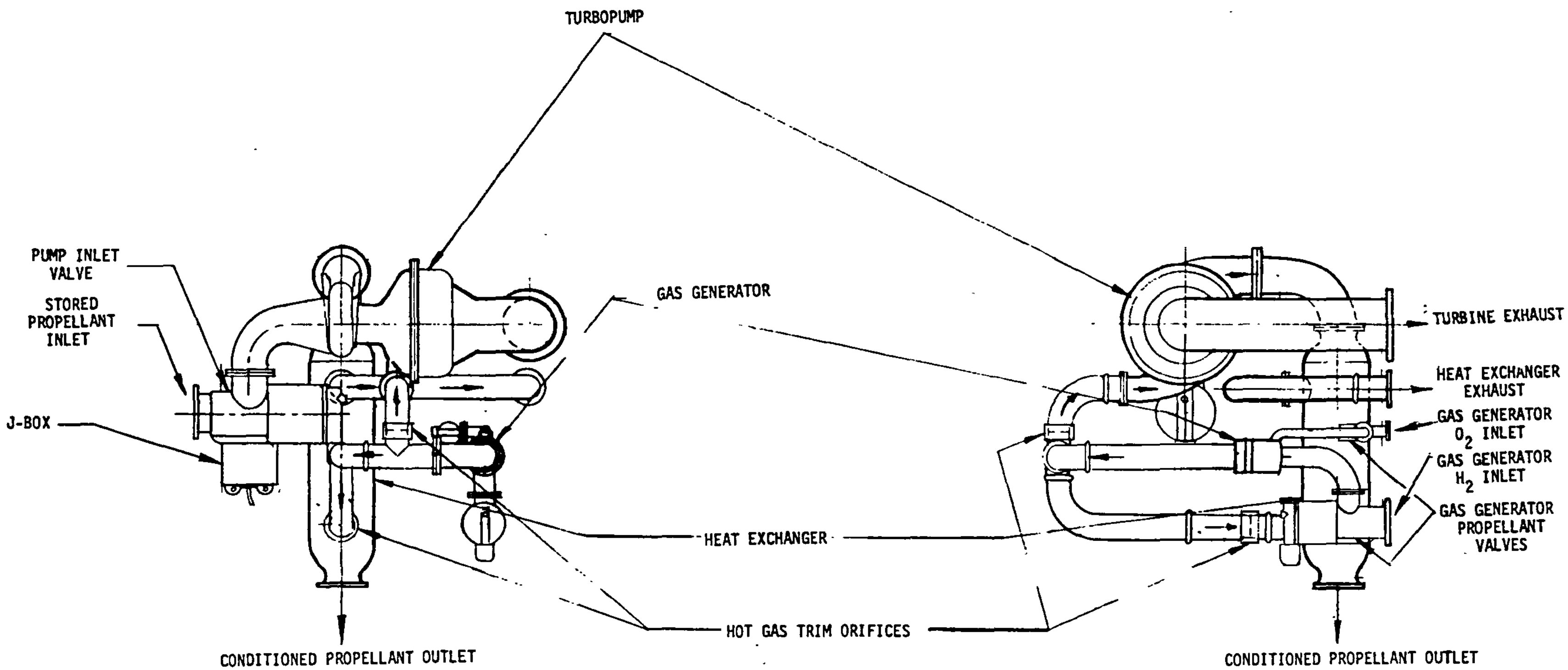


FIGURE 14.

Scale 1/10

LOW VELOCITY APS INERT WEIGHTS

HIGH PRESSURE

COMPONENTS	Quantity of			Unit Weight, lbs		Total Weight		
	Component in Module	Modules in APS	Component in APS	H_2 Side	O_2 Side	H_2 Side	O_2 Side	Both Sides
Propellant Supply Module								$\boxed{675}$
Propellant Acquisition Device	1	2	2	16	11	16	11	27
Propellant Tank	1	2	2	87	37	87	37	124
Pressurization System	1	2	2	68	37	68	21	89
Vent Valves	3	2	6	1.7	1.7	5	5	10
Insulation	1	2	2	93	100	93	100	193
Lines and Valves	--	--	--	--	--	17	17	34
Outer Structure Weight	--	--	--	--	--	72	126	198
Propellant Distribution Module								$\boxed{1438}$
Propellant Conditioner Isolation Valves	16	2	32	4	4	64	64	128
Propellant Conditioner Isolation Valves	8	1	8	4	4	16	16	32
Vent Isolation Valves	5	4	20	1.5	1.5	15	15	30
Regulator Isolation Valves	5	6	30	3	3	45	45	90
Accumulator Tank	1	2	2	260	68	260	68	328
Check Valves	9	2	18	2	2	18	18	36
Vent Valves	3	6	18	2	2	18	18	36
Regulators	3	4	12	20	20	120	120	240
Acc. Fill & Drain Valves	1	2	2	2	2	2	2	4
Lines, Hangers, Expansion Loops	--	--	--	469		469		469
Insulation	--	--	--	--	--	--	--	45
Propellant Conditioning Module								$\boxed{1093.7}$
Main Propellant Control Valve	2	3	6	14	20	42	60	102
Turbopumps	2	3	6	50	25	150	75	225
Heat Exchangers	2	3	6	32	85	96	255	351
Gas Generators for Heat Exchangers	2	3	6	7	6	21	18	39
Gas Generators for Turbopump	2	3	6	4.2	3.3	12.6	9.9	22.5
Gas Generator Propellant Control Valve for Heat Ex.	4	3	12	2.8	2.4	16.8	14.4	31.2
Gas Generator Propellant Control Valve for Turbopump	4	3	12	1.5	1.2	9	7.2	16.2
Start Accumulator Tank and Insulation	2	3	6	58	6	174	18	192
Vent Valve	2	3	6	2	2	6	6	12
Fill & Drain Valve	2	3	6	2	2	6	6	12
Regulators	2	3	6	10	10	30	30	60
Check Valves	2	3	6	1.8	1.8	5.4	5.4	10.8
Insulation	--	--	--	10	10	10	10	20
Thruster Module								$\boxed{1415.6}$
Thrusters	1	32	32	30.5		976		976
Propellant Control Valves	2	32	64					
Filters	2	32	64	2	2	64	64	128
Isolation Valves	4	32	128	2.1	2.3	134.4	147.2	281.6
Insulation	--	--	--	--	--	--	--	30
TOTAL APS								$\boxed{4622.1}$

FIGURE 15. BOOSTER LOW PRESSURE APS PACKAGING

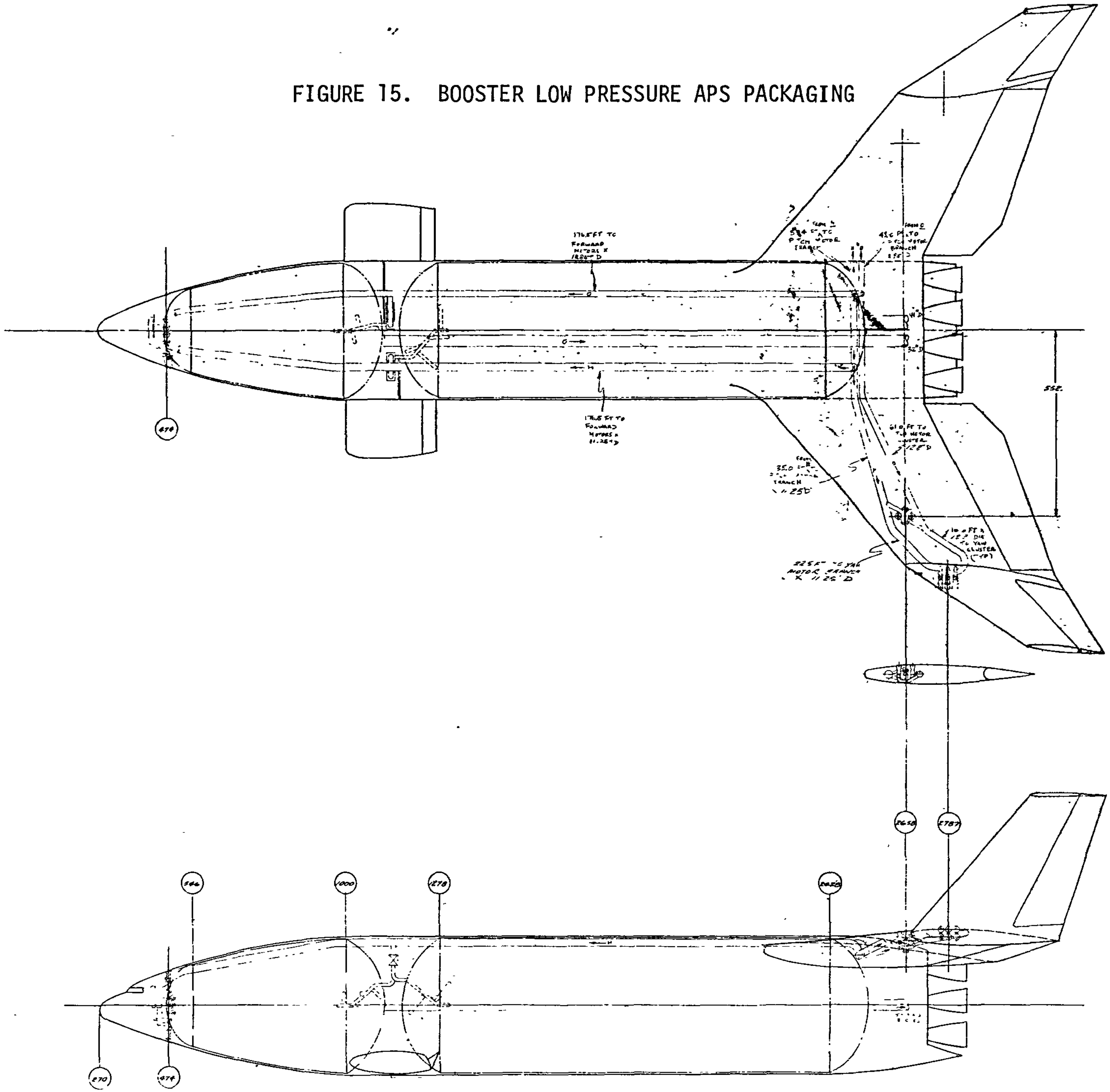
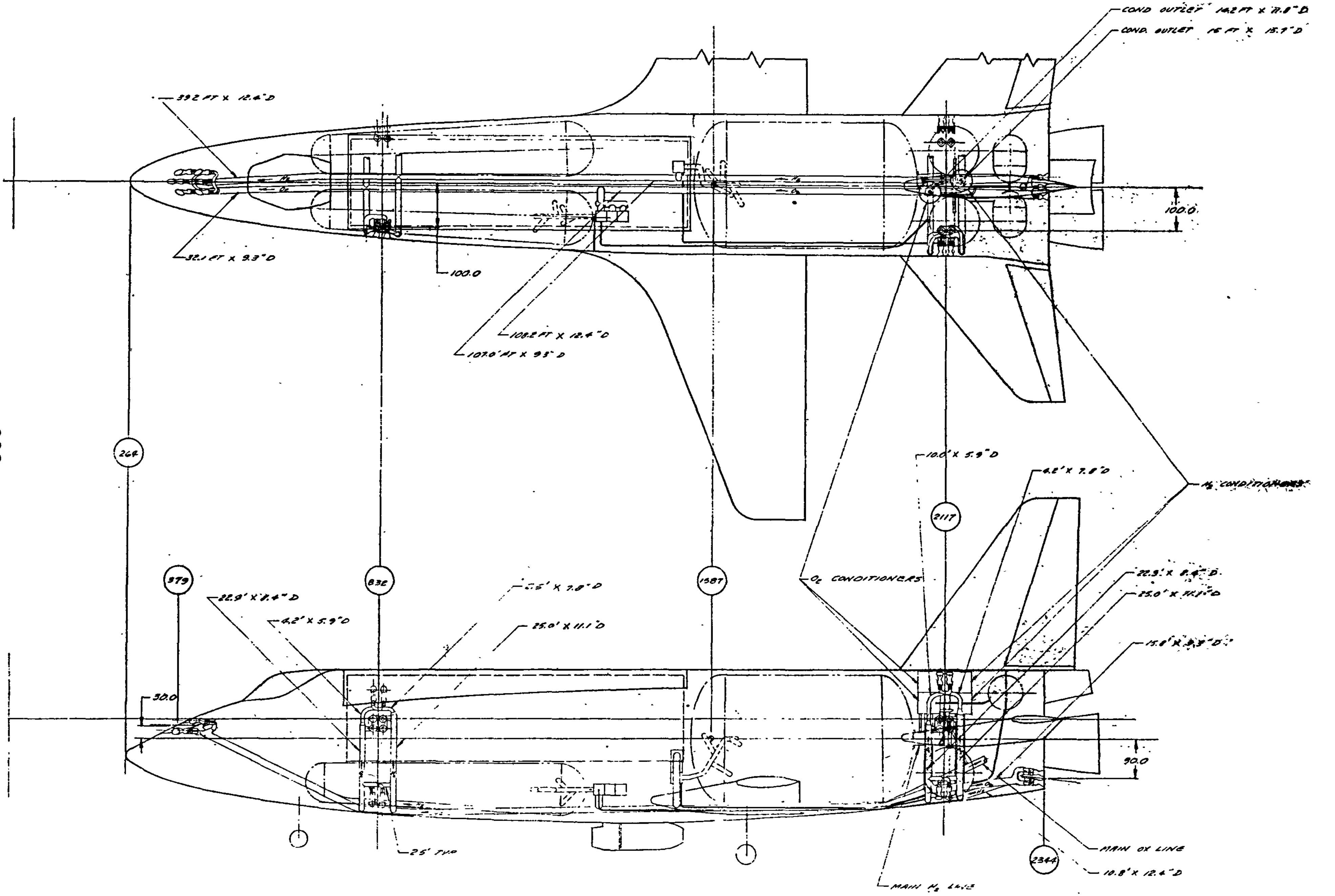


FIGURE 16. ORBITER LOW PRESSURE APS PACKAGING



ORBITER LOW PRESSURE APS WEIGHT SUMMARY

	Active APS	Passive APS
Conditioning Assembly	4,140	4664
Supplementary Propellant Assembly	9,273	8368
Propulsive Vent Assembly	40	----
Distribution Network	1306	1306
Thrusters	1716	1716
Thruster Linked Bipropellant Valves	572	572
Thruster Isolation Valves	915	915
Propellant in Distribution Network	28	28
Total APS Weight	17,990	17,569

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BOOSTER LOW PRESSURE APS WEIGHT SUMMARY

Oxygen Side

Temperature Range - 165 - 520°R
Maximum Pressure - 27 psia
Maximum Flowrate - 23.4 pps

Hydrogen Side

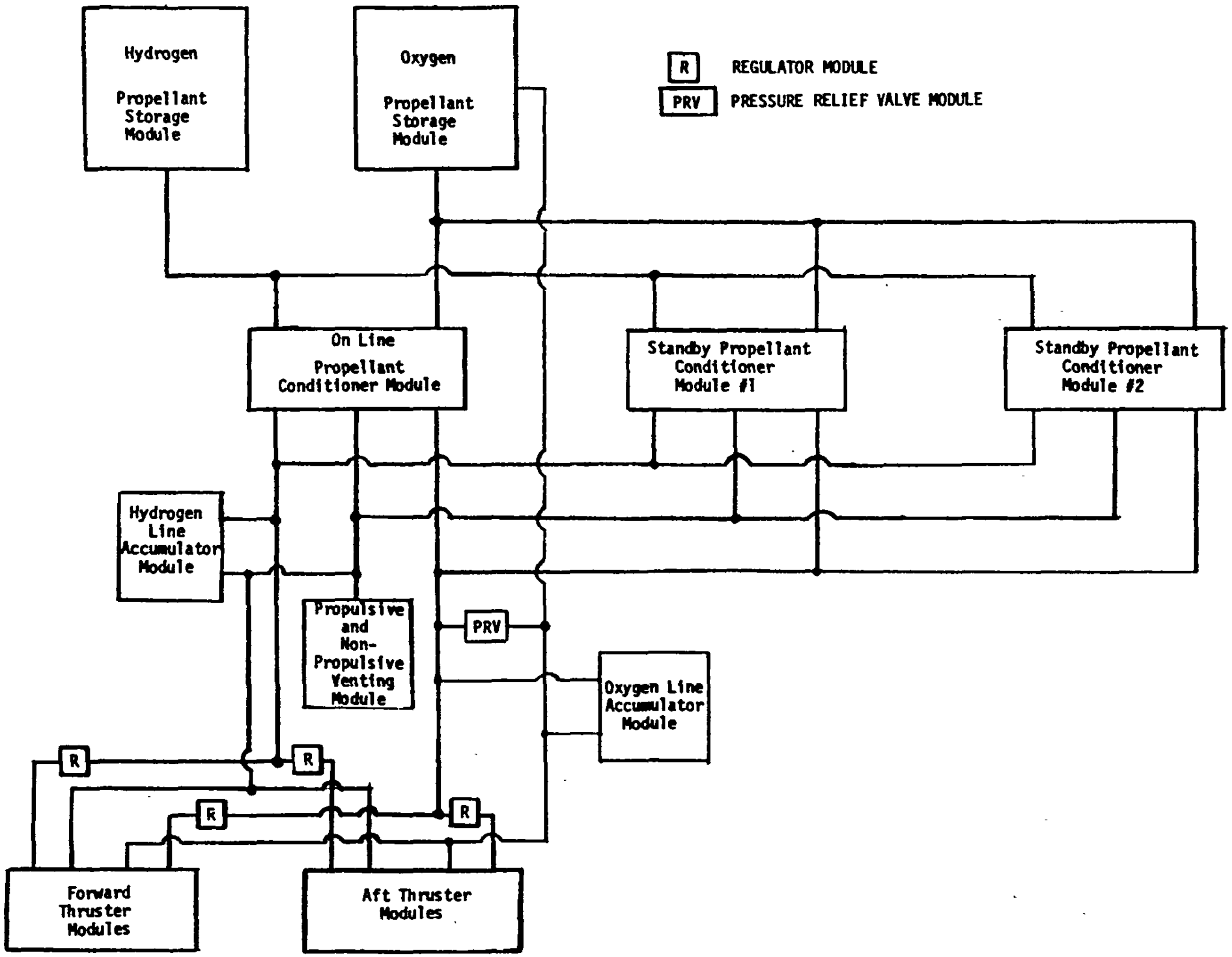
Temperature Range - 37 - 450°R
Maximum Pressure - 27 psia
Maximum Flowrate - 9.36 pps

		Quantity Required	Flowrate PPS	Total Weight LBS	Required Power Per Unit, Watts	Number of Cycles/Mission
Ascent Tank	O_2	3	11.75	61	7	1
Shutoff Valves	H_2	3	4.68	61	7	1
Liquid Vapor Separator	O_2	1	23.5	66		
Liquid Vapor Separator	H_2	1	23.5	54		
Vortex Generator	O_2	1	23.5	Included In Separator		
Vortex Generator	H_2	1	23.5			
Separator Regulator	O_2	3	0.9			≈10
Separator Regulator	H_2	3	0.9			≈10
Isolation Valves	O_2	6	0.9	36	96	
Isolation Valves	H_2	6	0.9	36		
Distribution Network	O_2	388 ft	23.5	983.0		
Distribution Network	H_2	388 ft	9.36	872.5		
Thruster Linked Bipropellant Valves	O_2	40	1.93	572	40	176 max.
	H_2		0.765			
Thrust Chambers, (P_c = 11.5, ϵ = 3)		40	2.73	3260	10	176 max.
Propellant in Lines				32.		
Thrust Chamber Isolation Valves	O_2	80	1.95	832	96	
	H_2	80	0.78			
TOTAL WEIGHT				6,865.5		

MAINTAINABILITY AND RELIABILITY

IN BOTH HIGH AND LOW PRESSURE EFFORTS THE APPROACH TO RELIABILITY AND MAINTAINABILITY WAS THROUGH THE USE OF FUNCTIONAL AND MODULAR REDUNDANCY. THE HIGH PRESSURE APS REDUNDANCY SYSTEM IS SCHEMATICALLY ILLUSTRATED IN FIGURE 17 AND LINE SCHEMATICALLY AS SHOWN IN FIGURE 18. AS SHOWN THE SYSTEM IS MODULARIZED INTO PROPELLANT CONDITIONER, REGULATOR AND THRUSTER MODULES. THE SYSTEM FEATURES EXTERNAL SWITCHING IN THE PROPELLANT CONDITIONER MODULES. THE OTHER MODULES USE ISOLATION VALVING AS APPROPRIATE TO THE INDIVIDUAL FUNCTION. CRITICAL ITEM MAINTENANCE IS SCHEDULED EVERY 3RD MISSION.

THE LOW PRESSURE ORBITER SYSTEM IS SHOWN SCHEMATICALLY IN FIGURE 19 AND 20, ILLUSTRATING A SIMILAR APPROACH. COMPLETE CRITICAL ITEM MAINTENANCE IS SCHEDULED EVERY 45TH MISSION FOR THE BOOSTER AND EVERY 10TH MISSION FOR THE ORBITER.



MAINTAINABILITY AND RELIABILITY

• MODULAR APPROACH

FIGURE 17.

HIGH PRESSURE APS SCHEMATIC

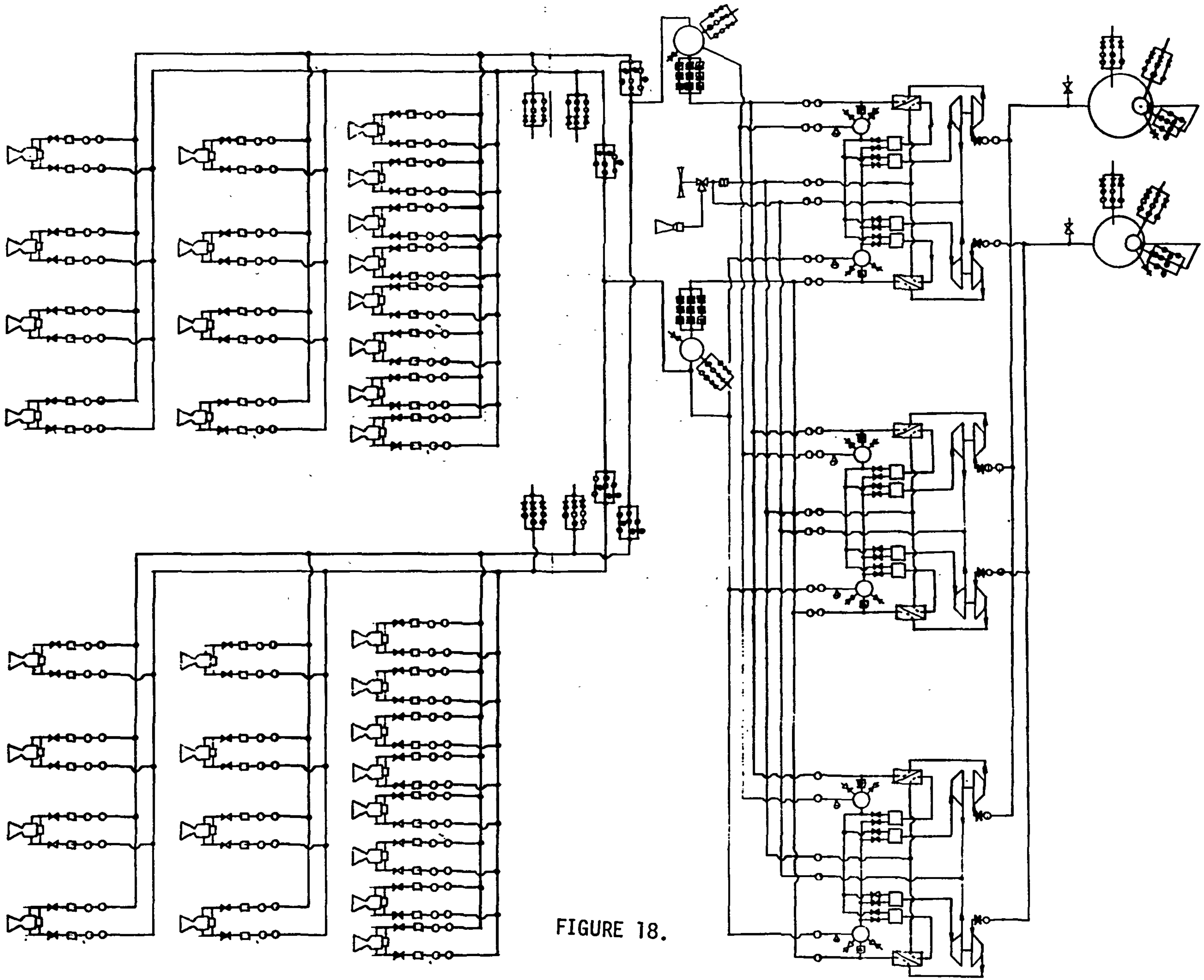


FIGURE 18.

LOW PRESSURE ORBITER REDUNDANT APS WITH PASSIVE CONDITIONER

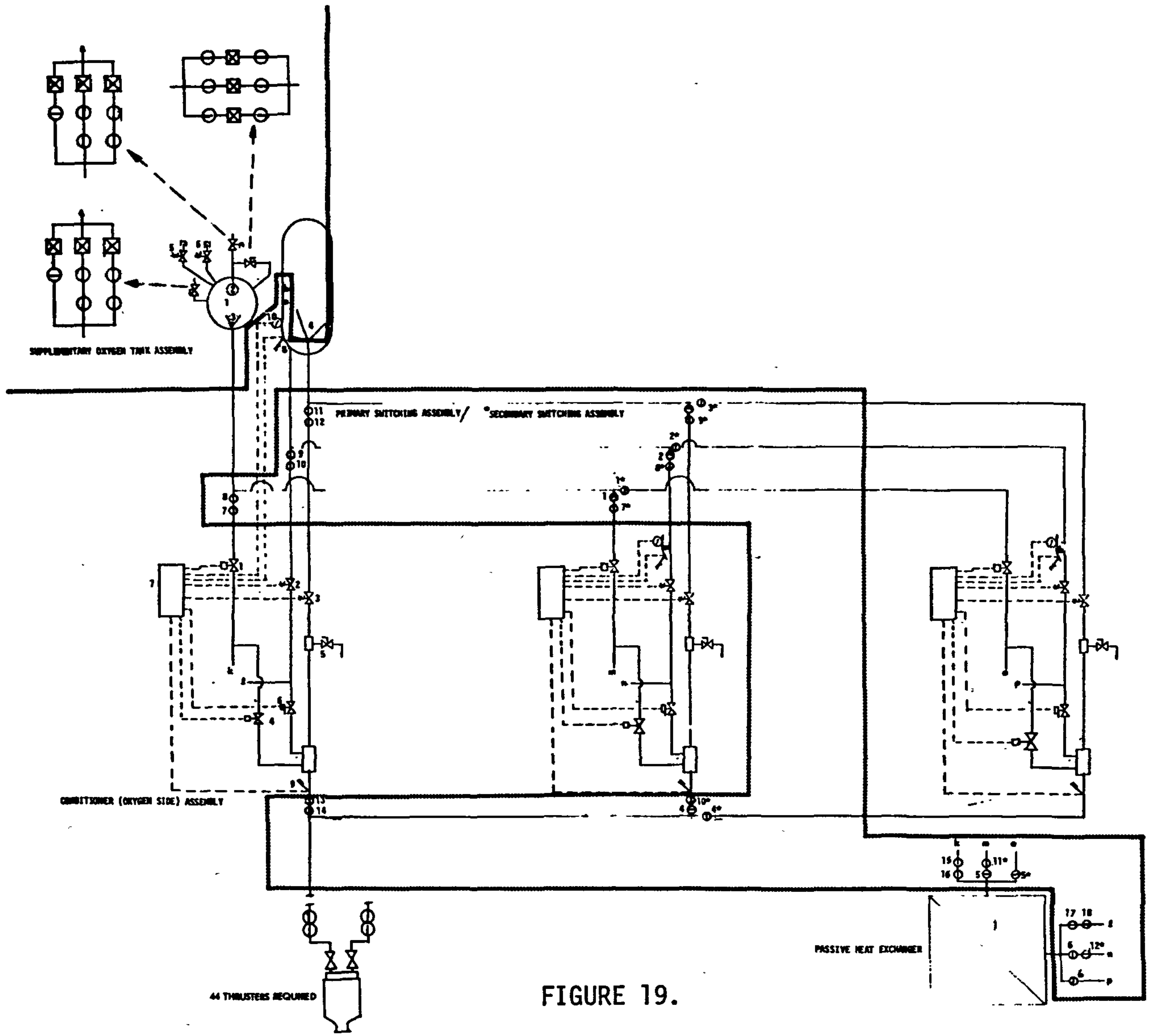


FIGURE 19.

LOW PRESSURE ORBITER REDUNDANT APS WITH ACTIVE CONDITIONING

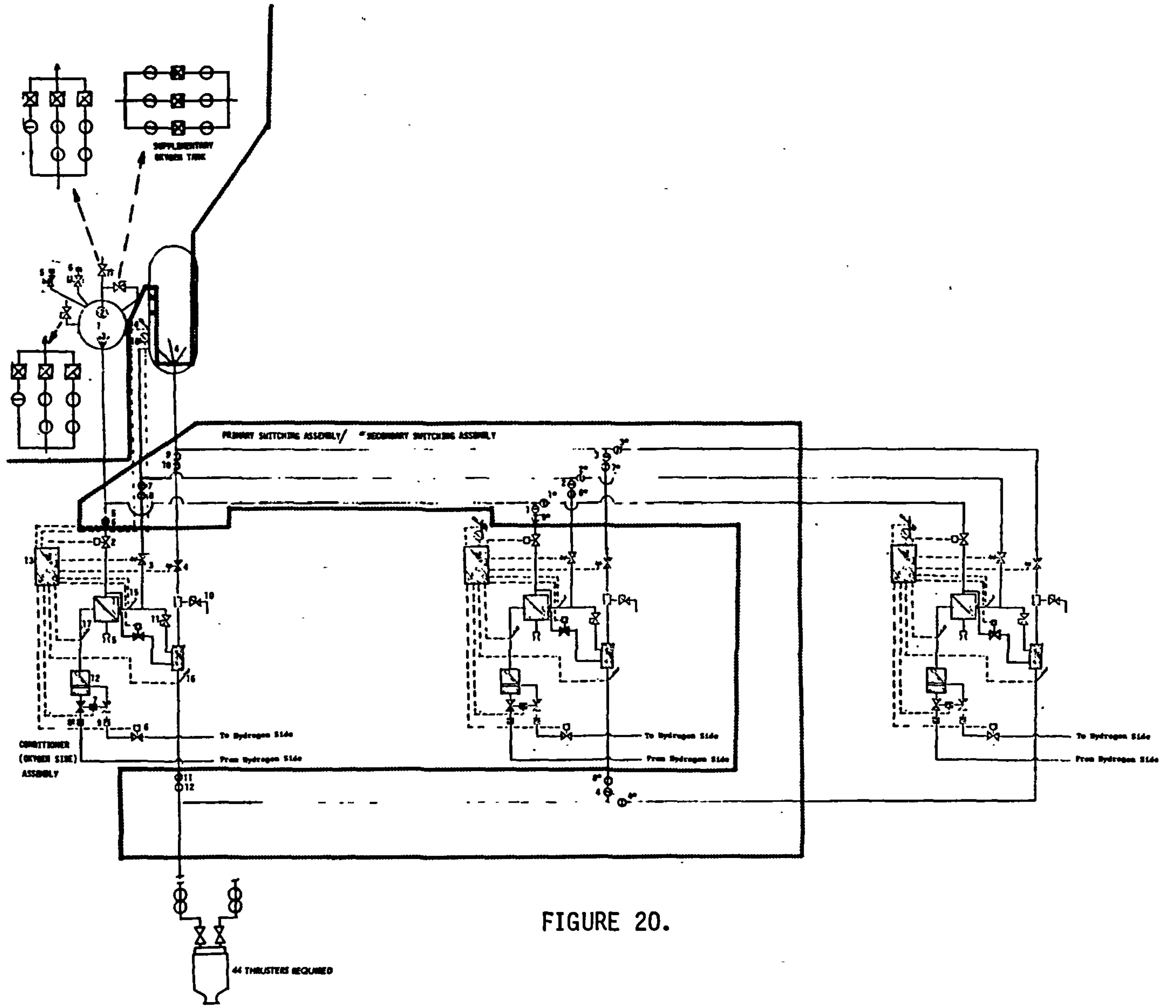


FIGURE 20.

SUMMARY CRITICAL TECHNOLOGY REQUIREMENTS

LOW PRESSURE

THE CRITICAL TECHNOLOGY AREAS FOR THE LOW PRESSURE APS ARE AS FOLLOWS:

- LIQUID GAS SEPARATORS - (LIQUID/GAS DISCRIMINATION, CYCLE LIFE, VENTING OF RESIDUAL LIQUIDS).
- THRUSTERS - (LONG LIFE DURABILITY, IGNITION OF LOW TEMPERATURE AND POSSIBLY DUAL PHASE PROPELLANTS, MR VARIABILITY, RE-ENTRY HEATING EFFECTS).
- GAS GENERATORS - (SAME AS HIGH PRESSURE SSAPS).
- PHASE ACQUISITION - (SIMILAR TO HIGH PRESSURE SSAPS REQUIREMENTS).
- HEAT EXCHANGERS - (RESPONSE TIMES, MR COMPATIBILITY CYCLE LIFE, STARTUP AND SHUTDOWN INTEGRATION, RECOVERY CYCLING AND OPERATION OF PASSIVE HEAT EXCHANGERS).
- REGULATORS AND VALVES - (SAME AS HIGH PRESSURE SSAPS).
- INSTRUMENTATION - (SAME AS HIGH PRESSURE SSAPS).
- DYNAMICS/CONTROLS - (SAME AS HIGH PRESSURE SSAPS).

SUMMARY CRITICAL TECHNOLOGY REQUIREMENTS

HIGH PRESSURE

THE CRITICAL TECHNOLOGY AREAS FOR THE HIGH PRESSURE APS ARE AS FOLLOWS:

- PHASE ACQUISITION - SURFACE TENSION SCREENS SELECTED (INERT GAS EFFECTS, COMPATIBILITY WITH PRESSURIZATION TECHNIQUE, NPSP CONTROL ACROSS SCREEN, INSPECTION AND CERITIFCATION).
- PRESSURIZATION - HELIUM AND AUTOGENOUS (INTEGRATION WITH ACQUISITION CONCEPT, HEAD LOSS DUE TO ENERGY INPUT, REQUIRED NPSH INTERNAL CONTROL).
- TANK INSULATION - (APPENDAGE HEAT LEAK CONTROL, MAXIMUM COORDINATED USE OF THERMODYNAMIC UNIT, COMPATIBILITY WITH COMPARTMENT PRESSURES, ASCENT/RE-ENTRY AND GROUND HOLD PUMPING REQUIREMENTS).
- GAS GENERATORS - (STARTUP/SHUTDOWN MR UNIFORMITY, ICING, FABRICATION AND MAINTAINABILITY).
- THRUSTERS - (SATISFACTORY PERFORMANCE TO BE EXPECTED IN CURRENT NASA CONTRACTS). (LONG LIFE DURABILITY, LIMIT CYCLE MR VARIABILITY, RE-ENTRY HEATING EFFECTS).
- TURBOPUMPS - (HIGH TRANSIENT EFFECTS ON BEARINGS AND SEALS, TOLERANCE EFFECTS IN TORQUE MATCHED SYSTEMS.
- HEAT EXCHANGERS - PARALLEL TUBE FLOW CONFIGURATION (ICING, MR COMPATIBILITY, STARTUP/ SHUTDOWN).
- REGULATORS AND THROTTLE VALVES - (PROPORTIONAL REGULATION FOR MAIN GAS FEED, THROTTLE TRIM VALVES FOR GAS GENERATORS).
- INSTRUMENTATION - (THERMAL SENSORS FOR CONTROL, FAULT ISOLATION INSTRUMENTATION).
- DYNAMICS/CONTROL - (PUMP STARTUP/SHUTDOWN, HEAT EXCHANGER STARTUP AND SHUTDOWN, PRESSURE SWITCHING NETWORKS, RAPID VERSUS LIMIT CYCLE DRAW-OFF.

CONCLUSIONS

HIGH PRESSURE

THE PUMPED LIQUID CYCLE IS ADAPTABLE TO BOTH BOOSTER AND ORBITER CONFIGURATIONS WITH A MAXIMUM OF COMMONALITY. BY TAKING A CONSERVATIVE BUT REASONABLY RESULTING EFFICIENT APPROACH TO THE MR SELECTION OF ~1:1 ALL CONDITIONER COMPONENTS CAN BE BUILT WITH TODAY'S TECHNOLOGY. THE BASIC CYCLE LENDS ITSELF TO HIGH EFFICIENCY CONTROLS AS WELL AS ABSOLUTE SIMPLICITY IN CONTROL. THE CYCLE WILL MEET HIGH DELTA-V AS WELL AS LOW DELTA-V MISSION DEMANDS AND PROVIDES GROWTH CAPABILITY INTO INTEGRATED APS, OMS, ETCS, ETC., SYSTEMS

LOW PRESSURE

THE BLOWDOWN CYCLE CONTINUES TO APPEAR TO BE ATTRACTIVE FOR THE BOOSTER APPLICATION WHERE MISSION FLEXIBILITY REQUIREMENTS WILL BE A MINIMUM. ITS SUCCESSFUL OPERATION IS CONTINGENT UPON REASONABLE ASCENT TANK THERMODYNAMIC CONDITIONS BEING ACHIEVED. IT IS NOT COMPATIBLE WITH COMMONALITY BETWEEN THE BOOSTER AND ORBITER.

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"AUXILIARY PROPULSION SYSTEM DEFINITION STUDY"

P. J. KELLY

McDONNELL DOUGLAS

TECHNICAL MANAGER

J. McCARTY

MARSHALL SPACE FLIGHT CENTER

N. CHAFFEE

MANNED SPACECRAFT CENTER

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SPACE SHUTTLE AUXILIARY PROPULSION SUBSYSTEM DEFINITION STUDY RESULTS

6-7 April 1971

HIGH PRESSURE APS STUDY – MSFC/MDAC–EAST – CONTRACT NAS8–26248

LOW PRESSURE APS STUDY – MSC/MDAC–EAST – CONTRACT NAS9–11012

P.J. KELLY, MCDONNELL DOUGLAS ASTRONAUTICS – EAST

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INTRODUCTION

THE NASA SPACE SHUTTLE VEHICLE SYSTEM FOR FUTURE MANNED SPACE OPERATIONS REQUIRES DEVELOPMENT OF A NUMBER OF SUBSYSTEMS WHICH ARE EITHER NEW OR SIGNIFICANT EXTENSIONS OF STATE-OF-THE-ART TECHNOLOGY. AMONG THESE IS THE AUXILIARY PROPULSION SUBSYSTEM (APS) USED FOR CONTROL AND MANEUVERING OF THE SHUTTLE VEHICLE AFTER MAIN ENGINE CUT-OFF. TO PROVIDE A HIGH PERFORMANCE APS AND, AT THE SAME TIME, TO TAKE ADVANTAGE OF BENEFITS IN THE AREAS OF PROPELLANT LOGISTICS, SAFETY, REUSE, AND PERFORMANCE, A GASEOUS HYDROGEN/OXYGEN AUXILIARY PROPULSION SUBSYSTEM WAS IDENTIFIED AS THE MOST DESIRABLE TYPE OF SUBSYSTEM.

THESE ARE TWO BASIC METHODS OF IMPLEMENTING AN APS OF THIS TYPE. THESE ARE:

(1) HIGH PRESSURE APS, IN WHICH PROPELLANTS ARE STORED AT, OR CONDITIONED TO, THE MOST DESIRABLE THRUSTER OPERATING PRESSURES;

(2) LOW PRESSURE APS, IN WHICH PROPELLANTS ARE SUPPLIED TO THE CONTROL THRUSTERS FROM MAIN ASCENT PROPELLANT TANKS AT NORMAL ULLAGE PRESSURES.

TO FULFILL SHUTTLE NEEDS, NASA CONTRACTED FOR APS DEFINITION STUDIES OF BOTH HIGH AND LOW PRESSURE APS. THE MCDONNELL DOUGLAS ASTRONAUTICS COMPANY-EAST CONDUCTED A HIGH PRESSURE APS STUDY UNDER CONTRACT NO. NAS8-26248 AND A LOW PRESSURE APS STUDY UNDER CONTRACT NO. NAS9-11012. NASA TECHNICAL DIRECTION OF THIS EFFORT WAS PROVIDED BY THE MARSHALL SPACE FLIGHT CENTER ON THE HIGH PRESSURE APS STUDY AND BY THE MANNED SPACECRAFT CENTER ON THE LOW PRESSURE APS STUDY. THE AEROJET LIQUID ROCKET COMPANY, UNDER SUB-CONTRACT TO MDAC-EAST, PROVIDED THE ANALYSES AND DESIGN SUPPORT NECESSARY TO DEFINE THE ACTIVE COMPONENTS FOR APS EVALUATION.

PRESENTED IN THIS PAPER ARE SUMMARY DESCRIPTIONS OF PRELIMINARY DESIGNS BOTH HIGH AND LOW PRESSURE APS AS DETERMINED FROM THE TWO STUDY PROGRAMS.

ORBITER-HIGH PRESSURE APS SUMMARY

THE HIGH PRESSURE APS FOR THE SPACE SHUTTLE ORBITER IS DESIGNED FOR ALL ORBITER ATTITUDE CONTROL AND MANEUVERING FUNCTIONS AND THUS NO SEPARATE ORBIT MANEUVERING SUBSYSTEM IS REQUIRED. THE TOTAL IMPULSE CAPABILITY OF THE SUBSYSTEM IS APPROXIMATELY 12,700,000 LB-SEC FOR BOTH HIGH (ORBITER C) AND LOW (ORBITER B) CROSS RANGE DESIGNS.

A TURBOPUMP APS CONCEPT WAS SELECTED AS THE PREFERRED HIGH PRESSURE APS APPROACH. THE ACCOMPANYING FIGURE PRESENTS A SIMPLIFIED SCHEMATIC OF THE HYDROGEN SIDE OF THE APS. THE SCHEMATIC FOR THE OXYGEN SIDE IS SIMILAR. THE THRUSTER ASSEMBLIES WHICH OPERATE AT REGULATED INLET PRESSURE ARE DECOUPLED FROM THE CONDITIONING ASSEMBLIES BY ACCUMULATORS WHICH PROVIDE SUFFICIENT GAS STORAGE CAPABILITY FOR A NUMBER OF THRUSTER FIRINGS.

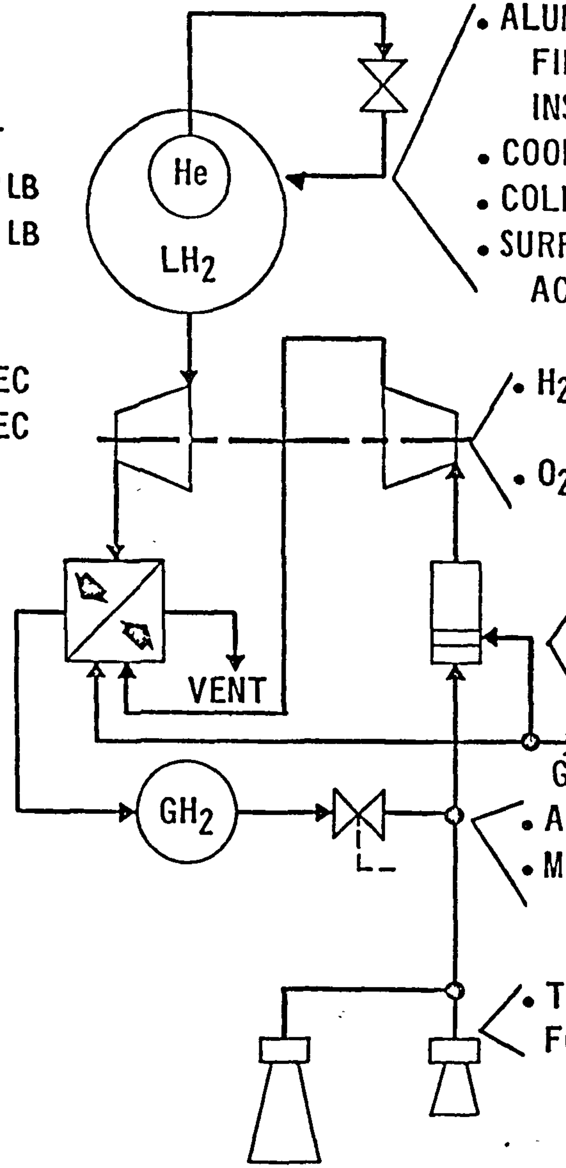
THE CONDITIONER ASSEMBLIES ARE SIZED TO PROVIDE A FLOW RATE WHICH IS EQUIVALENT TO THE FLOW RATE OF THE MAXIMUM NUMBER OF THRUSTERS THAT CAN OPERATE AT ANY GIVEN TIME. THESE ASSEMBLIES CONSIST OF A SINGLE 2000°R BIPROPELLANT GAS GENERATOR, A CONVENTIONAL TURBOPUMP, AND A REBURN HEAT EXCHANGER. ALL GAS GENERATOR PRODUCTS ARE FIRST PASSED THROUGH THE TURBOPUMP, WHERE THE ENERGY TO RAISE THE REQUIRED LIQUID PROPELLANT FLOW RATE TO THE REQUIRED PRESSURE IS EXTRACTED. THE FUEL-RICH GASSES ARE THEN DIRECTED TO THE HEAT EXCHANGER WHERE SUPPLEMENTAL OXYGEN IS ADDED TO PROVIDE THE ENERGY REQUIRED TO CONVERT THE LIQUID PROPELLANT TO GAS AT THE DESIRED TEMPERATURE. THE EXHAUST PRODUCTS ARE DISCHARGED FROM THE VEHICLE THROUGH OPPOSING NOZZLES TO ELIMINATE DISTURBANCE FORCES OR, IF A +X AXIS MANEUVER IS IN PROCESS, THROUGH AN AFT DIRECTED CONVERGENT-DIVERGENT NOZZLE TO PROVIDE USEFUL +X IMPULSE.

THE PROPELLANT TANKAGE ASSEMBLY OPERATES SIMILARLY TO CONVENTIONAL STORABLE PROPELLANT TANKAGE. PRESSURE WITHIN THE TANK IS MAINTAINED BY MECHANICAL REGULATION OF THE HELIUM PRESSURE SUPPLY. PROPELLANTS ARE MAINTAINED IN A LIQUID STATE BY A COMBINATION OF HIGH PERFORMANCE INSULATION AND PROPELLANT VAPORIZATION. NORMAL ON-ORBIT HEATING IS ABSORBED BY A COOLANT LOOP IN WHICH PROPELLANT IS EXTRACTED FROM THE TANK, AND PASSED OVER THE OUTER SHELL, WHERE THE HEAT LEAK IS TAKEN UP BY THE PROPELLANT HEAT OF VAPORIZATION. PROPELLANTS ARE MAINTAINED AT THE TANK OUTLET BY A SURFACE TENSION SCREEN DEVICE. THIS DEVICE PROVIDES POSITIVE PROPELLANT POSITIONING IN ZERO G AND DURING MULTIAXIS, LOW-G OPERATION.

HIGH PRESSURE ORBITER APS SUMMARY

TOTAL SUBSYSTEM WEIGHT
- ORBITER B – 35,879 LB
- ORBITER C – 37,070 LB

SPECIFIC IMPULSE
- ATTITUDE – 416.0 SEC
- TRANSLATION – 423.7 SEC



- ALUMINUM TANK WITH HPI INSULATION, FIBERGLASS COVER (POLYURETHANE INSULATION H_2 ONLY)
- COOLING – VAPOR VENT USING THROTTLED H_2 LIQUID
- COLD GAS HELIUM PRESSURIZATION
- SURFACE TENSION SCREEN CHANNELS FOR PROPELLANT ACQUISTION

- H_2 – 2 STAGE CENTRIFUGAL PUMP AND 3 STAGE AXIAL TURBINE
- O_2 – 1 STAGE CENTRIFUGAL PUMP AND 2 STAGE AXIAL TURBINE

- REBURN COUNTERFLOW HEAT EXCHANGER (H_2 – 3750°R, O_2 – 4150°R) PROPULSIVE VENT
- 2000°R GAS GENERATOR LINKED VALVES WITH O_2 TEMPERATURE FEEDBACK CONTROL, H_2 ACCUMULATOR PRESSURE FEEDBACK CONTROL

- ALUMINUM ACCUMULATOR (HPI INSULATED)
- MECHANICAL PRESSURE REGULATOR

- THRUSTER ε = 60 FOR ATTITUDE CONTROL ε = 120 FOR +X TRANSLATION, REGENERATIVE/FILM COOLED

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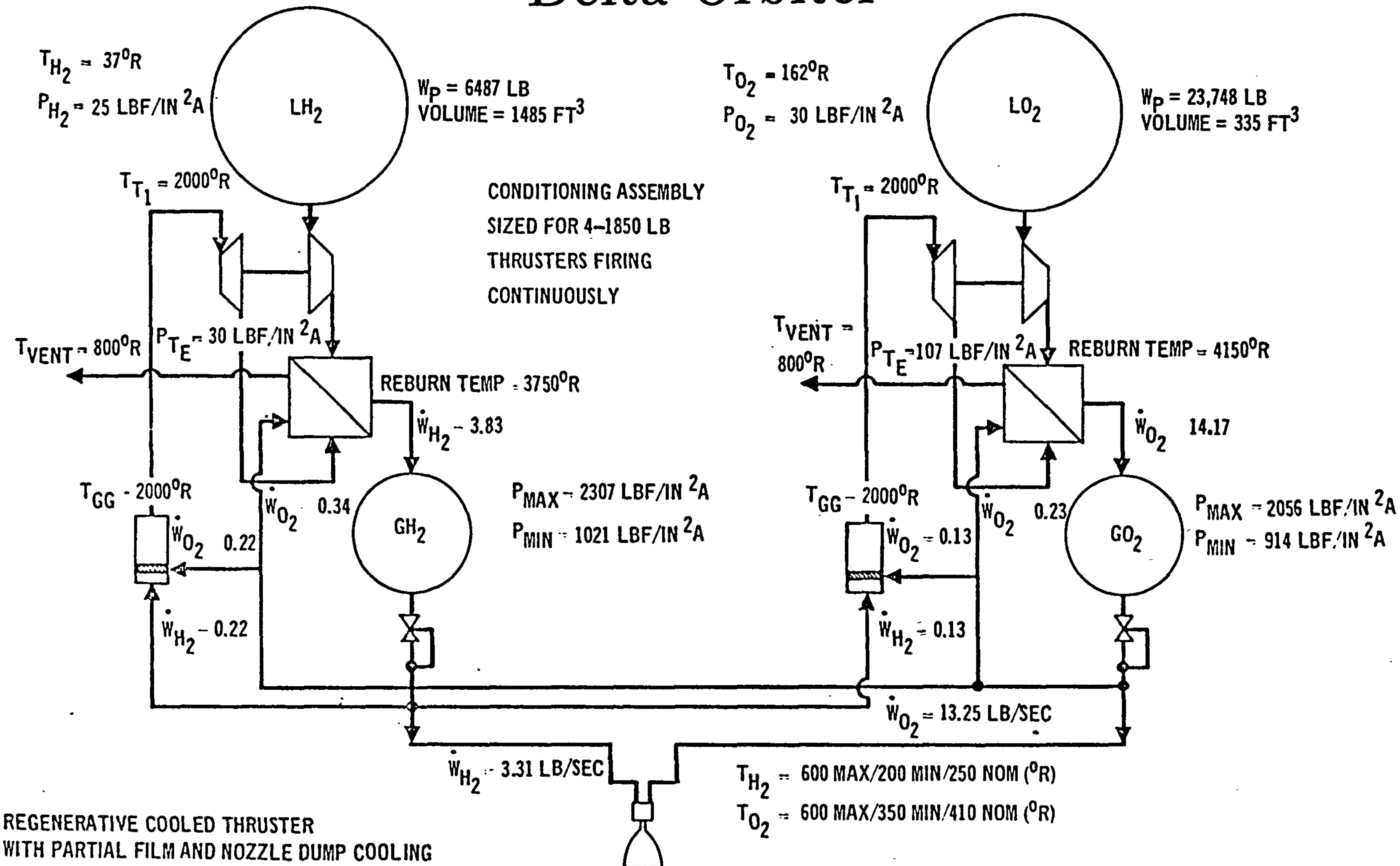
ORBITER-HIGH PRESSURE APS

THE PRESSURE, TEMPERATURE AND FLOW BALANCE CORRESPONDING TO THE PRECEDING HIGH PRESSURE APS DESIGN SUMMARY FOR ORBITER C ARE SHOWN IN THIS SCHEMATIC. THE OXYGEN AND HYDROGEN FLOW RATES TO THE THRUSTER ASSEMBLIES ARE FOR SIMULTANEOUS FIRING OF FOUR 1850 LB THRUST UNITS. THE THRUSTERS OPERATE AT A MIXTURE RATIO OF 4:1 AND THE CONDITIONER ASSEMBLIES OPERATE AT OVERALL MIXTURE RATIOS OF 2.4 AND 2.7 FOR THE HYDROGEN AND OXYGEN CONDITIONERS RESPECTIVELY. THE OVERALL SUBSYSTEM MIXTURE RATIO IS 3.87:1.

HIGH PRESSURE APS PRESSURES, TEMPERATURES AND FLOWS
Delta Orbiter



REGENERATIVE COOLED THRUSTER
WITH PARTIAL FILM AND NOZZLE DUMP COOLING

P_c – 500 LBF/IN ^{2}A MR – 4:1 F – 1850 LBF
F – 60 ATTITUDE CONTROL, 27 THRUSTERS
F – 120 + X TRANSLATION, 6 THRUSTERS

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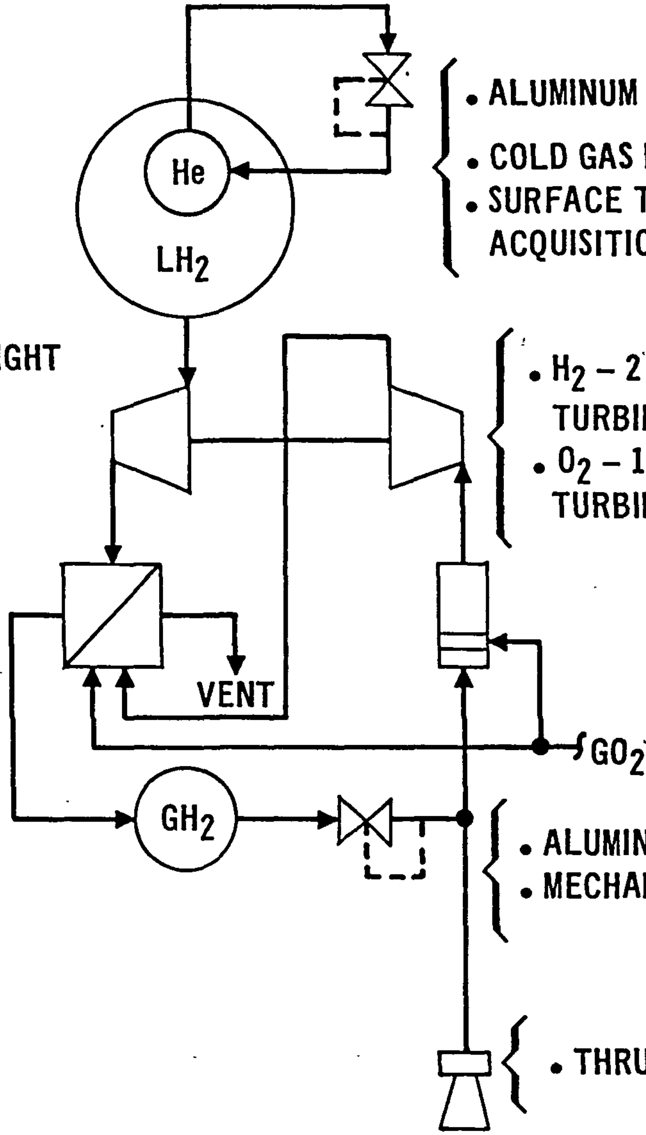
BOOSTER-HIGH PRESSURE APS SUMMARY

THIS FIGURE PROVIDES A SUMMARY OF THE HIGH PRESSURE APS DESIGN FOR THE SPACE SHUTTLE BOOSTER. THE DESIGN IS FUNCTIONALLY THE SAME AS THAT PREVIOUSLY DESCRIBED FOR THE ORBITER AND WAS DESIGNED IN THIS MANNER TO PROVIDE BOOSTER-ORBITER HARDWARE COMMONALITY. THE DIFFERENCES ARE SIMPLIFICATION OF THE PROPELLANT TANK AS THE THERMODYNAMIC VENT IS NOT REQUIRED AND THE PROPULSIVE VENT ASSEMBLY IS ELIMINATED.

HIGH PRESSURE BOOSTER APS SUMMARY



TOTAL SUBSYSTEM WEIGHT
5310 LB

SPECIFIC IMPULSE
410.8 SECONDS

- ALUMINUM TANK (POLYURETHANE INSULATION H_2 ONLY)
- COLD GAS HELIUM PRESSURIZATION
- SURFACE TENSION SCREEN CHANNELS FOR PROPELLANT ACQUISITION

- H_2 – 2 STAGE CENTRIFUGAL PUMP AND 3 STAGE AXIAL TURBINE
- O_2 – 1 STAGE CENTRIFUGAL PUMP AND 2 STAGE AXIAL TURBINE

- REBURN COUNTERFLOW HEAT EXCHANGER (H_2 – 3840°R, O_2 – 4160°R) PROPULSIVE VENT
- 2000°R GAS GENERATOR LINKED VALVES WITH O_2 TEMPERATURE FEEDBACK CONTROL, H_2 ACCUMULATOR PRESSURE FEEDBACK CONTROL

- ALUMINUM ACCUMULATOR (HPI INSULATED)
- MECHANICAL PRESSURE REGULATOR

- THRUSTER –18 THRUSTERS AT $\epsilon = 40$

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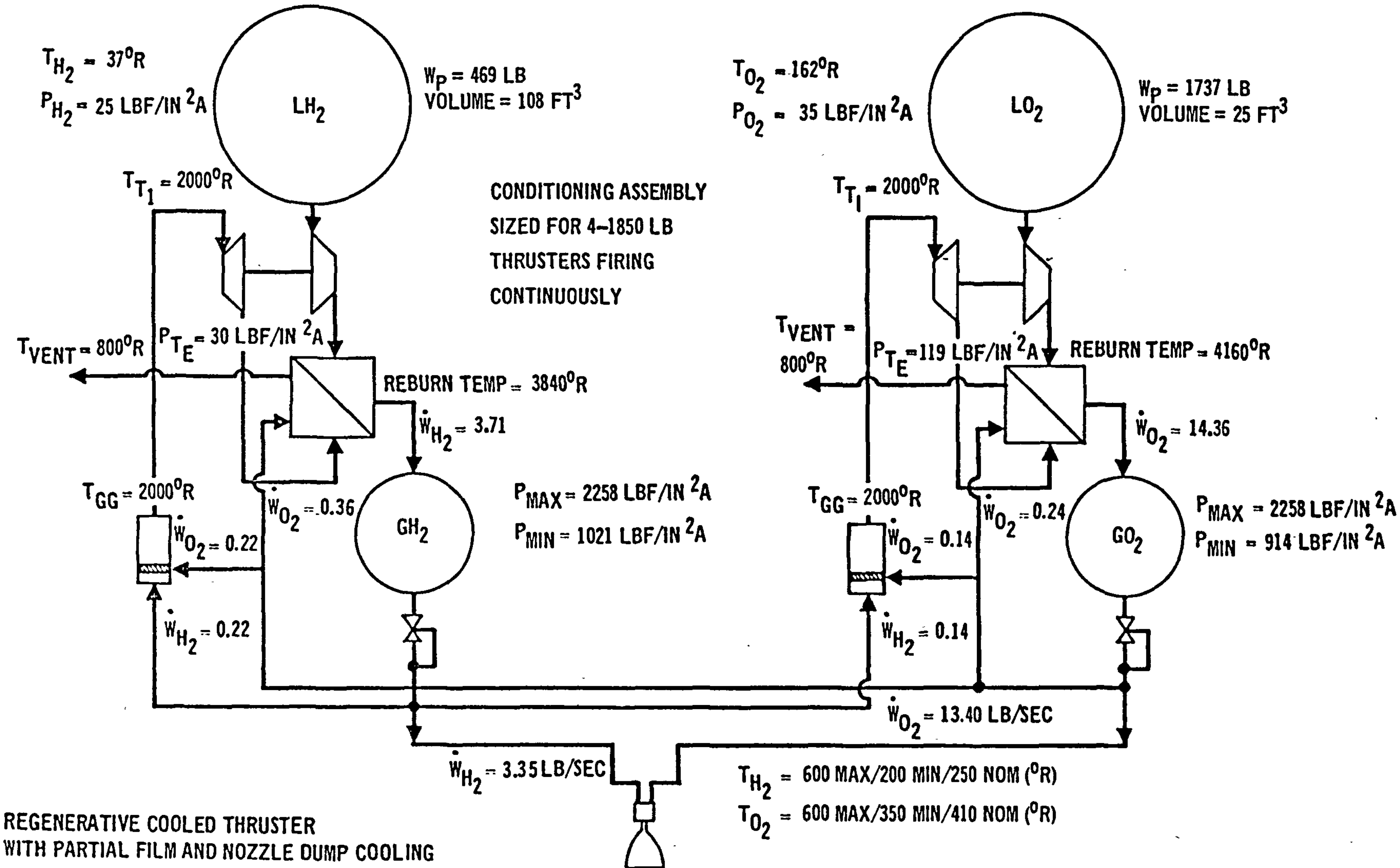
BOOSTER-HIGH PRESSURE APS

THIS FIGURE PROVIDES THE PRESSURE TEMPERATURE AND FLOW BALANCES FOR THE BOOSTER APS. WITH ONLY MINOR DIFFERENCES DESIGN LEVELS ARE AS PREVIOUSLY DESCRIBED FOR THE ORBITER.

HIGH PRESSURE APS PRESSURES, TEMPERATURES AND FLOWS
Booster



REGENERATIVE COOLED THRUSTER
WITH PARTIAL FILM AND NOZZLE DUMP COOLING
P_C = 500 LBF/IN²A MR = 4:1 F = 1850 LBF
ε = 40 ATTITUDE CONTROL, 18 THRUSTERS

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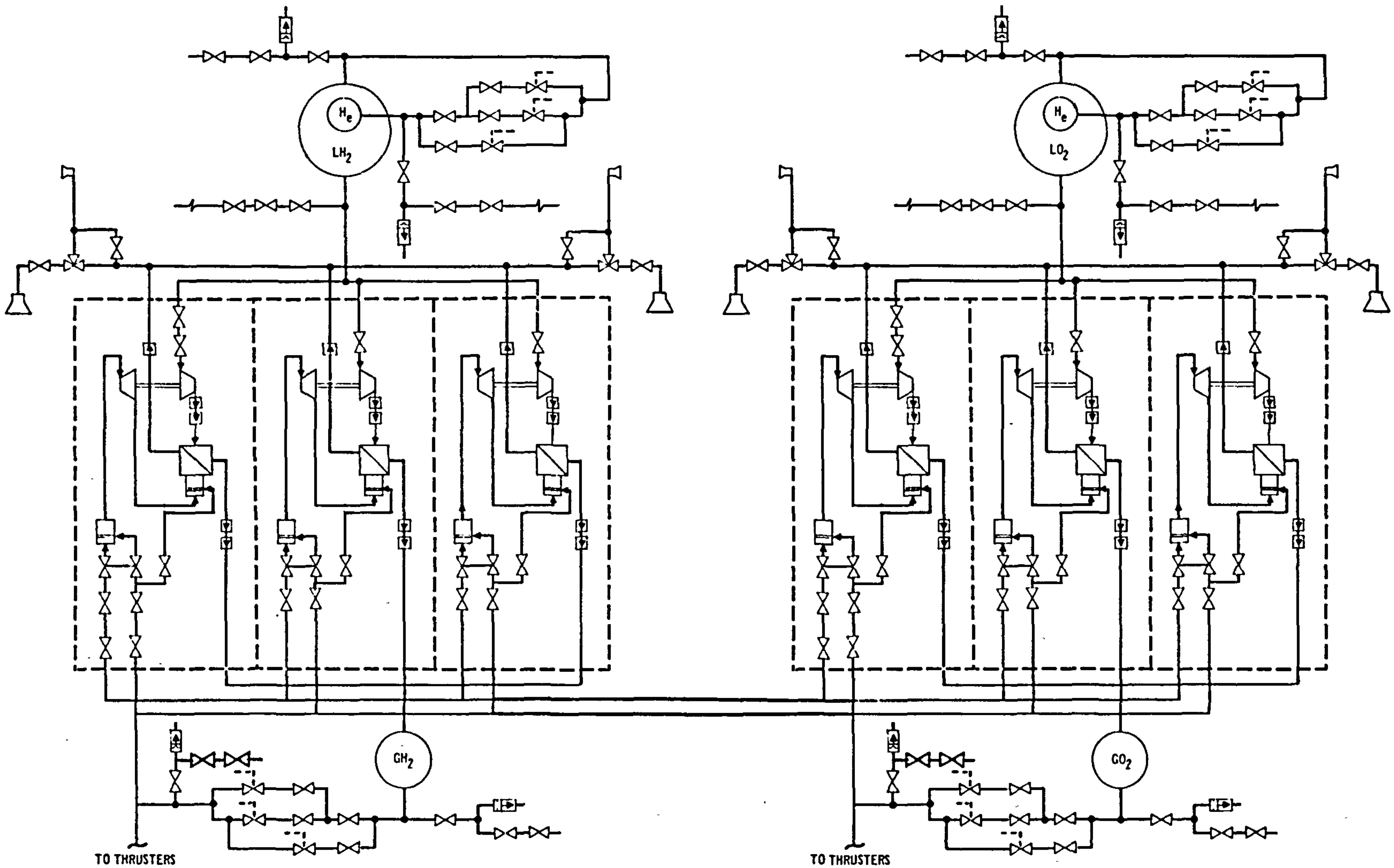
HIGH PRESSURE APS SCHEMATIC

THIS FIGURE PRESENTS THE HIGH PRESSURE APS SCHEMATIC WITH COMPLETE COMPONENT REDUNDANCY FOR ORBITER C. SUBSYSTEM SCHEMATICS FOR ORBITER B AND THE BOOSTER ARE SIMILAR. STRUCTURAL COMPONENTS SUCH AS TANKS, ACCUMULATORS, AND LINES ARE NOT DUPLICATED, ON THE ASSUMPTION THAT STRUCTURAL RELIABILITY IS EQUAL TO 1.0.

IN GENERAL, THE PHILOSOPHY IN IMPLEMENTING FAIL-OPERATIONAL/FAIL-SAFE REDUNDANCY IS TO PROVIDE TRIPLE REDUNDANCY WHERE FEASIBLE. THREE PARALLEL REDUNDANT REGULATORS ARE PROVIDED FOR EACH PRESSURE REGULATING FUNCTION. THREE COMPLETELY INDEPENDENT CONDITIONING ASSEMBLIES ARE PROVIDED FOR EACH PROPELLANT LOOP. WHEN THE PRIMARY CONDITIONING ASSEMBLY FAILS, IT IS ISOLATED AND A NEW CONDITIONING ASSEMBLY IS ACTIVATED.

THE REMAINING COMPONENTS ARE EITHER DOUBLY REDUNDANT, OR ARE DESIGNED IN SUCH A MANNER THAT THE FUNCTION OF A FAILED COMPONENT CAN BE MET BY ANOTHER COMPONENT. THE FOLLOWING FIGURE PROVIDES A SUPPLY LINE/THRUSTER SCHEMATIC.

HIGH PRESSURE ORBITER/BOOSTER SCHEMATIC



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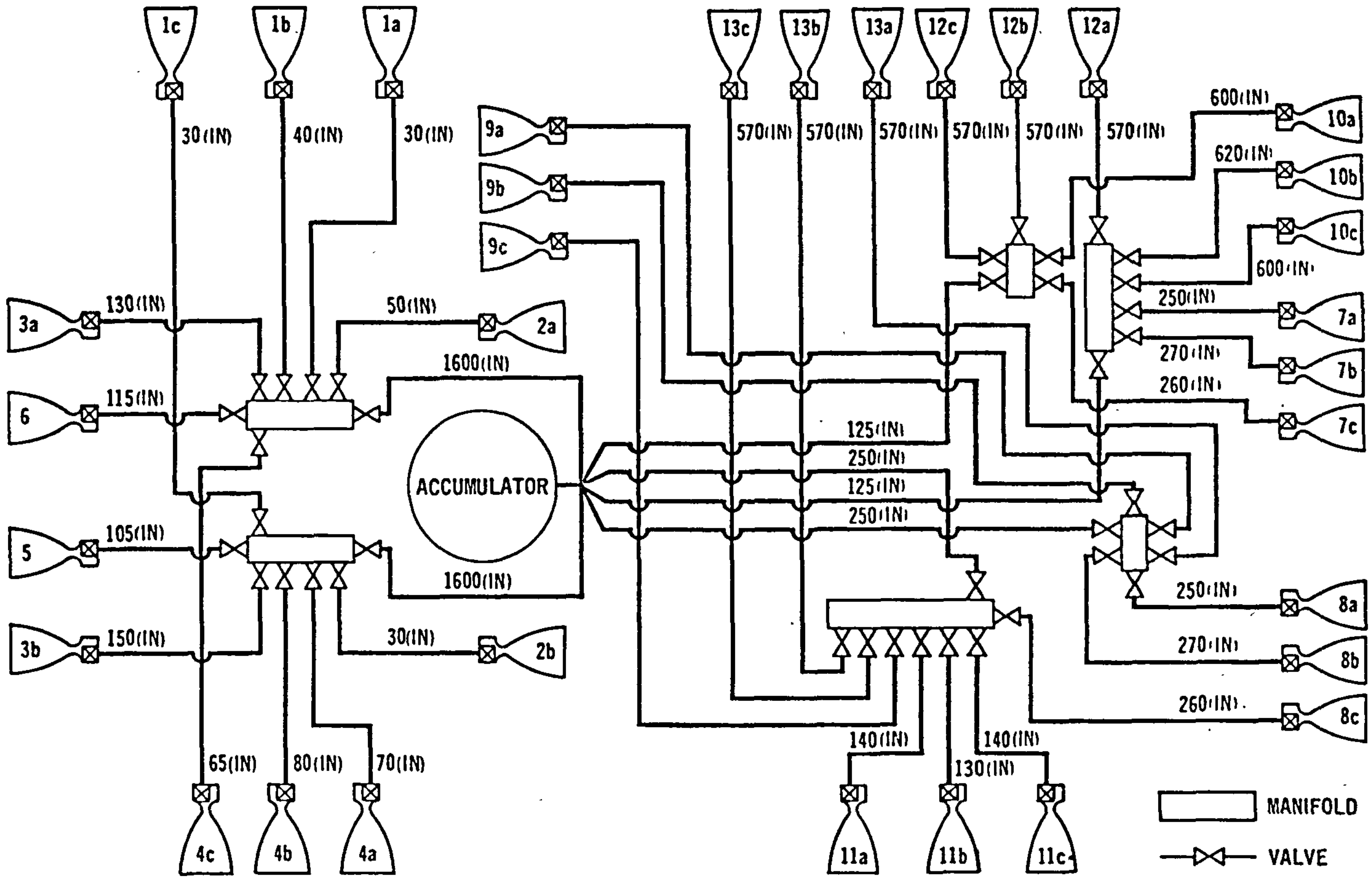
HIGH PRESSURE APS-SUPPLY LINE SCHEMATIC

PROVIDED IN THIS FIGURE IS A SCHEMATIC OF THE ORBITER C THRUSTER ASSEMBLIES AND THEIR PROPELLANT SUPPLY LINES FOR THE HYDROGEN SIDE OF THE APS. THE OXYGEN SIDE IS SIMILARLY IMPLEMENTED.

EACH THRUSTER HAS ISOLATION VALVES IN SERIES WITH THE THRUSTER PROPELLANT VALVES, ALLOWING INDIVIDUAL ISOLATION OF A FAILED-OPEN THRUSTER. A SECOND SET OF VALVES ISOLATES EACH PROPELLANT MANIFOLD TO PROVIDE ISOLATION OF A DOUBLE THRUSTER FAILURE AND ITS INDIVIDUAL ISOLATION VALVE.

IN GENERAL THE MAIN SUPPLY LINE SIZES ARE APPROXIMATELY TWO INCH DIAMETER AND THE BRANCH LINES TO THE THRUSTERS ARE APPROXIMATELY ONE INCH DIAMETER.

HIGH PRESSURE APS LINE LENGTHS AND DIAMETERS
Delta Orbiter



LINE TYPE	REQUIRED INNER DIAMETER	TUBE OUTER DIAMETER USED	MIN WALL THICK REQ'D	ACTUAL WALL THICKNESS
MAIN (H_2)	1.83 (IN)	2.12 (IN)	0.0316 (IN)	0.042 (IN)
MAIN (O_2)	1.90 (IN)	2.12 (IN)	0.0316 (IN)	0.042 (IN)
BRANCH (H_2)	0.92 (IN)	1.00 (IN)	0.015 (IN)	0.028 (IN)
BRANCH (O_2)	0.95 (IN)	1.00 (IN)	0.015 (IN)	0.028 (IN)

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HIGH PRESSURE APS-THRUSTER ASSEMBLY

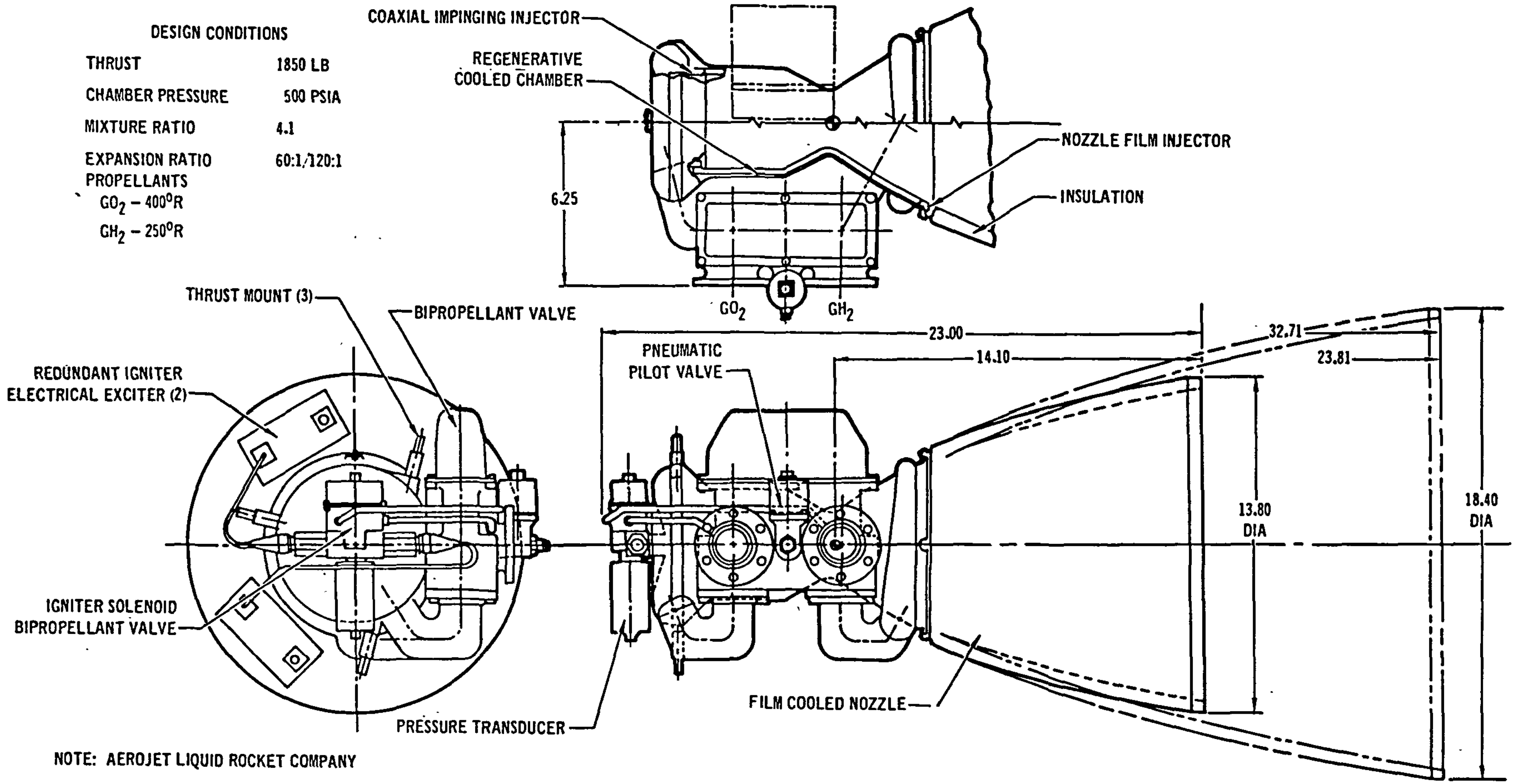
THE APS USES GASEOUS HYDROGEN-OXYGEN THRUSTERS TO PROVIDE BOTH VEHICLE CONTROL AND MANEUVERING IMPULSE. THE SELECTED CHAMBER CONFIGURATION, A PARTIAL REGENERATIVELY COOLED DESIGN (AS SHOWN IN THE ACCOMPANYING FIGURE), REPRESENTED A COMPROMISE BETWEEN A HIGH PERFORMANCE, FULLY REGENERATIVE COOLED ENGINE, AND EASILY SCARFED, LOWER PERFORMANCE FULLY FILM COOLED ENGINES. THE CHAMBER CYCLE LIFE REQUIREMENT IS MET BY THE ADDITION OF APPROXIMATELY 5 PERCENT FILM COOLING AT THE INJECTOR FACE ALONG THE CHAMBER WALL. THE SELECTED CHAMBER IS A SINGLE UP-PASS REGENERATIVE DESIGN, FROM AN AREA RATE OF 11 TO 1 TO THE INJECTOR FACE. THE SCARFING REQUIRED BY VARIOUS INSTALLATIONS IN THE VEHICLE IS EASILY ACCOMMODATED BY THE FILM COOLED NOZZLE ATTACHED AT THE 11 TO 1 AREA RATIO. THE NOZZLE SKIRT USES APPROXIMATELY 3 PERCENT OF THE HYDROGEN FLOW FOR COOLING

THE IGNITER SUBASSEMBLY CONSISTS OF A SEPARATE HIGH RESPONSE BIPROPELLANT VALVE, A COOLED IGNITION CHAMBER, AND THE SPARK PLUG. PRIMARY PROPELLANT FLOW TO THE THRUSTER IS CONTROLLED BY A PNEUMATICALLY ACTUATED BIPROPELLANT POPPET VALVE.

HIGH PRESSURE APS THRUSTER

DESIGN CONDITIONS

THRUST	1850 LB
CHAMBER PRESSURE	500 PSIA
MIXTURE RATIO	4.1
EXPANSION RATIO	60:1/120:1
PROPELLANTS	
GO_2 – 400°R	
GH_2 – 250°R	



NOTE: AEROJET LIQUID ROCKET COMPANY

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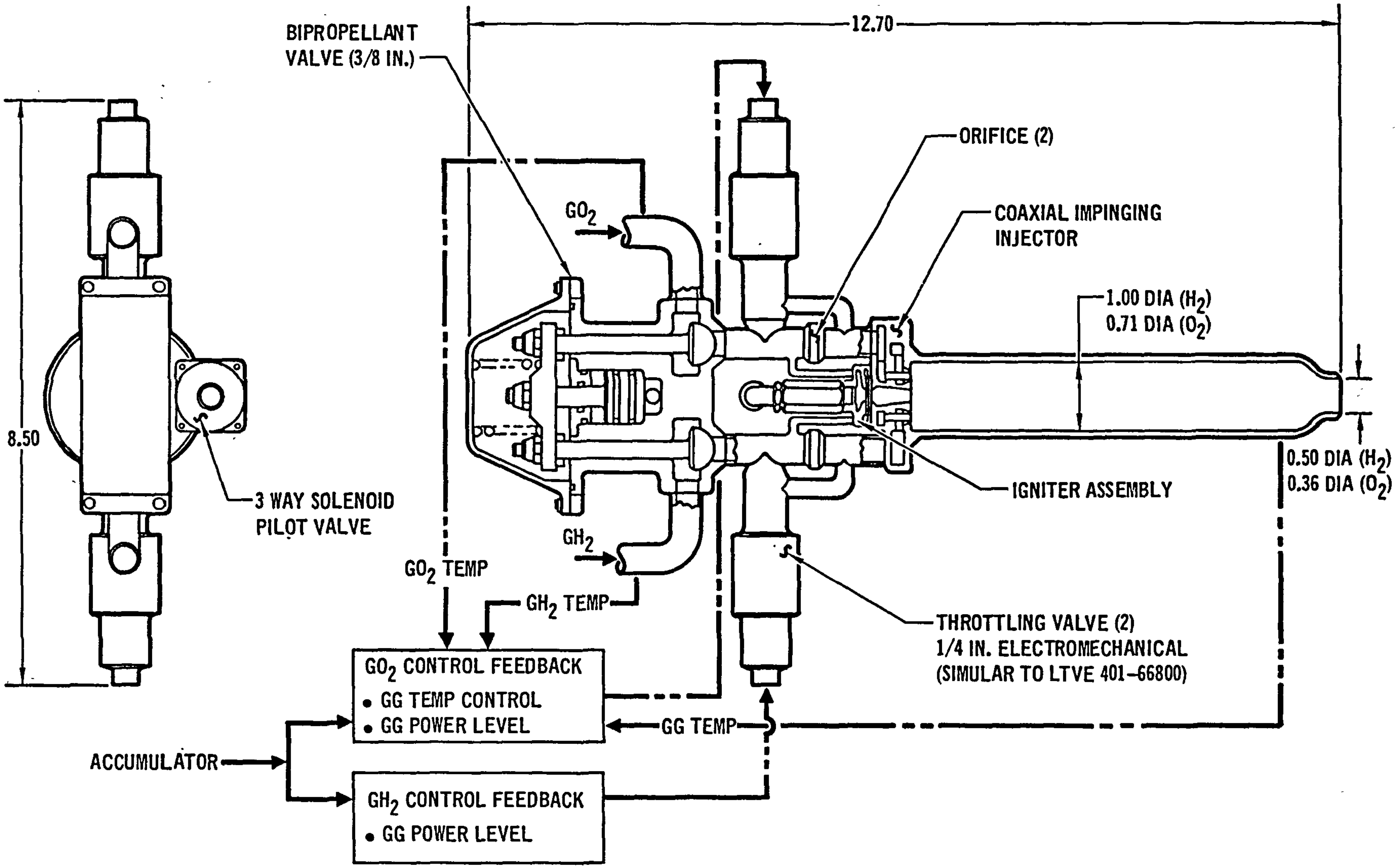
HIGH PRESSURE APS-GAS GENERATOR

THE HIGH PRESSURE APS USES GAS GENERATOR PRODUCTS TO PROVIDE POWER FOR TURBOPUMP OPERATION AND ENERGY TO THE HEAT EXCHANGERS. GAS GENERATORS ARE REQUIRED TO HAVE THROTTLING CAPABILITY TO MAINTAIN ACCUMULATOR PRESSURE AT OR NEAR SWITCHING PRESSURE LEVEL DURING STEADY STATE OPERATION AND PROVIDE INCREASED FLOW AND POWER TO THE TURBINE DURING CONDITIONER START-UP. IN ADDITION, THE GAS GENERATORS MUST MAINTAIN EXHAUST TEMPERATURES WITHIN LIMITS NECESSARY TO ENSURE TURBINE BLADING STRUCTURAL INTEGRITY.

THE HYDROGEN GAS GENERATOR DESIGN SELECTED FOR THE APS IS SHOWN IN THIS FIGURE.

THE GAS GENERATOR IS SEQUENCED ON WITH A SIGNAL TO OPEN THE LINKED GAS GENERATOR VALVES AND A SIGNAL TO THE ELECTRICAL IGNITER. OPENING THE BIPROPELLANT VALVE SENDS GASEOUS OXYGEN AND HYDROGEN THROUGH THE IGNITER AND THE PRIMARY INJECTOR PARALLEL FLOW CIRCUITS. CALIBRATED ORIFICES IN EACH PROPELLANT FLOW CIRCUIT BETWEEN THE LINKED BIPROPELLANT VALVE SEATS AND THE GAS GENERATOR INJECTOR LIMIT GAS GENERATOR OPERATION TO 80% POWER LEVEL. A BYPASS FLOW CIRCUIT AROUND EACH ORIFICE, WITH SEPARATELY ACTUATED THROTTLING VALVES, ALLOWS BYPASSING OF ADDITIONAL HYDROGEN OR OXYGEN AROUND THE CALIBRATED ORIFICES TO ADJUST THE POWER LEVEL OF THE GAS GENERATOR ON DEMAND. THE OXYGEN THROTTLE VALVE ALSO CONTROLS GAS GENERATOR TEMPERATURE AND MIXTURE RATIO IN RESPONSE TO GAS GENERATOR EXHAUST TEMPERATURES.

HIGH PRESSURE APS GAS GENERATOR



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HIGH PRESSURE APS-TURBOPUMPS

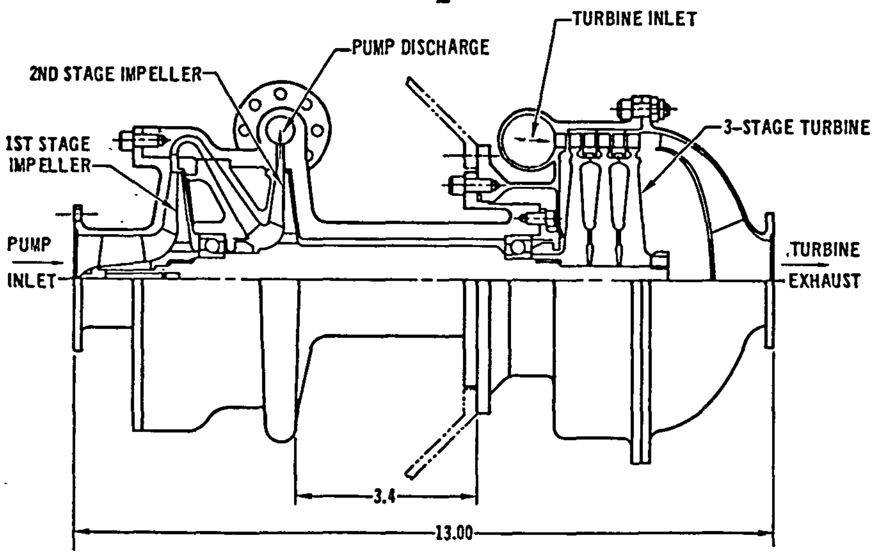
THE HIGH PRESSURE APS REQUIRES TURBOPUMPS TO DELIVER PROPELLANT FROM LOW PRESSURE CRYOGENIC SUPPLY TANKS TO A CONDITIONER ASSEMBLY AT THE PRESSURE REQUIRED FOR SUBSYSTEM OPERATION.

BASELINE DESIGNS SELECTED FOR OXYGEN AND HYDROGEN UNITS ARE PRESENTED IN THIS FIGURE. THE LO_2 TURBOPUMP CONSISTS OF A SINGLE STAGE PUMP AND A TWO-STAGE, PRESSURE COMPOUNDED, AXIAL FLOW TURBINE. PUMP IMPELLER AND TURBINE ROTORS ARE MOUNTED ON A COMMON SHAFT, SUPPORTED BY LO_2 COOLED/LUBRICATED ROLLING ELEMENT BEARINGS. BEARING COOLING/LUBRICATING FLOW IS TAPPED FROM THE HIGH PRESSURE PUMP DISCHARGE, DIRECTED THROUGH THE BEARING, AND REINTRODUCED TO THE MAINSTREAM FLOW IN A LOW PRESSURE REGION AT THE IMPELLER BACKSIDE HUB. THE MAGNITUDE OF THE BEARING COOLANT FLOW (5 PERCENT ALLOCATED) IS CONTROLLED BY HYDROSTATIC SEALS. THE FLOATING FEATURE OF HYDROSTATIC SEALS ELIMINATES THE RUB HAZARD NORMALLY ASSOCIATED WITH A FIXED-FLOW CONTROL LABYRINTH. THE INTERPROPELLANT SEAL, WHICH SEALS LO_2 FROM THE FUEL-RICH HOT GASES OF THE TURBINE, USES A TRIPLE VENT.

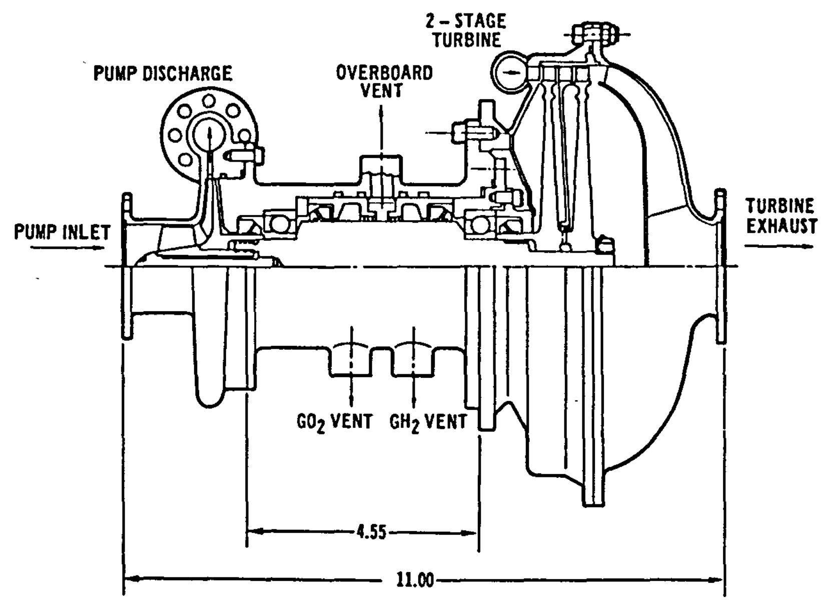
THE FUEL TURBOPUMP IS SIMILAR TO THE LO_2 TURBOPUMP. TWO PUMP STAGES ARE USED TO DEVELOP REQUIRED PRESSURE, WHILE THREE, PRESSURE COMPOUNDED, AXIAL FLOW STAGES ARE USED IN THE TURBINE. PUMP IMPELLERS AND TURBINE ROTORS ARE MOUNTED ON A COMMON SHAFT, AGAIN SUPPORTED BY LH_2 COOLED/LUBRICATED ROLLER ELEMENT BEARINGS. HYDROSTATIC SHAFT RIDING SEALS ARE USED TO CONTROL BEARING COOLANT FLOW ALLOCATION OF 5 PERCENT. THE FUEL TURBOPUMP DOES NOT REQUIRE AN INTERPROPELLANT SEAL TO SEPARATE THE PROPELLANT FROM HOT TURBINE GAS, SINCE LH_2 IS NONREACTIVE WITH THE FUEL-RICH TURBINE GASES. LIQUID HYDROGEN FLOW FROM THE TURBINE AND BEARING TO THE TURBINE IS MINIMIZED BY THE USE OF A HYDROSTATIC SEAL.

HIGH PRESSURE APS TURBOPUMPS

LH$_2$



LO$_2$



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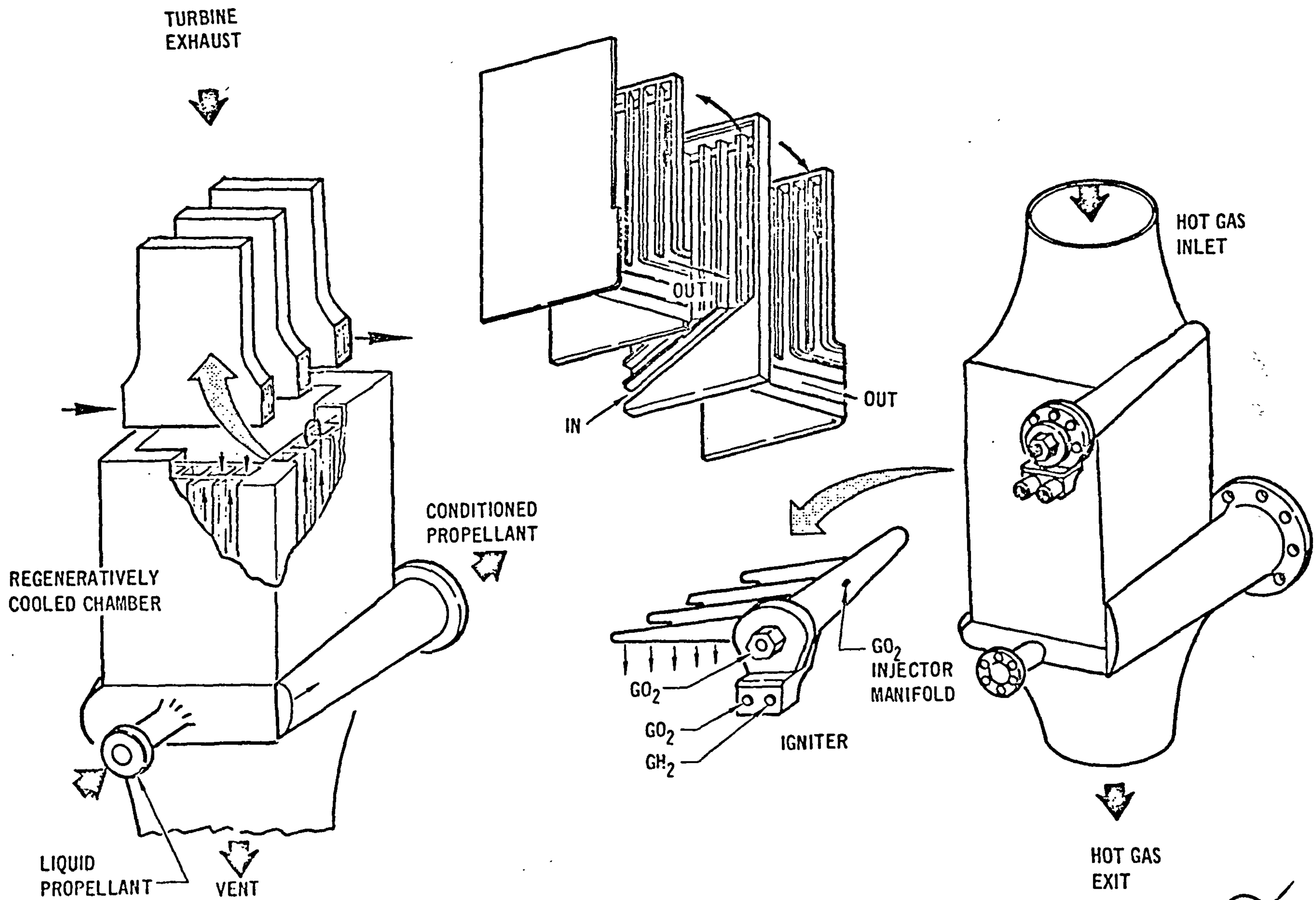
HIGH PRESSURE APS-HEAT EXCHANGER

THE HIGH PRESSURE APS USES A REBURN HEAT EXCHANGER SUBASSEMBLY TO HEAT PROPELLANT TO TEMPERATURES REQUIRED FOR RELIABLE THRUSTER IGNITION. IN THIS ASSEMBLY, COMBUSTION PRODUCTS FROM THE GAS GENERATOR AND TURBINE UPSTREAM ARE DIRECTED TO THE HEAT EXCHANGERS, WHERE SUPPLEMENTAL OXYGEN IS ADDED TO REBURN THE FUEL-RICH TURBINE EXHAUST, TO SUPPLY THE ENERGY NECESSARY FOR PROPELLANT CONDITIONING.

THE SELECTED DESIGN IS BASED ON APPLICATION OF INJECTOR PLATE FABRICATION TECHNOLOGY DEVELOPED FOR STAGED COMBUSTION CYCLES. THE DESIGN OF THE HEAT EXCHANGER IS SHOWN IN THIS FIGURE WHICH ILLUSTRATES THE PLATELET CONSTRUCTION TECHNIQUE. IN THIS SELECTED DESIGN, THE HOT AND COLD SIDE HEAT TRANSFER COEFFICIENTS ARE CONTROLLED BY FLOW PASSAGE TAILORING. AS SHOWN IN THE BLOWUP OF THE FIGURE, PROPELLANTS ENTER THE HEAT EXCHANGER BASE, FLOW TO THE TOP AND RETURN TO THE BASE TO REGENERATIVELY COOL THE HEAT EXCHANGER SHELL. AFTER RETURN, THE PROPELLANTS ARE COLLECTED IN A MANIFOLD AND DIRECTED INTO THE PLATES. PROPELLANT FLOWS UP THROUGH THE CENTER OF THE PLATES, SPLITS, AND IS DIVERTED TO THE OUTSIDE OF THE PLATE, WHERE IT RETURNS AND IS DISCHARGED INTO A COLLECTION MANIFOLD. DURING FLOW ALONG THE OUTSIDE OF THE PLATE, HEAT EXCHANGE BETWEEN PROPELLANTS AND HOT GAS PRODUCTS IS ACCOMPLISHED. TO DISTRIBUTE THE SUPPLEMENTAL OXYGEN UNIFORMLY, A DISTRIBUTION MANIFOLD IS LOCATED BETWEEN THE PLATES. A CATALYTIC IGNITER IN THE MANIFOLD IS USED AS THE IGNITION SOURCE FOR THE TURBINE EXHAUST GAS AND THE GASEOUS OXYGEN. DURING ACCUMULATOR RECHARGE, OXYGEN ADDITION IS CONTROLLED TO MAINTAIN A NEAR CONSTANT PROPELLANT TEMPERATURE AT THE HEAT EXCHANGER OUTLET. THIS PROVIDES THE HIGH HOT SIDE TEMPERATURES NECESSARY TO PRECLUDE CONDENSATION OF WATER VAPOR IN THE EXHAUST GAS.

HIGH PRESSURE APS HEAT EXCHANGER



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HIGH PRESSURE APS-TECHNOLOGY CRITIQUE

DURING THIS STUDY, ALL VIABLE CANDIDATES FOR HIGH PRESSURE AUXILIARY PROPULSION SUBSYSTEMS WERE COMPARED ON THE BASIS OF WEIGHT, TECHNOLOGY, SIMPLICITY, AND FLEXIBILITY TO REQUIREMENTS. FROM THIS COMPARISON, THE TURBOPUMP APS CONCEPT WAS CLEARLY THE BEST SELECTION FOR BOTH BOOSTERS AND ORBITERS. ON THE BASIS OF THESE RESULTS, NASA SELECTED THE TURBOPUMP APS FOR PRELIMINARY DESIGN, AND SPECIFIED THAT PRELIMINARY DESIGN EFFORT BE DIRECTED TO AN APS CAPABLE OF PERFORMING ALL MANEUVERS.

IN A PRELIMINARY DESIGN THE SPECIFIED CONCEPT WAS STUDIED TO DEVELOP AN OPTIMIZED DESIGN CAPABLE OF SATISFYING SHUTTLE REQUIREMENTS. THIS WAS ACCOMPLISHED THROUGH A SERIES OF INTERRELATED STUDIES ASSOCIATED WITH APS COMPONENTS, ASSEMBLIES, AND SUBSYSTEMS WHICH PROVIDED AN INTEGRATED APS DESIGN AND ESTABLISHED ITS PERFORMANCE CHARACTERISTICS IN AN INSTALLED CONFIGURATION. THE RESULTING PRELIMINARY DESIGN EVIDENCES NO REQUIREMENTS WHICH ARE CONSIDERED TO BE UNREASONABLE EXTENSIONS OF STATE-OF-THE-ART TECHNOLOGY. HOWEVER, IN TERMS OF THRUST LEVEL, TOTAL IMPULSE, AND REUSE CAPABILITY, SHUTTLE REQUIREMENTS FOR FAR BEYOND THOSE FOR ANY PREVIOUS CONTROL PROPULSION SUBSYSTEM. CONSEQUENTLY, NO APS COMPONENTS CAPABLE OF SATISFYING THESE REQUIREMENTS EXIST TODAY.

FUNDED TECHNOLOGY DEVELOPMENT PROGRAMS ARE CURRENTLY UNDERWAY FOR SOME OF THE MORE CRITICAL COMPONENTS, SUCH AS VALVES, IGNITION, AND THRUST CHAMBER COOLING, AND FOR CERTAIN ASPECTS OF THE PROPELLANT STORAGE ASSEMBLY. THOSE AREAS OF TECHNOLOGY WHICH APPEAR TO REQUIRE ADDITIONAL EFFORT IN SUPPORT OF APS DEVELOPMENT ARE SUMMARIZED IN THIS FIGURE ALONG WITH AN ASSESSMENT OF THE IMPACT, IN TERMS OF WEIGHT, ASSOCIATED WITH FAILURE TO DEVELOP THE SPECIFIED TECHNOLOGY.

CRITIQUE OF HIGH PRESSURE APS TECHNOLOGY

TECHNOLOGY CONCERN	ALTERNATIVE APPROACH	IMPACT OF CHANGE
• PROPELLANT ACQUISITION ASSEMBLY DESIGN AND VERIFICATION	• USE OF MULTIPLE SMALL REFILLABLE TANKS	• INCREASED WEIGHT (APPROXIMATELY 400 LB), INCREASED DESIGN AND CONTROL COMPLEXITY AND REDUCED APS FLEXIBILITY
• HIGH TEMPERATURE REBURN HEAT EXCHANGER DESIGN	• SERIES-STAGED COMBUSTION HEAT EXCHANGERS TO LIMIT MATERIAL TEMPERATURE • CONVENTIONAL–MODERATE TEMPERATURE HEAT EXCHANGERS (2000°R) • NO REBURN HEAT EXCHANGER	• INCREASED OPERATIONAL AND CONTROL COMPLEXITY WITH MULTIPLE OXYGEN INJECTION (IGNITION) • INCREASED APS WEIGHT, APPROXIMATELY 1800 LB • INCREASED APS WEIGHT, APPROXIMATELY 2200 LB
• HIGH PERFORMANCE INSULATION REUSABILITY – PROPELLANT TANKS – DISTRIBUTION LINES	• VACUUM JACKETED DEWARS • VACUUM JACKETED LINES	• INCREASED APS WEIGHT, APPROXIMATELY 650 LB • MAJOR INCREASES IN INSTALLATION/DESIGN COMPLEXITY, INCREASED APS WEIGHT, APPROXIMATELY 400 LB
• TURBOPUMP LIFE CAPABILITY	• REDUCE OPERATING REQUIREMENTS • PERIODIC REPLACEMENT	• INCREASED APS WEIGHT, APPROXIMATELY 700 LB FOR FACTOR OF 2 REDUCTION IN CYCLE CAPABILITY PREDICTION • INCREASED MAINTENANCE/TURN AROUND TIME

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CRITIQUE OF HIGH PRESSURE APS TECHNOLOGY
Continued

TECHNOLOGY CONCERN	ALTERNATIVE APPROACH	IMPACT OF CHANGE
• TURBOPUMP COOLING/RESPONSE	• INCREASED COOLANT FLOW • REDUCED RESPONSE REQUIREMENT	• 40 LB INCREASE FOR TWICE DESIGN COOLANT FLOW • 300 LB INCREASE FOR FACTOR OF FOUR IN EQUIVALENT START TIME
• THRUSTER ASSEMBLY PERFORMANCE	• REDUCTION IN PERFORMANCE REQUIREMENTS	• INCREASED APS WEIGHT. APPROXIMATELY 100 LB PER SEC I_{SP} REDUCTION
• THRUSTER ASSEMBLY LIFE CAPABILITY	• INCREASED COOLANT FLOW • PERIODIC REPLACEMENT	• INCREASED APS WEIGHT, APPROXIMATELY 300 LB FOR FACTOR OF 2 ERROR IN CYCLE CAPABILITY PREDICTION • INCREASED MAINTENANCE/TURN-AROUND TIME
• PRESSURE VESSEL CYCLE LIFE CAPABILITY	• INCREASED DESIGN MARGIN	• INCREASED APS WEIGHT, APPROXIMATELY 500 LB FOR 50% INCREASE IN SAFETY FACTORS
• CONTROL COMPONENT LIFE CAPABILITY VALVES, IGNITERS, REGULATORS	• PERIODIC REPLACEMENT	• INCREASED MAINTENANCE/TURN-AROUND TIME

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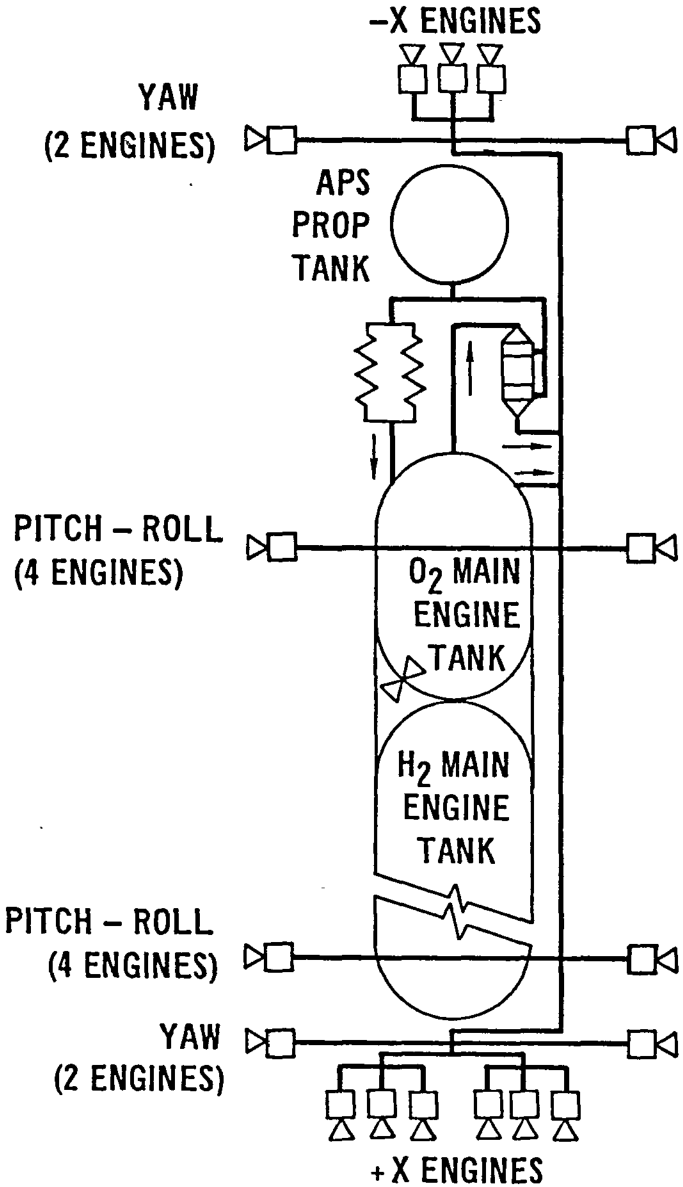
ORBITER-LOW PRESSURE APS SUMMARY

FOR THE SPACE SHUTTLE ORBITER, THE PREFERRED LOW PRESSURE APS APPROACH WAS IDENTIFIED AS ONE IN WHICH AN ORBIT MANEUVERING SUBSYSTEM (OMS) PERFORMS ALL HIGH TOTAL IMPULSE MANEUVERS, SUCH AS ORBIT CIRCULARIZATION, PLANE CHANGES, AND DEORBIT FUNCTIONS, WHILE THE APS PROVIDES ALL ATTITUDE CONTROL AND VERNIER MANEUVERS, SUCH AS MIDCOURSE CORRECTIONS AND DOCKING. APS VELOCITY INCREMENTS OF APPROXIMATELY 40 FT/SEC MAXIMUM WERE DETERMINED TO BE THE MOST FAVORABLE ALLOCATION OF +X AXIS MANEUVERS BETWEEN APS AND OMS.

THIS FIGURE PROVIDES A SUMMARY OF THE LOW PRESSURE APS DESIGNED FOR THE ORBITER ONLY. THE OXYGEN SIDE OF THE APS IS SHOWN. THE HYDROGEN SIDE IS SIMILAR. THE SELECTED APS DESIGN USES THE MAIN ENGINE TANKS AS LOW PRESSURE GAS ACCUMULATORS. PROPELLANTS FROM SEPARATE LIQUID TANKS ARE USED FOR MAIN ENGINE TANK RESUPPLY. PROPELLANTS ARE FIRST CIRCULATED THROUGH TUBULAR, PASSIVE HEAT EXCHANGERS WHERE THEY ARE VAPORIZED AND SUPERHEATED PRIOR TO INJECTION INTO THE MAIN ENGINE TANKS. DURING MAJOR APS OPERATION, WARM PROPELLANT VAPORS FROM THE MAIN ENGINE TANKS ARE MIXED WITH ADDITIONAL LIQUID PROPELLANTS IN A DOWNSTREAM LIQUID/VAPOR MIXER AND SUPPLIED TO THE ENGINES AT CONSTANT TEMPERATURE AND PRESSURE (CONSTANT DENSITY).

LOW PRESSURE ORBITER APS SUMMARY



TOTAL SUBSYSTEM WEIGHT	SPECIFIC IMPULSE
ORBITER B – 12,868 (APS)	381 SEC
– 24,703 (OMS)	439 SEC
– 37,571 TOTAL	

PROPELLANT STORAGE ASSEMBLY

- ALUMINUM TANK WITH HPI INSULATION, FIBERGLASS COVER
- COOLING – VAPOR VENT USING THROTTLED H_2 LIQUID
- PRESSURIZATION – COLD GAS HELIUM (O_2)
  – PUMPS (HELIUM PREPRESS) (H_2)
- PROPELLANT ACQUISITION – SURFACE TENSION SCREEN CHANNELS

PROPELLANT CONDITIONING ASSEMBLY

- TYPE – MULTIPLE TUBE/HEAT SINK
- LOCATION – INTEGRAL WITH MAIN ENGINE TANK WALL
- ATTACHMENT – TUBE FLANGE RIVETED TO TANK STIFFENERS
- MATERIAL – 2014–T6 ALUMINUM

LIQUID/VAPOR MIXING ASSEMBLY

- LIQUID INJECTION ELEMENT LOCATED IN MAIN SUPPLY LINES
- LIQUID CONTROL – CAVITATING VENTURI THROTTLE VALVE
- GAS CONTROL – IRIS THROTTLE VALVE

ENGINE ASSEMBLIES

- HYDROGEN FILM – COOLED DESIGN
- PNEUMATICALLY ACTUATED, PILOT OPERATED, COAXIAL VALVES
- OPTIMUM EXPANSION RATIO IS 8:1

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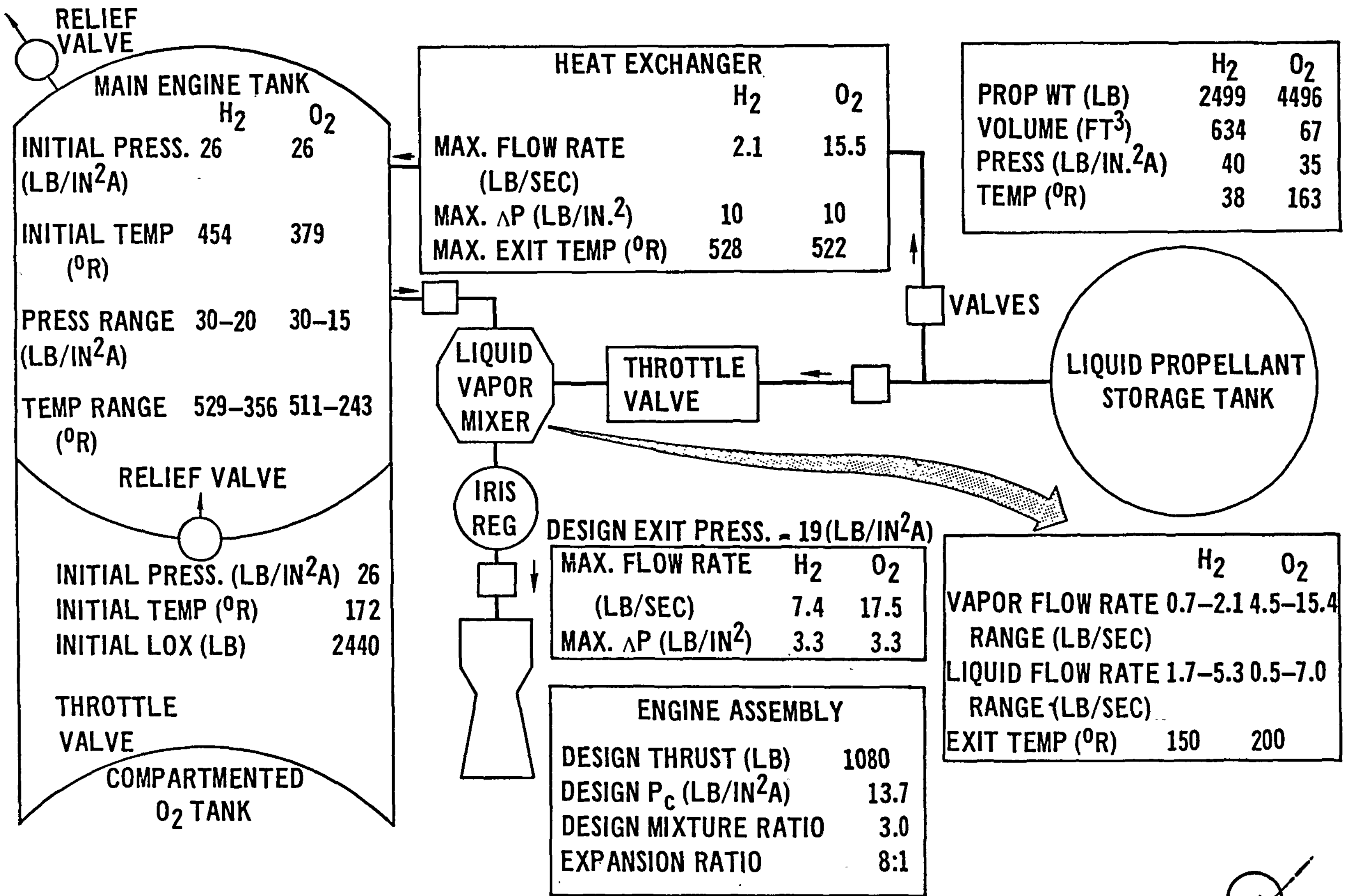
ORBITER-LOW PRESSURE APS

THIS FIGURE PROVIDES PRESSURE, TEMPERATURE AND FLOW BALANCES FOR THE ORBITER LOW PRESSURE APS DESCRIBED PREVIOUSLY. THE APS DESIGN USES THIRTY-THREE 1080 LB THRUST ENGINES WITH AN 8:1 NOZZLE EXPANSION RATIO OPERATING AT A MIXTURE RATIO OF 3.0.

MAIN ENGINE TANKS ARE USED AS GAS ACCUMULATORS WITH AN OPERATING PRESSURE RANGE OF 16 TO 30 LBF/IN^{2}A. WHEN PRESSURES DROP BELOW 24 LBF/IN^{2}A, ADDITIONAL PROPELLANT IS RESUPPLIED FROM THE LIQUID STORAGE ASSEMBLY. RESUPPLY PROPELLANT IS FIRST VAPORIZED AND SUPERHEATED BY PASSIVE HEAT EXCHANGERS BEFORE INJECTION INTO MAIN ENGINE TANKS. DURING MAJOR APS MANEUVERS, WARM MAIN TANK PROPELLANT VAPORS ARE MIXED WITH COLD PROPELLANTS IN A DOWNSTREAM MIXING CHAMBER, FOR SUPPLY AT CONSTANT TEMPERATURE AND PRESSURE.

LOW PRESSURE ORBITER APS PRESSURES, TEMPERATURES AND FLOWS



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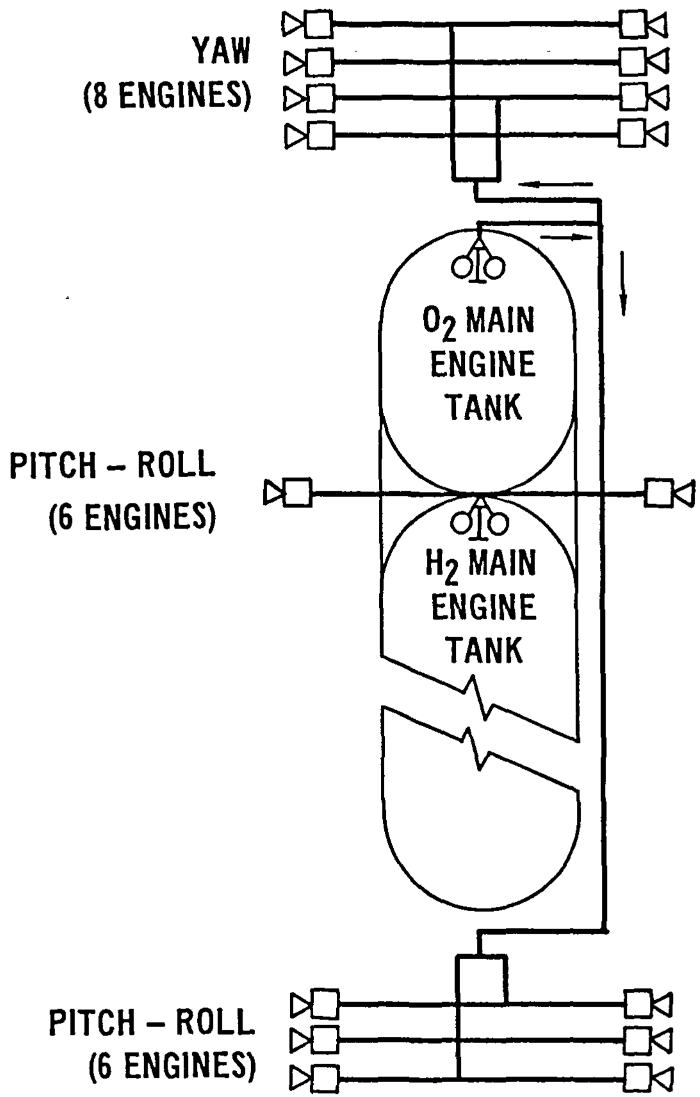
BOOSTER-LOW PRESSURE APS SUMMARY

THIS FIGURE PROVIDES A DESIGN SUMMARY OF THE LOW PRESSURE APS FOR THE SPACE SHUTTLE BOOSTER. THE RESULTING DESIGN USES THE LOW PRESSURE CONCEPT IN ITS SIMPLEST FORM.

THE SELECTED BOOSTER APS CONSISTS ONLY OF PROPELLANT DISTRIBUTION AND ENGINE ASSEMBLIES. IT OPERATES ENTIRELY FROM MAIN ENGINE TANKS RESIDUAL PROPELLANTS, REQUIRING NO ADDITIONAL PROPELLANT TANKAGE, PUMPS, CONDITIONING EQUIPMENT, OR MIXING ASSEMBLIES.

LOW PRESSURE BOOSTER APS SUMMARY



TOTAL SUBSYSTEM WEIGHT, LB

BOOSTER APS – 5647

PROPELLANT DISTRIBUTION NETWORK

- 2219 ALUMINUM DISTRIBUTION LINES
- ALUMINUM SOCKET/BELLOWS ANGULAR COMPENSATORS
- ALUMINUM BELLOWS LINEAR COMPENSATORS
- ALUMINUM VISOR TYPE ISOLATION VALVES (DC REVERSIBLE MOTOR DRIVE WITH CLUTCH BREAK)

ENGINE ASSEMBLIES

- HYDROGEN FILM-COOLED DESIGN
- PNEUMATICALLY ACTUATED, PILOT OPERATED, COAXIAL VALVES
- ELECTRIC TORCH IGNITION
- OPTIMUM EXPANSION RATIO IS 2:1

SPECIFIC IMPULSE

340 SEC

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BOOSTER-LOW PRESSURE APS

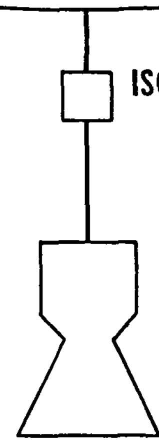
THIS FIGURE PROVIDES PRESSURE, TEMPERATURE AND FLOW BALANCES FOR THE BOOSTER, LOW PRESSURE APS. THE SUBSYSTEM OPERATES ENTIRELY FROM AVAILABLE MAIN ENGINE TANK PROPELLANT VAPORS IN A BLOWDOWN MODE, OVER A PRESSURE RANGE OF 26 TO 17 LBF/IN^2. LIQUID RESIDUALS REMAINING IN THE TANK AT MAIN ENGINE SHUTDOWN ARE PREVENTED FROM ENTERING THE FEED SYSTEM BY G-SENSITIVE VALVES LOCATED AT TANK OUTLETS. DISTRIBUTION AND ENGINE ASSEMBLIES ARE SIZED TO PROVIDE 2500 LB THRUST AT THE END OF BLOWDOWN, WHEN TANK PRESSURES AND TEMPERATURES ARE LOWEST. A MIXTURE RATIO OF 2.0 AND A NOZZLE EXPANSION RATIO OF 2:1 PROVIDE MINIMUM SUBSYSTEM WEIGHT.

INITIAL BOOSTER RESIDUALS, AS SPECIFIED BY A CONTRACT, WERE AT VAPOR PRESSURES OF 26 LBF/IN^{2}A AND VAPOR TEMPERATURES OF 450°R (H_2) and 520°R (O_2). THESE HIGH PRESSURANT TEMPERATURES INCURRED A SEVERE SUBSYSTEM WEIGHT PENALTY DUE TO REDUCED PROPELLANT DENSITY. AT THIS LOWER INITIAL DENSITY, THE AMOUNT OF PRESSURE DECAY IS INCREASED AND DESIGN PRESSURE OF LINES AND ENGINES IS REDUCED. THIS, OF COURSE, RESULTS IN INCREASED LINE AND ENGINE SIZE AND WEIGHT. MINIMUM MAIN ENGINE TANK PRESSURES DURING APS USAGE AND RESULTANT APS WEIGHTS WERE DETERMINED AS A FUNCTION OF INITIAL VAPOR TEMPERATURE AND APS MIXTURE RATIO. THIS ANALYSIS DEMONSTRATED THAT HIGH TEMPERATURE PROPELLANT VAPOR WOULD UNNECESSARILY PENALIZE THE LOW PRESSURE APS. FOR THIS REASON, THE INITIAL HYDROGEN VAPOR TANK VAPOR TEMPERATURE WAS REDUCED FROM 450°R TO 260°R AND THE INCREASED VAPOR RESIDUAL ASSOCIATED WITH THIS TEMPERATURE REDUCTION WAS ASSESSED AGAINST THE APS AS A WEIGHT PENALTY. NO TEMPERATURE REDUCTIONS WERE NECESSARY FOR THE OXYGEN TANK. THESE TEMPERATURES CONSTRAIN OPERATING TANK PRESSURES TO A MINIMUM OF 17 LBF/IN^{2}A.

LOW PRESSURE BOOSTER APS PRESSURES, TEMPERATURES AND FLOWS

MAIN ENGINE TANK

	H_2	O_2
INITIAL PRESSURE (LB/IN.^{2} A)	26	26
INITIAL TEMPERATURE (^oR)	260	520
PRESSURE RANGE (LB/IN.^{2})	26–17.3	26–17.5
TEMPERATURE RANGE (^oR)	260–156	520–384
LIQUID RESIDUAL	6060	14,285



	H_2	O_2
MAXIMUM FLOW RATE (LB/SEC)	16.0	36.8
MAXIMUM ΔP (LB/IN^2)	3.0	3.0

ENGINE ASSEMBLY

DESIGN THRUST (LB)	2500
DESIGN P_C (LB/IN^{2}A)	11
DESIGN MIXTURE RATIO	2
EXPANSION RATIO	2:1

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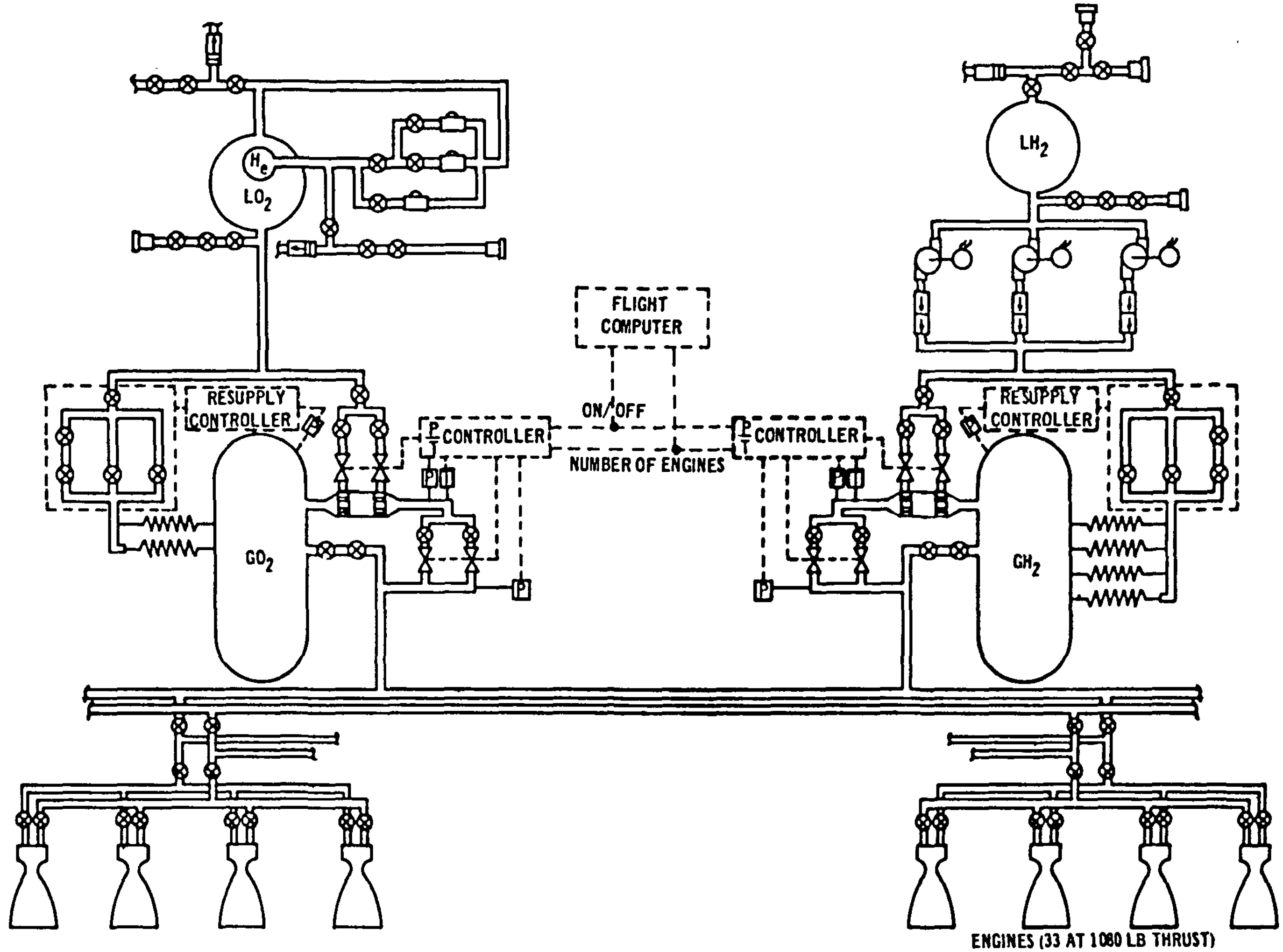
LOW PRESSURE APS-ORBITER SCHEMATIC

THIS FIGURE PROVIDES A COMPLETE SCHEMATIC FOR THE ORBITER, LOW PRESSURE APS. THE GENERAL PHILOSOPHY FOR IMPLEMENTING FAIL OPERATIONAL/FAIL SAFE REDUNDANCY FOR THE LOW PRESSURE APS WAS TO MINIMIZE THE NUMBER OF COMPONENTS IN ORDER TO SAVE WEIGHT WHILE MAINTAINING FULL FAIL OPERATIONAL/FAIL SAFE CAPABILITY. THREE PARALLEL REDUNDANT REGULATORS CONTROL LO_2 TANK PRESSURE. THE VALVE MODULE CONTROLLING FLOW THROUGH THE PROPELLANT HEAT EXCHANGERS PROVIDES THREE PARALLEL FLOW PATHS AS WELL AS SERIES SHUTOFF VALVES TO CONTROL LEAKAGE. BECAUSE THE LOW PRESSURE APS CAN PROVIDE LIMIT CYCLE OPERATION IN A SIMPLE BLOWDOWN MODE WITHOUT LIQUID VAPOR MIXING, ONLY DOUBLE REDUNDANCY WAS NECESSARY FOR LIQUID VAPOR MIXING AND ENGINE PROPELLANT PRESSURE REGULATION. THE NUMBER OF ENGINES NECESSARY TO PROVIDE NOMINAL IMPULSE REQUIREMENTS USUALLY ALLOWED ISOLATION OF ENGINES BY GROUPS AFTER THE FIRST FAILURE. IN SOME INSTANCES, HOWEVER, INDIVIDUAL ISOLATION FOR EACH ENGINE IS PROVIDED AFTER FIRST FAILURE. A SECOND LEVEL OF ISOLATION VALVES PROVIDE BACK-UP FOR A DOUBLE FAILURE OF AN ENGINE VALVE AND A FIRST LEVEL ISOLATION VALVE.

IN GENERAL, SUPPLY LINE SIZES TO ENGINE GROUPS ARE APPROXIMATELY FIVE INCH DIAMETER AND BRANCH LINES TO INDIVIDUAL ENGINES ARE APPROXIMATELY THREE INCH DIAMETER.

LOW PRESSURE APS
ORBITER BASELINE SCHEMATIC



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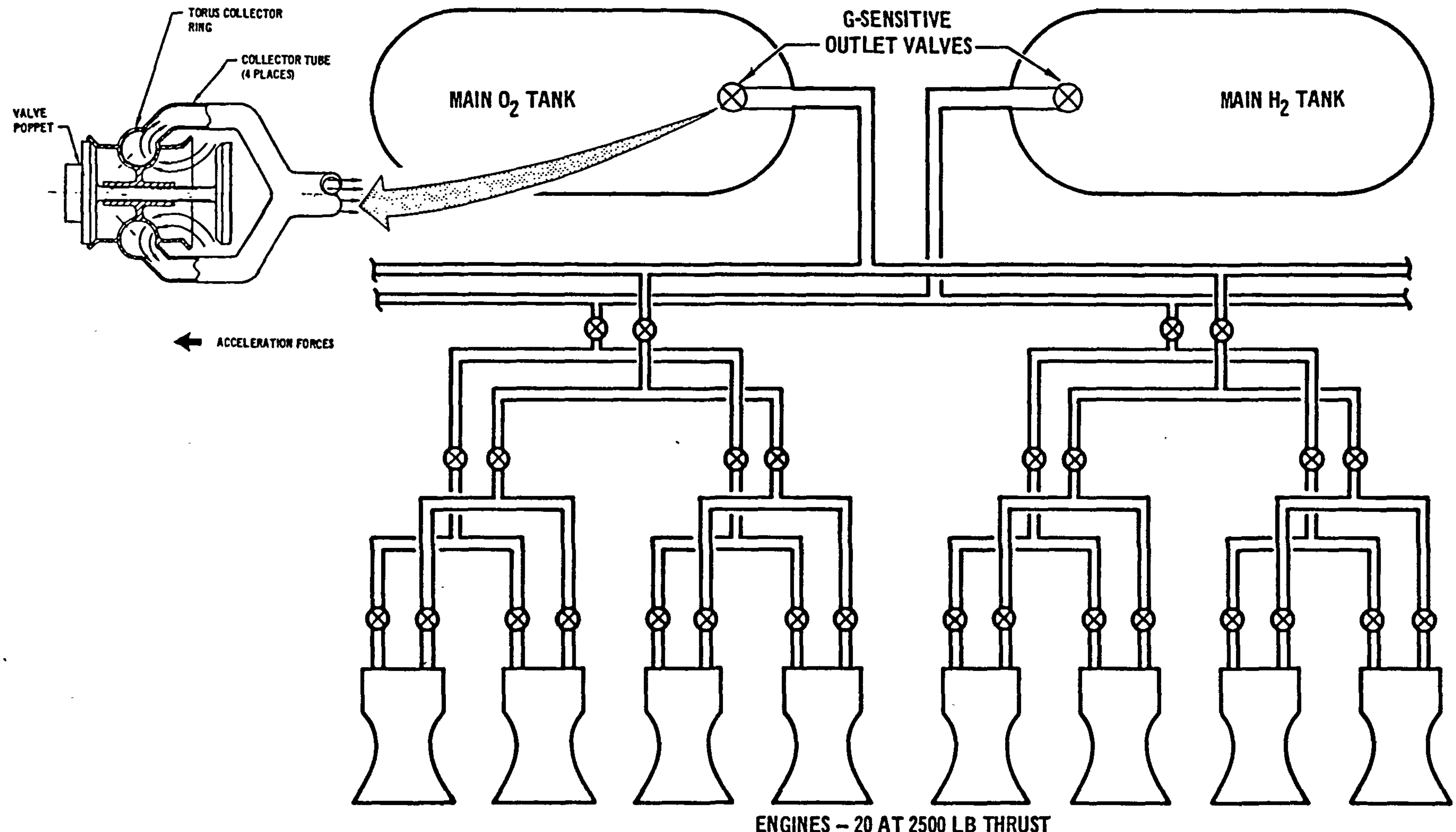
LOW PRESSURE APS-BOOSTER SCHEMATIC

THIS FIGURE PROVIDES A SCHEMATIC OF THE BOOSTER LOW PRESSURE APS. THE DESIGN PHILOSOPHY FOR THRUSTER ISOLATION IN THE EVENT OF FAILURE IS SIMILAR TO THAT FOR THE ORBITER. TWO THRUSTERS ARE ISOLATED AFTER FIRST FAILURE AND A BANK OF FOUR IS ISOLATED AFTER THE SECOND FAILURE. LINE SIZES TO SUPPLY A BANK OF ENGINES ARE APPROXIMATELY EIGHT INCH DIAMETER AND INDIVIDUAL ENGINE SUPPLY LINES ARE APPROXIMATELY SIX INCH DIAMETER.

LOW PRESSURE APS BOOSTER BASELINE SCHEMATIC



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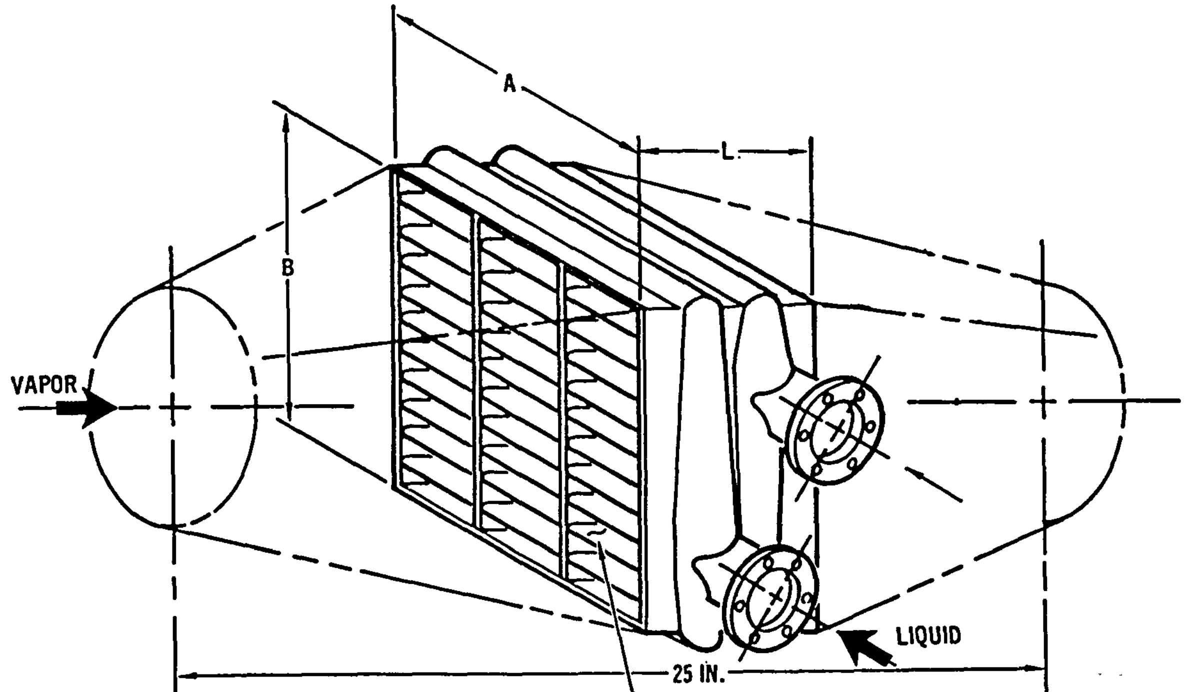
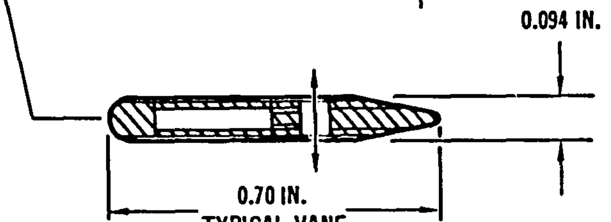
LOW PRESSURE APS-LIQUID/VAPOR MIXER

THE LOW PRESSURE APS USES A LIQUID VAPOR MIXER TO CONTROL ENGINE INLET CONDITIONS DURING MAJOR APS OPERATIONS. DURING LOW DEMAND ATTITUDE CONTROL OPERATIONS, ALL PROPELLANT IS EXTRACTED FROM THE MAIN ENGINE TANKS; THERE IS NO DOWNSTREAM LIQUID INJECTION IN THE LIQUID/VAPOR MIXER. FOR MAJOR APS OPERATIONS, THE MIXER ASSEMBLY PROVIDES CONSTANT PRESSURE AND TEMPERATURE AT THE ENGINE INLETS, THUS ACHIEVING CONSTANT THRUST LEVEL AND MIXTURE RATIO. THE MIXER ASSEMBLY CONSISTS OF A LIQUID INJECTION/VAPOR MIXING CHAMBER AND TWO INDEPENDENT CONTROLS - A PRESSURE REGULATOR LOCATED DOWNSTREAM OF THE MIXING CHAMBER, AND A LIQUID FLOW RATE CONTROLLER. COLD LIQUID PROPELLANT IS INJECTED INTO THE MIXING CHAMBER, WHERE IT IS COMBINED WITH WARM PROPELLANT VAPORS (EXTRACTED FROM THE TANK) TO ACHIEVE A CONSTANT PROPELLANT DENSITY CORRESPONDING TO PREDEFINED MIXER TEMPERATURE AND REGULATED PRESSURE. MINIMUM ENGINE INLET TEMPERATURES ARE 200°R FOR OXYGEN AND 150°R FOR HYDROGEN, BASED ON ENGINE IGNITION CRITERIA AND MAXIMUM ALLOWABLE INJECTOR TEMPERATURE DIFFERENTIAL. MIXER PHYSICAL CHARACTERISTICS REQUIRED TO ACHIEVE THESE CONDITIONS ARE AS SHOWN IN THIS FIGURE.

LOW PRESSURE APS PROPELLANT LIQUID/VAPOR MIXER





ITEM	O_2	H_2
MIXER WIDTH A, IN	7.40	8.20
HEIGHT B, IN	7.40	8.20
LENGTH L, IN	5.00	5.00
DROPLET DIAMETER, IN.	0.002	0.002
TOTAL PRESSURE DROP GAS STREAM, LBF/IN^2	1.0	1.5
PRESSURE DROP, LIQUID, LBF/IN^2	10.0	7.8
GAS STREAM MACH NO.	0.2	0.2
LIQUID DISCHARGE VELOCITY, MAX, FPS	20	47

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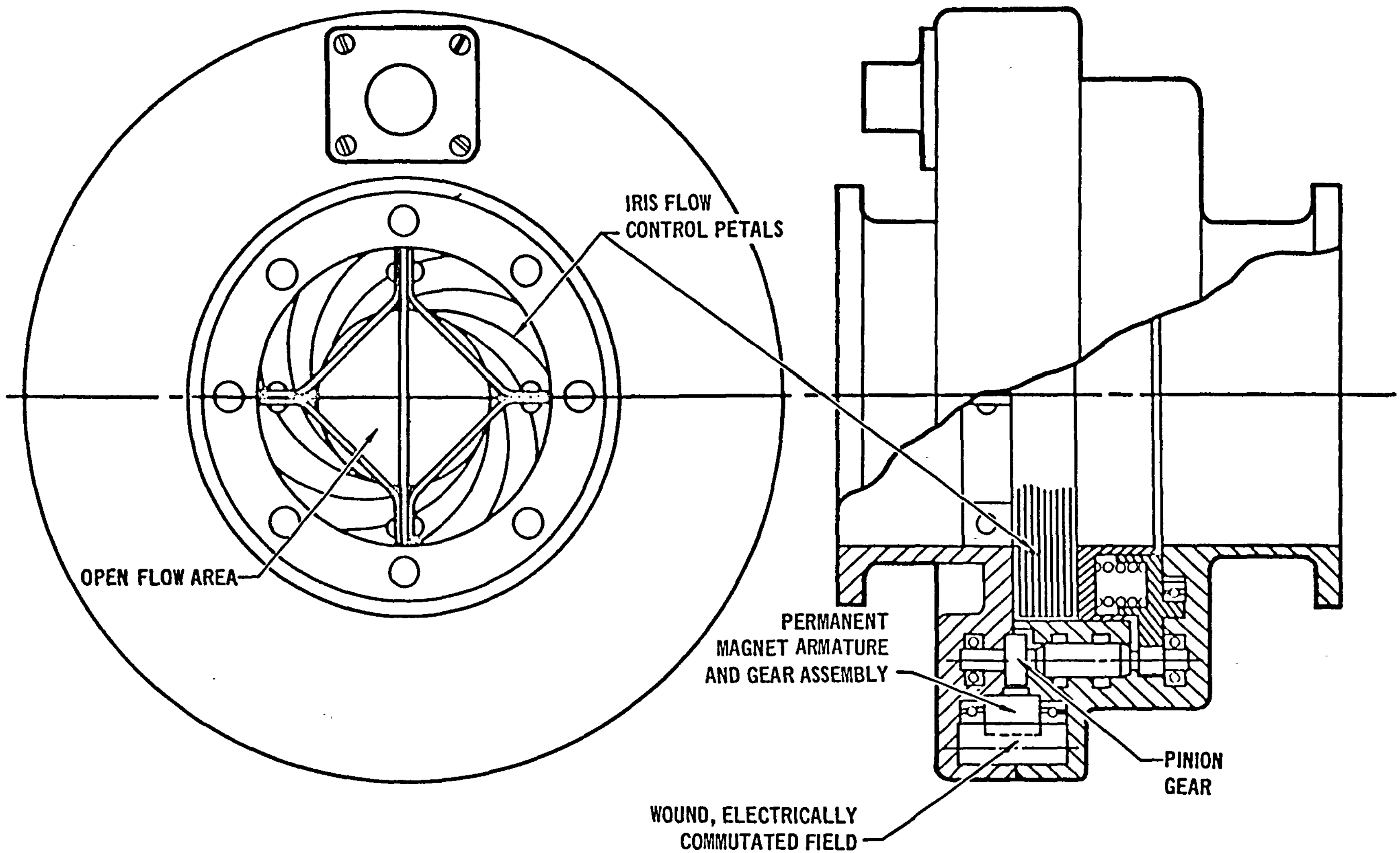
LOW PRESSURE APS-PRESSURE REGULATOR

A PRESSURE REGULATOR DOWNSTREAM OF THE LIQUID/VAPOR MIXING CHAMBER IS REQUIRED TO CONTROL ENGINE INLET PRESSURE. IN THE LOW PRESSURE APS DESIGN THE NONSEALING "IRIS" VALVE PICTURED IN THIS FIGURE IS USED FOR PRESSURE CONTROL. THE ANNULAR TORQUE MOTOR/GEAR SYSTEM PICTURED POSITIONS THE VALVE APETURE ACCORDING TO MEASURED PRESSURE AT THE VALVE OUTLET, MAINTAINING 19 PSIA AT THE ENGINE INLET DURING ALL MAJOR APS OPERATIONS. DURING LOW DEMAND PULSE MODE OPERATIONS THE O_2 REGULATOR IS DRIVEN FULL OPEN AND THE H_2 REGULATOR IS MAINTAINED AT ITS LAST SET POSITION.

LOW PRESSURE APS
IRIS REGULATOR DESIGN



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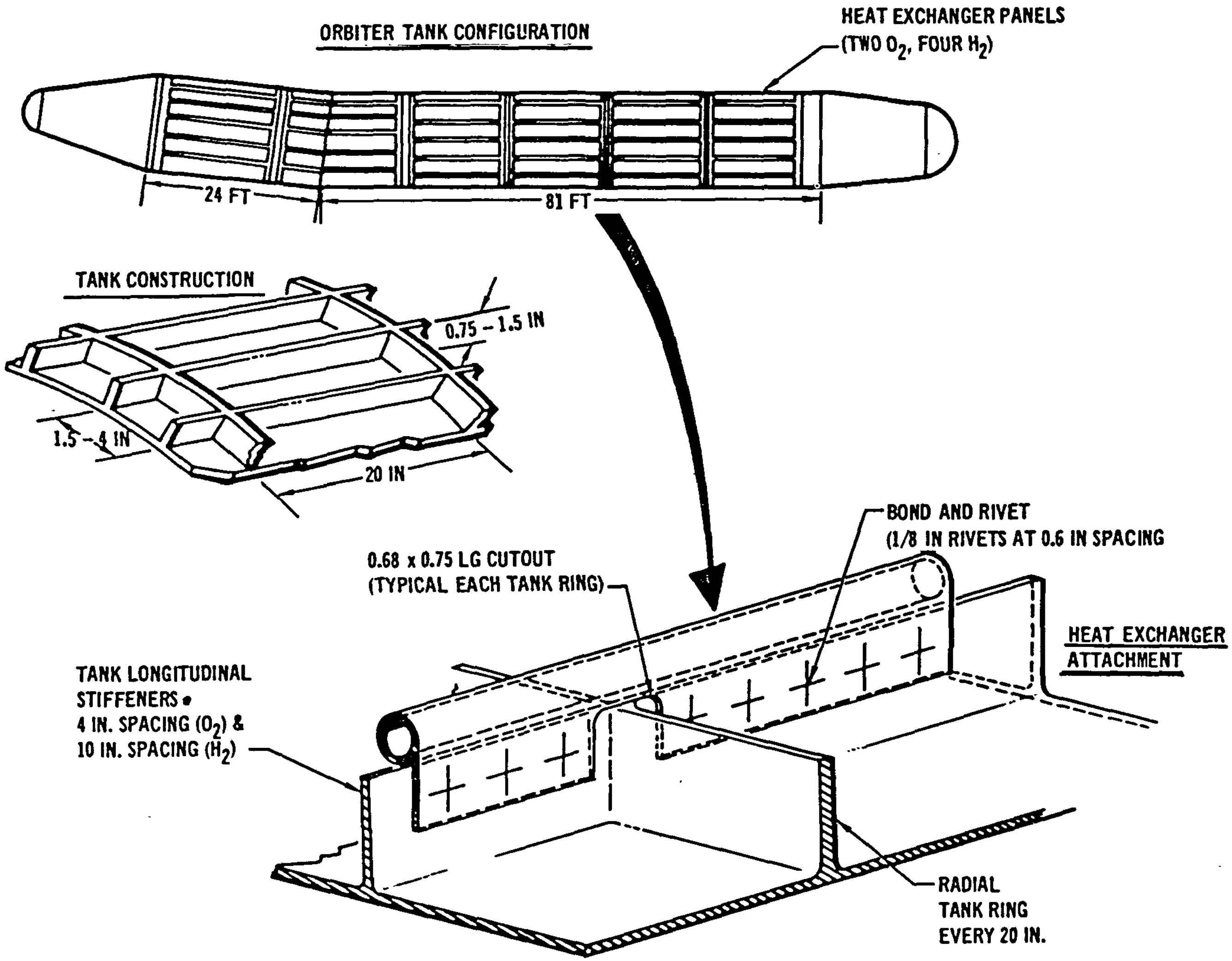
LOW PRESSURE APS-PASSIVE HEAT EXCHANGER

DURING LOW PRESSURE APS OPERATION, WHEN THE AMOUNT OF PROPELLANT VAPOR WITHIN THE MAIN ENGINE TANKS FALLS BELOW 24 LBF/IN^{2}A, ADDITIONAL PROPELLANT IS RESUPPLIED FROM LIQUID STORAGE TANKS. THIS PROPELLANT IS FIRST CIRCULATED THROUGH A PASSIVE HEAT EXCHANGER, WHERE IT IS VAPORIZED AND SUPERHEATED TO DESIRED RESUPPLY TEMPERATURES. THE PROPELLANT CONDITIONING ASSEMBLY IS COMPOSED OF MAIN ENGINE TANKS, MULTIPLE TUBE/HEAT EXCHANGER, AND ASSOCIATED CONTROLS FOR PROPELLANT RESUPPLY. THE PASSIVE HEAT EXCHANGER DESIGN CONCEPT IS SHOWN IN THIS FIGURE. HYDROGEN AND OXYGEN HEAT EXCHANGERS ARE MADE OF ALUMINUM TUBING TO ACHIEVE HIGH HEAT TRANSFER RATES AND MINIMUM WEIGHT. THE FIGURE SHOWS HOW THE TUBES ARE ATTACHED TO TANK LONGITUDINAL STRUCTURAL STIFFENERS. THE SECTION MODULUS OF THE TUBE AND FLANGE ADDS TO LONGITUDINAL RIB STIFFNESS, PERMITTING A REDUCTION IN TANK RIB HEIGHT AND WEIGHT. HOWEVER, THE WEIGHT REDUCTION CAN ONLY BE APPLIED TO THE OXYGEN TANK SINCE THE HYDROGEN RIB HEIGHT IS AT THE MINIMUM REQUIRED FOR RIVETING. THE OXYGEN HEAT EXCHANGER IS DIVIDED INTO TWO PANELS, 17.5 FEET LONG, EACH WITH 154 TUBES, APPROXIMATELY 0.4 INCH IN DIAMETER. THE HYDROGEN HEAT EXCHANGER IS DIVIDED INTO FOUR 15 FOOT PANELS, EACH CONSISTING OF SIXTY-TWO 0.3 INCH DIAMETER TUBES. PROPELALNT GAS VELOCITIES IN THE TUBES ARE LIMITED TO MACH 0.3. CONDITIONING ASSEMBLIES WERE SIZED (TUBE LENGTH, SPACING, ETC.) TO MAINTAIN FINAL TANK PRESSURES OF APPROXIMATELY 20 LBF/IN^{2}A. HEAT EXCHANGER DESIGN INLET PRESSURES ARE 35 LBF/IN^{2}A FOR OXYGEN AND 57 TO 35 LBF/IN^{2}A FOR HYDROGEN.

LOW PRESSURE APS TANK-MOUNTED HEAT EXCHANGER ASSEMBLY



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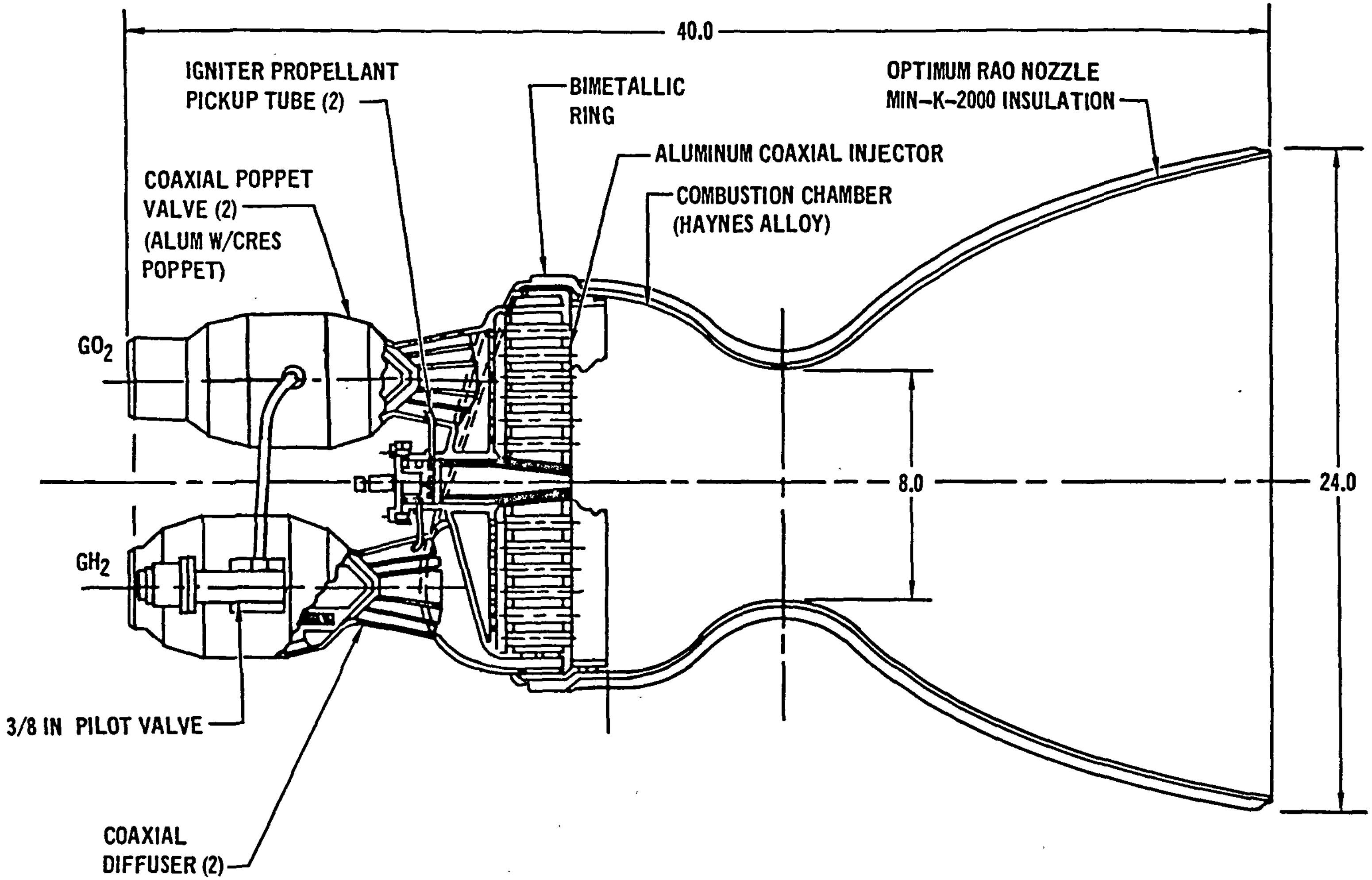
LOW PRESSURE APS-ENGINE ASSEMBLY

APS ENGINE ASSEMBLIES PROVIDE ALL ATTITUDE CONTROL, Y AND Z AXES TRANSLATION MANEUVERS, AND X AXIS VERNIER TRANSLATION MANEUVERS FOR THE SPACE SHUTTLE ORBITER. THIRTY-THREE 1080 LB THRUST ENGINES ARE USED IN THE ORBITER SUBSYSTEM AND TWENTY 2,500 LB THRUST ENGINES ARE NEEDED IN THE BOOSTER APS. AN ENGINE ASSEMBLY INCLUDES PROPELLANT CONTROL VALVES, INJECTOR, COMBUSTION CHAMBER, AND NOZZLE. ORBITER ENGINE DESIGN AND DIMENSIONS ARE GIVEN IN THIS FIGURE. THE ENGINE ASSEMBLY IS A HYDROGEN FILM-COOLED DESIGN CONTAINING A MULTIPLE ELEMENT, COAXIAL INJECTOR AND AN 8:1 OPTIMUM RAO CONTOUR NOZZLE FOR IMPROVED PERFORMANCE. COMBUSTION CHAMBER AND NOZZLE ARE FABRICATED OF THIN WALL, HIGH TEMPERATURE STEELS, WHILE THE ENGINE HEAD END ASSEMBLY IS CONSTRUCTED OF ALUMINUM TO MINIMIZE WEIGHT. THE DISSIMILAR MATERIALS ARE ATTACHED BY MEANS OF A BIMETALIC JOINT. IGNITION IS ACHIEVED BY MEANS OF AN ELECTRIC SPARK TORCH IGNITER. PNEUMATICALLY ACTUATED, PILOT OPERATED, COAXIAL VALVES PROVIDE A 50 MS OPENING RESPONSE AT A PILOT VALVE INLET PRESSURE OF 250 LBF/IN^{2}A. A SINGLE SOLENOID PILOT VALVE ON EACH ENGINE PROVIDES SIMULTANEOUS ACTUATION OF BOTH PROPELLANT VALVES.

LOW PRESSURE APS ENGINE ASSEMBLY



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LOW PRESSURE APS-TECHNOLOGY CRITIQUE

THE LOW PRESSURE AUXILIARY PROPULSION SUBSYSTEM (APS) CONCEPT, AS DEMONSTRATED BY RESULTS OF THIS STUDY, IS A PRACTICAL APPROACH TO SPACE SHUTTLE CONTROL AND MANEUVER REQUIREMENTS.

IN TERMS OF THRUST LEVELS AND REUSE CAPABILITY, APS REQUIREMENTS ARE FAR BEYOND THOSE FOR ANY PREVIOUS CONTROL PROPULSION SUBSYSTEM. THEREFORE, NO LOW-PRESSURE APS COMPONENTS CAPABLE OF SATISFYING THESE REQUIREMENTS EXIST TODAY. SUBSYSTEM DEFINITION STUDIES HAVE SHOWN THAT A LOW PRESSURE APS CAN POTENTIALLY FULFILL SHUTTLE REQUIREMENTS, AND THAT SUCH AN APS IS SIMPLE IN DESIGN AND OPERATIONAL APPROACH. NONE OF THE COMPONENTS REQUIRED IN THE SUBSYSTEM ARE CURRENTLY AVAILABLE, BUT EXPLORATORY PROGRAMS ARE UNDERWAY ON ENGINES, VALVES, AND CERTAIN ASPECTS OF STORAGE TANK DESIGN. IN MOST CASES, TECHNOLOGICAL ISSUES CENTER ON COMPONENT SIZE AND DYNAMIC RESPONSE. THESE CAN BE RESOLVED THROUGH NORMAL SUBSYSTEM DEVELOPMENT EVOLUTION; I.E., PROGRESSION FROM ANALYSIS TO COMPONENT AND ASSEMBLY TESTS, TO (FINALLY) BREADBOARD TESTS WITH FULL SCALE HARDWARE. A CRITIQUE OF THE MAJOR TECHNOLOGY ISSUES AND/OR CONCERNS RELATED TO A LOW PRESSURE APS ARE PRESENTED IN THE ACCOMPANYING FIGURE TOGETHER WITH AN ASSESSMENT, IN TERMS OF APS WEIGHT, OF FAILURE TO DEVELOP THE REQUIRED TECHNOLOGY.

CRITIQUE OF LOW PRESSURE APS TECHNOLOGY

TECHNOLOGY CONCERN	ALTERNATIVE APPROACH	IMPACT OF CHANGE
• HIGH PERFORMANCE INSULATION REUSABILITY AND RATE OF OUTGASSING	• VACUUM – JACKETED DEWARS	• INCREASES APS WEIGHT 198 LB AND WEIGHT OF COMBINED APS/OMS 481 LB
• PROPELLANT ACQUISITION ASSEMBLY	• USE OF SMALL REFILLABLE TANKS	• INCREASES DESIGN AND CONTROL COMPLEXITY AND REDUCES APS FLEXIBILITY. INCREASES WEIGHT APPROXIMATELY 200 LB
• HEAT EXCHANGER ATTACHMENT AND THERMAL EFFICIENCY	• WELD TUBES TO TANK STIFFENERS • INTEGRATED TANK/HEAT EXCHANGER DESIGN	• COMPLICATES TANK INTEGRATION AND REWORK POTENTIAL: REDUCES APS WEIGHT BY 210 LB • REQUIRES MAIN TANK REDESIGN; POTENTIAL APS WEIGHT REDUCTION OF 456 LB
• MAIN TANK RESIDUAL PROPELLANT UTILIZATION	• INCREASE APS PROPELLANT STORAGE	• INCREASES APS WEIGHT 15 LB PER PERCENT REDUCTION IN APS RESIDUAL UTILIZATION
• ENGINE INJECTOR PRESSURE DROP (STABILITY AND MIXING EFFECTIVENESS)	• INCREASE ENGINE PRESSURE BUDGET BY REDUCING LINE PRESSURE DROP	• INCREASES APS WEIGHT 240 LB FOR TWOFOLD INCREASE IN ΔP
• CONTROL COMPONENT LIFE CAPABILITY, I.E., VALVES AND REGULATORS	• PERIODIC REPLACEMENT	• INCREASES MAINTENANCE/TURN-AROUND TIME

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COMPARISON OF HIGH AND LOW PRESSURE APS

THE APS STUDIES CONDUCTED FOR MSFC AND MSC USED SLIGHTLY DIFFERENT GROUND RULES AS REGARDS MANEUVERING REQUIREMENTS. THE CONTROL ACCELERATIONS IMPOSED FOR THE LOW PRESSURE APS DESIGN WERE SLIGHTLY MORE STRINGENT THAN THOSE FOR THE HIGH PRESSURE APS AS IN THE LOW PRESSURE STUDY THE APS WAS REQUIRED TO SATISFY "NOMINAL" ACCELERATION REQUIREMENTS WITH ONE ENGINE FAILURE. THE EFFECT OF THIS WAS TO SLIGHTLY INCREASE LOW PRESSURE APS WEIGHT. TO COMPARE THE SUBSYSTEMS ON THE SAME BASIS THE LOW PRESSURE APS WEIGHTS WERE ADJUSTED AND THESE ARE PROVIDED IN THE ACCOMPANYING FIGURE WHICH RELATES THE WEIGHT OF THE TWO APS CONCEPTS.

APS WEIGHT COMPARISON
HIGH VS LOW PRESSURE

(WEIGHTS ADJUSTED TO COMMON SET "HIGH PRESSURE REQUIREMENTS")

	APS WEIGHTS-LBS	
	HI PC	LO PC
BOOSTER	5310	4980
ORBITER B	35880	12549 (APS) 24703 (OMS) 37252

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"ACOUSTIC CAVITY USE FOR CONTROL OF COMBUSTION INSTABILITY"

C. L. OBERG

T. L. WONG

W. M. FORD

ROCKETDYNE

TECHNICAL MANAGER

W. BRASHER

MANNED SPACECRAFT CENTER

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TITLE: ACOUSTIC CAVITIES FOR CONTROL OF COMBUSTION INSTABILITY

CONTRACT: NAS9 - 9866

PRESENTER: C. L. OBERG, T. L. WONG, AND W. M. FORD

COMPANY: ROCKETDYNE, DIV. OF NORTH AMERICAN ROCKWELL

NASA PROJECT MANAGER: W. L. BRASHER

Rocketdyne
North American Rockwell

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Studies have been underway at Rocketdyne to develop criteria for the design of acoustic cavities to suppress acoustic modes of combustion instability. Most of this work has been sponsored by NASA Manned Spacecraft Center under Contract NAS9-7498 CCA 29 (Lunar Module Ascent Engine) and NAS9-9866 (Evaluation of Acoustic Cavities). All of the hot firings done under these contracts have been made with LM ascent engine-type hardware, propellants and operating conditions. Nonetheless, the developed techniques are generally of equal applicability to Space Shuttle engines. However, some additional work is needed to define and allow for differences.

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347-710
3-71

ACOUSTIC CAVITIES FOR CONTROL OF COMBUSTION INSTABILITY

PRIMARY OBJECTIVE: DEVELOP SUFFICIENT UNDERSTANDING AND THE NECESSARY DATA CONCERNING CAVITY EFFECTS SO THAT THESE CAVITIES CAN BE EFFECTIVELY DESIGNED

APPLICABILITY TO SPACE SHUTTLE ENGINES: GENERAL TECHNIQUE IS DIRECTLY APPLICABLE BUT SOME SPECIFIC DATA ARE NEEDED

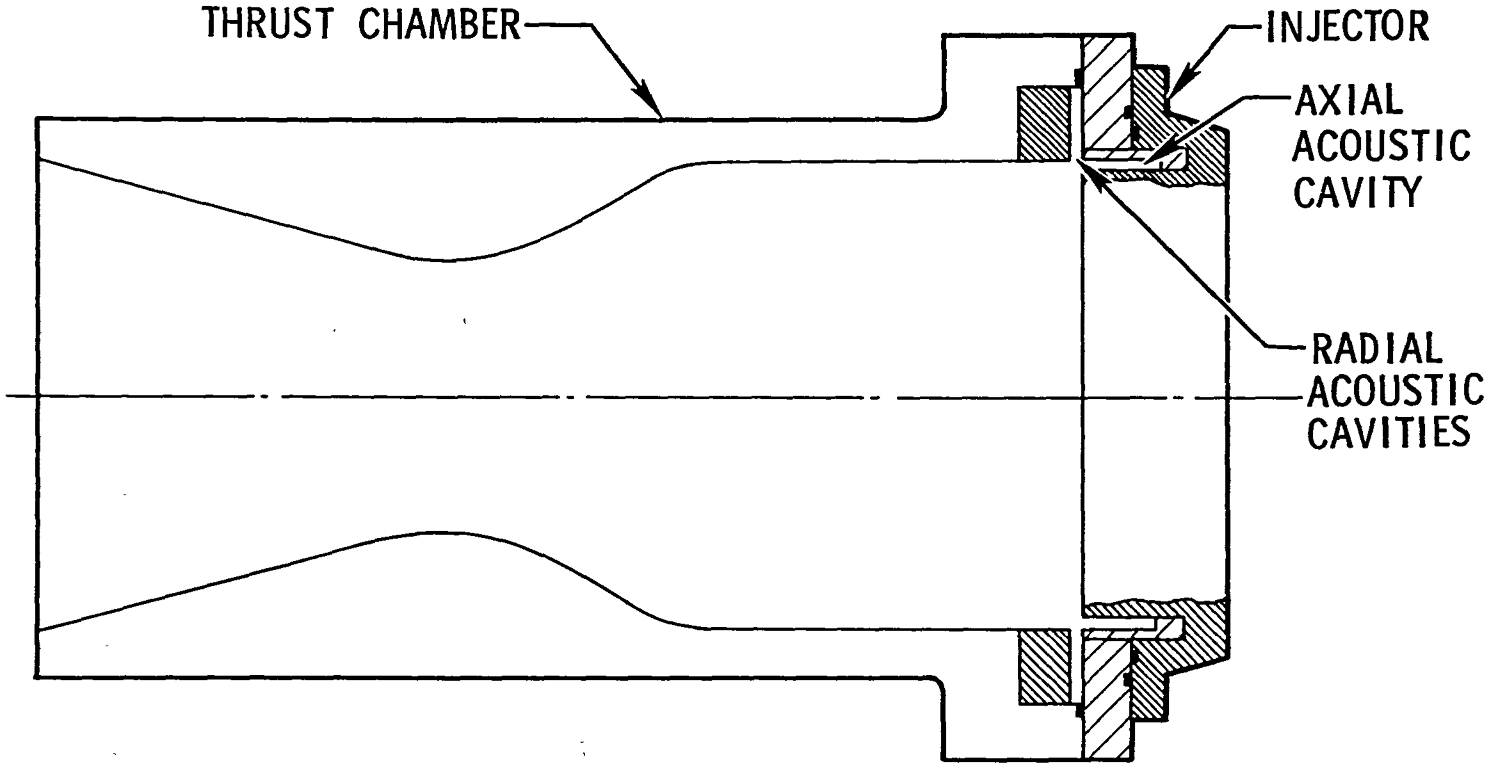
SPONSOR: NASA MANNED SPACECRAFT CENTER

TECHNICAL MONITOR: W.L. BRASHER

Acoustic modes of combustion instability in a rocket engine can be prevented through the use of acoustic absorbers. These absorbers are comprised of acoustic resonators distributed in some manner along the interior walls of a thrust chamber. One such arrangement, which is particularly attractive from a design and manufacturing standpoint, is a single row of acoustic resonators along the periphery of the injector. The term "acoustic cavity" has been loosely used to describe this simple arrangement of resonators, generally quarter wave resonators. A typical acoustic cavity arrangement is shown schematically in this slide. These cavities are simply narrow slots, either axially or radially divided, with uniform cross-sectional area. The slots are partitioned to prevent circumferential flow of hot gases. However, interest is not restricted to these simple slots. Any form of acoustic resonator should be useful, although some forms may be more effective than others.

347-554
4-70

TYPICAL CAVITY CONFIGURATION



- AXIALLY AND/OR RADIALLY DIRECTED
- PARTITIONED TO PREVENT CIRCUMFERENTIAL HOT GAS FLOW

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Results from these programs indicate that cavities promote stability by increasing the damping or rate of energy dissipation in the engine. The "wave cancellation" effect, which is often associated with quarterwave tubes, is not believed to be an important factor. The effect of a quarterwave tube in a feed line or wave guide is to reflect wave energy and prevent transmission beyond the quarterwave tube. The corresponding effect in the engine with a quarterwave resonator is an adjustment in the allowed acoustic modes of the chamber; this adjustment is accompanied by only small changes in the modal frequencies ($\sim$5 percent). Thus, the wave energy is not prevented from entering the chamber; rather, the acoustic energy is contained within the chamber as before, but with some adjustments in the wave motion.

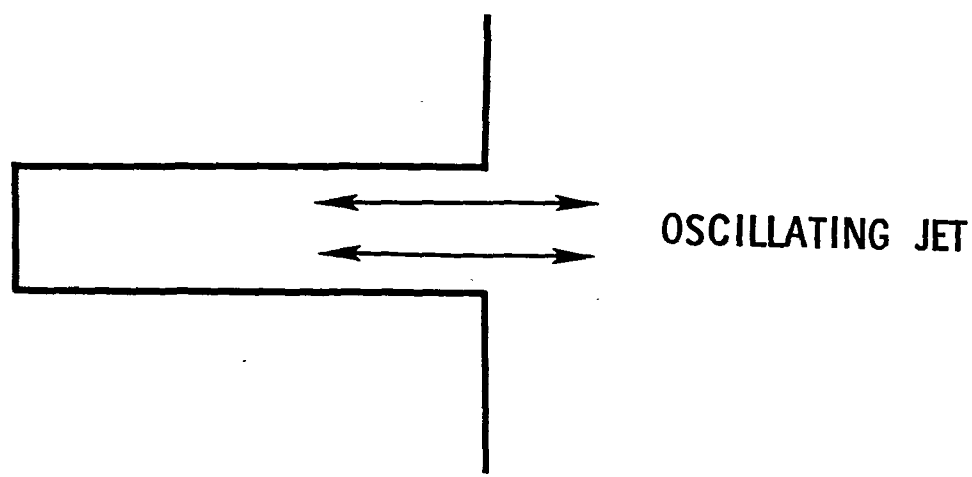
In addition, the damping contributed by the acoustic cavities is principally due to nonlinear (high amplitude) processes, the linear viscous and thermal losses being negligible relative to these nonlinear losses. This nonlinear effect is similar to that observed with Helmholtz resonators and is attributed to disintegration of an oscillatory jet formed at the cavity exit.

238-176
4-69

STABILIZING EFFECT OF CAVITIES

- PRINCIPAL STABILIZING EFFECT APPEARS TO BE DEGRADATION OF OSCILLATORY ENERGY, I.E., DAMPING
- "WAVE CANCELLATION" NOT A STABILITY FACTOR
- DAMPING DUE TO NONLINEAR (HIGH AMPLITUDE) PROCESSES; VISCOUS LOSSES NEGLIGIBLE

1009



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6633 Canoga Ave Canoga Park Calif 91304

The second investigation, which is now substantially complete, was based heavily on results from, and is largely a continuation of a previous study performed as part of the LM Ascent Engine program. During the LM study substantial progress was made concerning the influence of cavities on stability and how to design them. Further, the stabilization effectiveness of acoustic cavities was demonstrated. An analytical model was developed to predict the damping provided by a cavity. Also, subscale tests were made with a combustion-driven oscillator, a T-burner, to measure the high-amplitude acoustic characteristics of the cavity. In addition, a number of full-scale hot firings were made with LM-ascent-engine-type hardware and with an unbaffled injector. During this previous program, dynamic stability was obtained with five different cavity arrangements while instability was easily triggered without the cavity.

EXPERIENCE UNDER LM ASCENT ENGINE CONTRACT

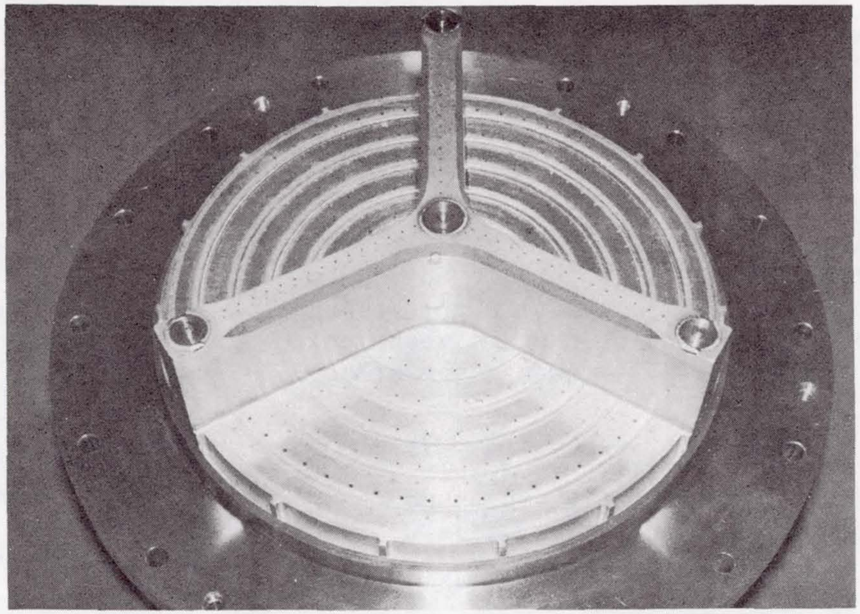
- SUBSTANTIAL PROGRESS WAS MADE - IT FORMS THE BASIS FOR THE RECENTLY COMPLETED PROGRAM
- AN ANALYTICAL MODEL WAS DEVELOPED FOR CAVITY DAMPING
- SUBSCALE (T-BURNER) TESTS WERE MADE TO MEASURE CAVITY CHARACTERISTICS
- FULL SCALE TESTS WERE MADE TO MEASURE CAVITY EFFECTIVENESS
- CAVITY EFFECTIVENESS (STABILIZATION) WAS DEMONSTRATED
- UNBAFFLED FLIGHT-DESIGN INJECTOR WAS STABILIZED WITH CAVITIES ALONE

One notable application of acoustic cavities is the current flight configuration LM ascent engine. This engine has been made dynamically stable through the combined use of baffles and acoustic cavities. The acoustic cavity used in this engine may be seen in this slide, which shows an injector from that engine. The acoustic cavity is partially exposed because the remaining portion of the cavity is formed by the thrust-chamber wall. The three-bladed baffle is used to suppress the first- and second-tangential modes of instability while the acoustic cavity is used to suppress the first-radial and third-tangential modes.

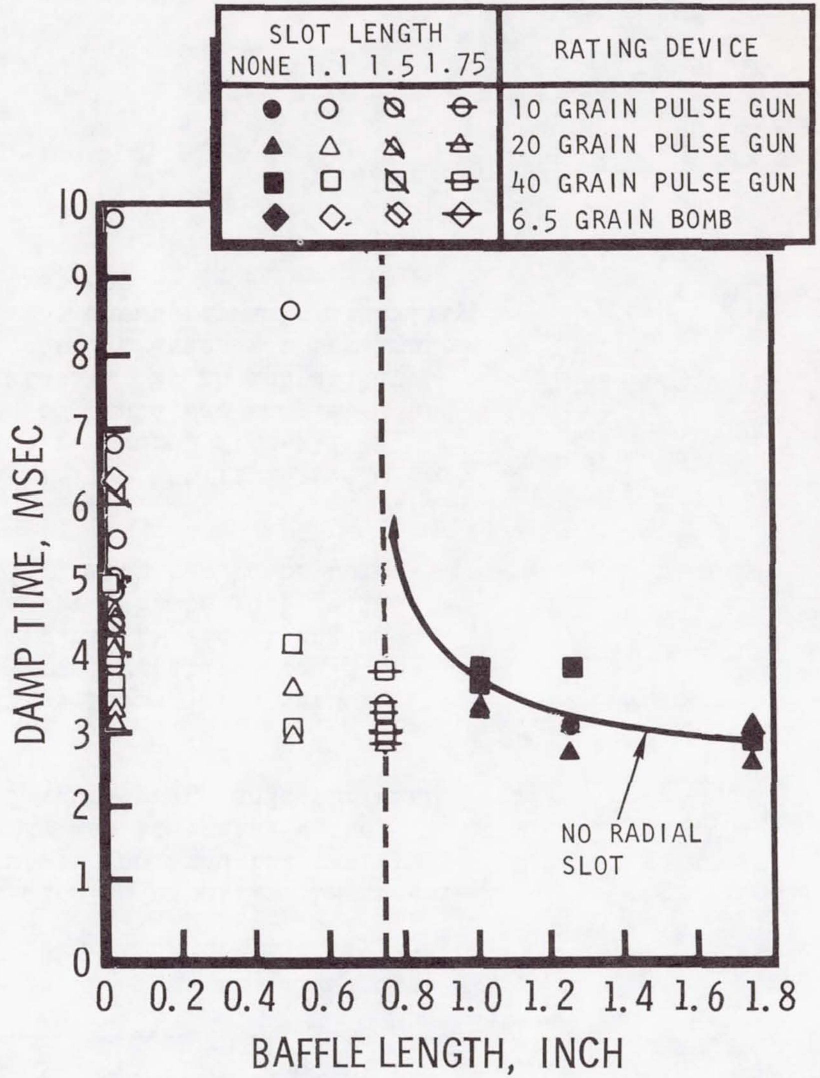
During the LM acoustic cavity study, a series of full-scale motor firings was made to survey the stabilizing effect of cavities with a wide range of cavity dimensions. Available LM ascent engine injectors, as shown in this slide, were modified for this purpose and a solid-wall thrust chamber was built to the internal dimensions of the ascent engine. The engine was operated at nominal ascent engine conditions using N_2O_4/N_2H_4-UDMH (50-50). The engine was stability rated with both bombs and pulse guns.

Initial tests were made with plugged injection elements in the baffle so that the baffle could be repeatedly shortened. Stability results from tests with several baffle lengths are shown in this slide. The observed damp times were very short until the baffle length was reduced to 0.75 inch, at which point the engine was easily driven unstable by any of the pulse guns. With an acoustic cavity, stability was regained, even when the baffle was completely removed.

EFFECT OF BAFFLE LENGTH AND CAVITY LENGTH ON DAMP TIME



LUNAR MODULE ASCENT ENGINE
PROGRAM INJECTOR



A second series of hot firings was made with an unbaffled LM ascent-type injector designed for stabilization with an acoustic cavity. The purpose of this portion of the program was to evaluate the potential for stabilizing an unbaffled flight-design injector with cavities only.

The injector was designed with the cavities formed by a set of replaceable rings so that several cavity configurations could be tested. The injector was tested under nominal LM ascent engine operating conditions in a solid-wall chamber. Bombs (6.5 grains) were used exclusively for stability rating, with two bombs being used on each testing firing.

For each cavity configuration, a series of five hot firings (10 bomb disturbances) was planned. If the engine recovered from all of these bomb disturbances, the full series of tests was completed and the engine was regarded as dynamically stable. If an instability was encountered, then testing with that configuration was terminated. Stability results are shown in Table 1. As shown, dynamic stability was obtained with several cavity configurations.

LM ASCENT ENGINE ACOUSTIC CAVITY PROGRAM

252-515A
4-70

FLIGHT-DESIGN INJECTOR TEST RESULTS

TEST SEQ.	CAVITY CONFIGURATION			NUMBER OF BOMBS	MAXIMUM DAMP TIME MILLISEC	RATING
	CAVITY ORIENTATION	WIDTH IN.	DEPTH IN.			
1	DUAL RADIAL	.20 + .10	1.10	9	*	MARGINAL
2	RADIAL	.30	1.10	10	32.7	MARGINAL
3	RADIAL (LOCATED .45 INCH DOWNSTREAM)	.30	1.10	1	**	UNSTABLE
4	RADIAL	.20	1.30	1	**	UNSTABLE
5	NONE	0	0	1	**	UNSTABLE
6	AXIAL	.30	1.30	10	5.7	STABLE
7	AXIAL	.20 + .10	1.10	10	***	MARGINAL
8	AXIAL	.20	1.30	1	**	UNSTABLE
9	AXIAL	.30	1.50	10	6.2	STABLE
10	AXIAL	.35	1.30	19	7.6	STABLE
11	AXIAL	.20	1.75	1	**	UNSTABLE
12	AXIAL	.35	1.10	10	7.2	STABLE
13	AXIAL	.20	1.50	1	**	UNSTABLE
14	AXIAL + RADIAL	.20 + .20	1.30	10	6.9	STABLE

NOTE: * EIGHTH BOMB TRIGGERED SUSTAINED INSTABILITY
** FIRST BOMB TRIGGERED SUSTAINED INSTABILITY
*** TENTH BOMB TRIGGERED SUSTAINED INSTABILITY

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During the current program which was recently completed, several areas of investigation have been pursued: Increased analytical capability was developed by removing or improving some of the previously used ap proximations and allowing for additional effects and configurations. Test firings were made to experimentally define an optimum slot width; modifications have been made to the analytical model to improve the ability to predict the optimum slot width. Also, test firings have been made to investigate the effects of cavity location and of cavity multiplicity (multiple rows of resonators) on stability. Detailed cavity temperature measurements have been made. Analytical and sub-scale experimental studies of unconventional cavity configurations (which emphasize hardware constraints) have been made; full-scale stability tests with these cavities have been made. Full-scale tests were also made to investigate the effect of varying the film coolant flowrate and at nominal conditions on cavity stabilization.

347-557A
3-71

CURRENT PROGRAM
AREAS OF INVESTIGATION

- CAVITY EFFECTIVENESS
- CAVITY LOCATION AND MULTIPLICITY
- IMPROVED ANALYTICAL CAPABILITY
- OPTIMUM SLOT WIDTH - ANALYTICAL AND EXPERIMENTAL
- CAVITY TEMPERATURES
- UNCONVENTIONAL CAVITIES (EMPHASIZE HARDWARE CONSTRAINTS)
- FILM COOLING

1017

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An analytical model was developed to describe the stabilizing effect of the cavity was based on the calculation of a temporal damping coefficient contributed by the absorber. This parameter is believed to most strongly reflect the stabilizing effect of the cavities. The damping coefficient was chosen because, if a system of equations describing the low-amplitude (linear) wave motion in a combustion chamber including the effects of combustion was solved, a time dependence would be obtained for each mode of the form shown in this slide.

The overall damping coefficient is α_N and the oscillatory frequency is $\omega_N/2\pi$. If α_N is negative, the system is unstable and the amplitude grows exponentially. Further, α_N is approximately the sum of a series of contributions as shown, where driving processes contribute negative α_{N_i}'s and damping processes contribute positive values. The utility of this approach is that a good estimate of particular contributions to this sum, e.g., a contribution due to an acoustic cavity, is obtained from an analysis which neglects other gains and losses. Consequently, the analytical model was based on the calculation of a damping coefficient contribution, an α_{N_j}, due to the cavity, while ignoring other gain and loss processes.[3] These other processes are generally, too poorly understood to justify attempts to calculate the remaining α_{N_i}'s. Therefore, it is assumed that the overall α_N will be positive and the engine stable if the contribution due to the cavities is made "large enough." The damping coefficient due to the acoustic cavities is believed to be the best available measure of the stabilizing effect of the cavities even under high amplitude (non-linear) conditions.

The damping coefficient may be suitably estimated by a calculation that neglects other gain and loss effects, an approach which is widely used in studies of solid-propellant instability. Toward this end, it is appropriate to solve the wave equation for a chamber of the size and shape of the combustion chamber. This solution is complicated somewhat by the fact that the boundary condition, the wall impedance, is nonuniform; therefore, the equation cannot be solved by the usual separation of variables. A suitable alternate technique is to convert the wave equation and boundary conditions to an integral equation and obtain an approximate solution to the latter equation by a variational technique.

ANALYTICAL APPROACH

- EMPLOY METHODS OF THEORETICAL ACOUSTICS TO ESTIMATE DAMPING (DAMPING COEFFICIENT) PRODUCED BY THE CAVITY

- RATIONALE

$$\alpha_N \simeq \sum_i \alpha_{Ni}$$

WHERE

$$\tilde{p}_N = F_N (r, \theta, Z)\, e^{-\alpha_N t} \cos \omega_N t$$

- ESTIMATE α_{Nj} DUE TO CAVITIES WHILE NEGLECTING COMBUSTION

- THEREFORE, EMPLOY QUASI-LINEAR MODEL TO CALCULATE DAMPING CONTRIBUTED BY CAVITIES, α_{Nj}

1019

Some of the equations which comprise the model are shown in this slide. As shown the equations pertain to an axially directed acoustic cavity in the injector face.

ANALYTICAL MODEL FOR CAVITY DAMPING

- WAVE EQUATION

$$\nabla^2 p = \frac{1}{c^2}\frac{\partial^2 p}{\partial t^2}$$

- BOUNDARY CONDITIONS

$$\frac{\vec{N}\cdot\nabla p}{p} = \begin{cases} -j\,\dfrac{\beta}{c\,\zeta} & \text{AT ABSORBER} \\ 0 & \text{ELSEWHERE} \end{cases}$$

- INTEGRAL EQUATION

$$p(\vec{r}) = \int_s G(\vec{r}\,|\,\vec{r}_0^{\,s})\ \vec{N}\cdot\nabla_0 p(\vec{r}_0^{\,s})\ dS_0$$

- CHARACTERISTIC EQUATION

$$\zeta(\phi) = j\,\frac{\phi}{r_w^2}\sum_n \frac{2\alpha_{\bar{m}n}^2}{\alpha_{\bar{m}n}^2 - \bar{m}^2}\ \frac{\left\{\int_{r_s} r\,J_{\bar{m}}\left(\frac{\alpha_{\bar{m}n} r}{r_w}\right) J_{\bar{m}}\left(\frac{\alpha_{\bar{m}n} r}{r_w}\right) dr\right\}^2}{J_{\bar{m}}^2\left(\alpha_{\bar{m}n}\right)\int_{r_s} r\,J_{\bar{m}}^2\left(\frac{\alpha_{\bar{m}n} r}{r_w}\right) dr}\ \frac{\cot\sqrt{\phi^2 - \alpha_{\bar{m}n}^2}\left(\frac{L}{r_w}\right)}{\sqrt{\phi^2 - \alpha_{\bar{m}n}^2}}$$

- NONDIMENSIONAL FREQUENCY

$$\phi = \frac{\omega r_w}{c} + j\,\frac{\alpha r_w}{c}$$

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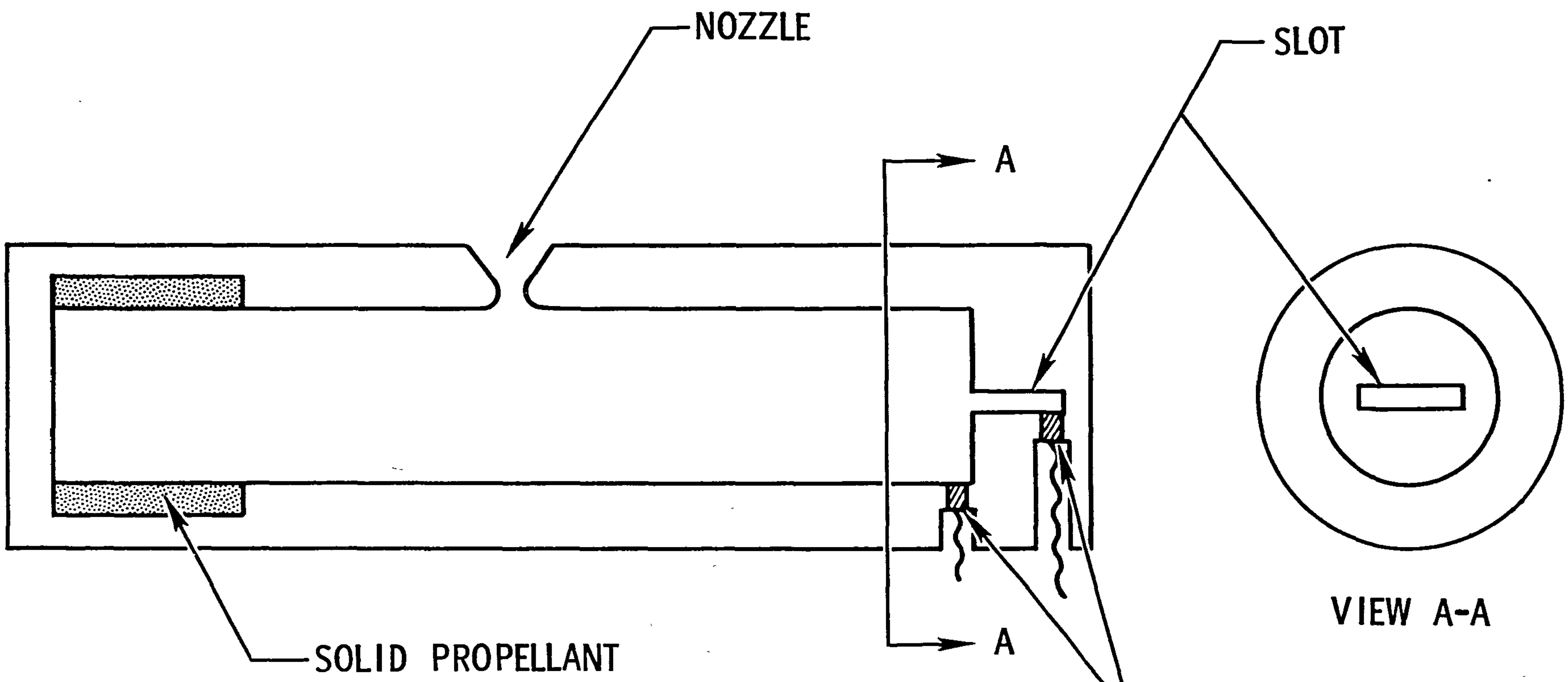
Subscale tests were made during both programs to measure the nonlinear acoustic impedance of the cavities described previously. A small combustion-driven acoustic oscillator known as a T-burner was used to generate a high-amplitude environment for the acoustic cavity. The T-burner is shown schematically in this slide. Oscillatory pressures were measured at the open and closed ends of the acoustic cavity with Kistler high-frequency transducers, as shown. The amplitude ratio and phase angle between these two pressures are related by the impedance of the cavity. Measured values of amplitude ratio and phase angle were used to calculate the cavity impedance.

SUBSCALE TESTS



T-BURNER

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Frequently, a simple, well-defined acoustic resonator of either the Helmholtz or quarterwave type is not practical in an engine because of hardware or spatial limitations. The hardware characteristics often suggest resonator configurations which do not clearly fit inot either category. Further, frequently neither of these resonator types represents the optimum configuration in terms of maximum damping and/or bandwidth attainable within a specified volume. Consequently, some work has been done to extend the range of resonator configurations which can be described analytically. Further, some experimental work has been done and more is planned to characterize these unconventional resonator configurations. Some of the configurations under consideration are shown in this slide.

Acoustic impedance expressions have been developed for each of these configurations by employing the same approach used for the slot. The intermediate resonator is so termed because it is intermediate between a Helmholtz resonator and a quarterwave resonator. The intermediate, "L"-shaped Helmholtz and quarter-wave resonators can all be considered special cases of the generalized resonators.

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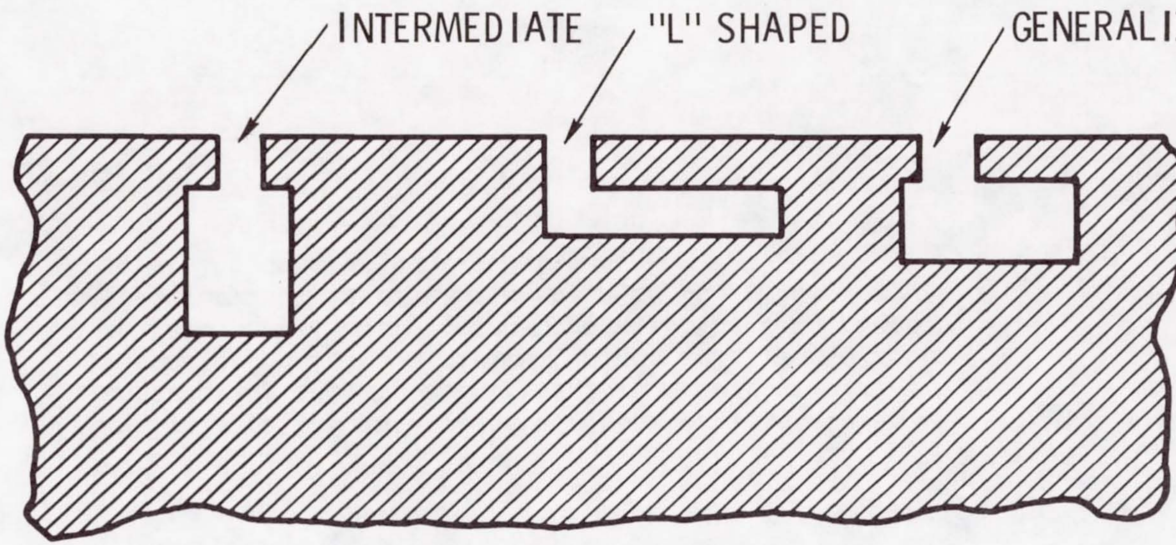
UNCONVENTIONAL CAVITIES



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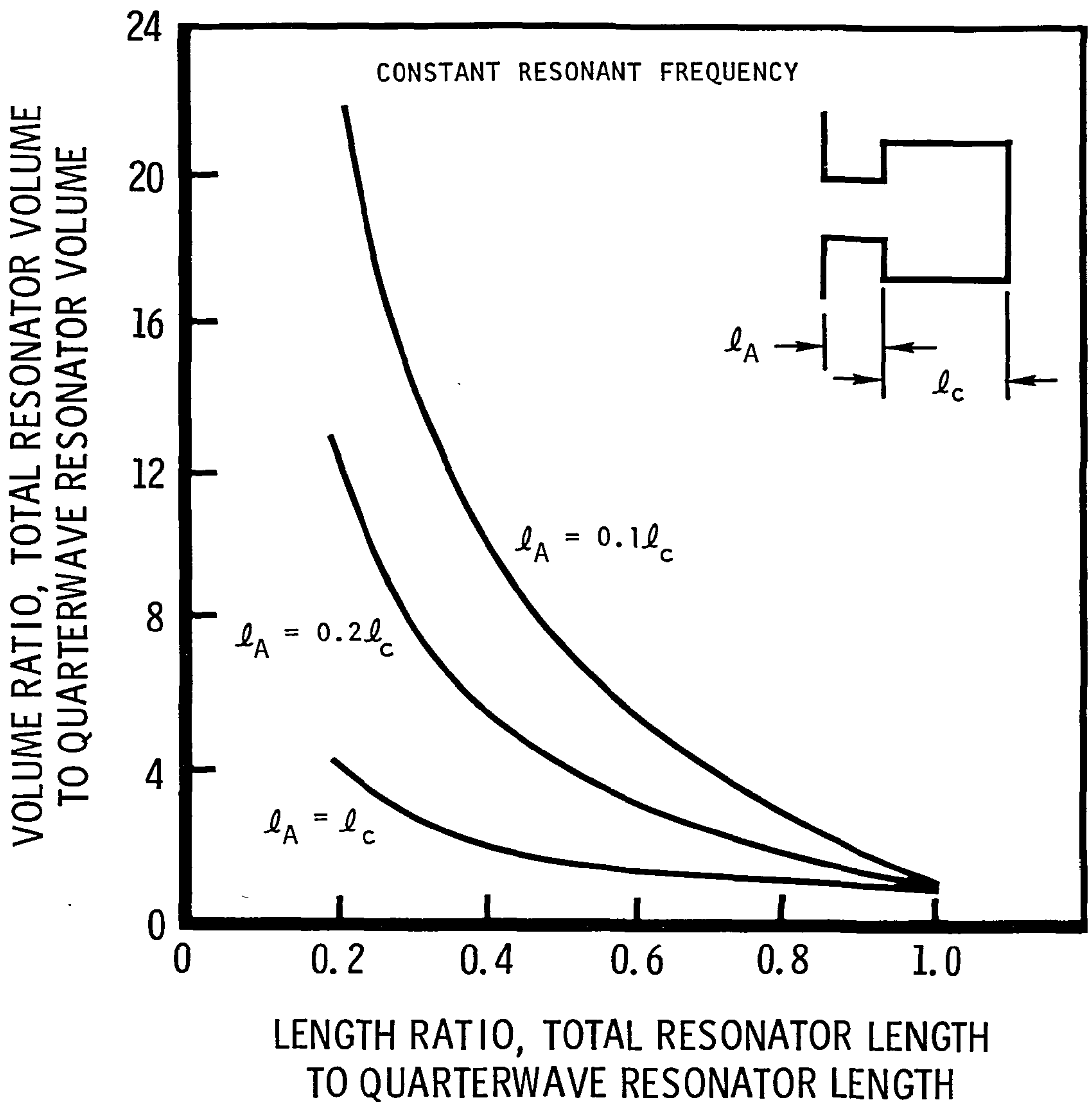
Analysis of the intermediate resonator leads to an interesting and important result: a quarterwave resonator occupies a minimum volume relative to intermediate or Helmholtz resonators with the same resonant frequency. This result is shown in the slide for several resonator configurations.

COMPARATIVE RESONATOR VOLUME REQUIREMENTS



Full scale stability rating tests have been made to evaluate the effects of several cavity-related parameters on stability and, conversely, to evaluate the influence of other parameters on stabilization with acoustic cavities. Four different series of tests were performed; these encompassed 125 hot firings.

The first series of tests (Phase I - Full Scale Test Series) was made to measure the influence of cavity (1) width, (2) axial position and (3) multiplicity on engine dynamic stability. The second series (Off-Nominal Test Series) was made to measure the influence of engine operating conditions, i.e., mixture ratio and chamber pressure on cavity stabilization. The third test series (Film Cooling Test Series) was made to measure the influence of film coolant flow rate on cavity stabilization. Finally, the fourth series of tests (Unconventional Cavity Test Series) was made to evaluate the effectiveness (ability to promote stability) of unconventional acoustic cavities. In addition, detailed cavity temperature measurements were made on all tests.

FULL SCALE TESTING

- PHASE I - DEVELOP DESIGN CRITERIA
  - SLOT WIDTH EFFECT
  - AXIAL POSITION EFFECT
  - MULTIPLE SLOTS
- PHASE II - UNCONVENTIONAL CAVITY TESTING
  - EVALUATE EFFECTIVENESS OF UNCONVENTIONAL CAVITIES
- PHASE III - OFF NOMINAL AND FILM COOLING TESTS
  - SENSITIVITY OF CAVITY STABILIZATION TO VARIATIONS IN P_C AND MIXTURE RATIO
  - EFFECTS OF VARIATIONS IN FILM COOLANT ON CAVITY STABILIZATION

1030

The hardware used for these tests was similar to the LM-Ascent engine but with an unbaffled injector being used. However, the feed systems were not similar. The injector is the same as that used previously for the LM cavity testing and includes 214 unlike doublet (pairs) elements and 30 equally spaced, axially directed, film coolant orifices. The injector was designed for nominal operation with N_2O_4/50% N_2H_4 - 50% UDMH propellants at a mixture ratio of 1.60 and a chamber pressure of 122 psia. The injector is shown in this slide.

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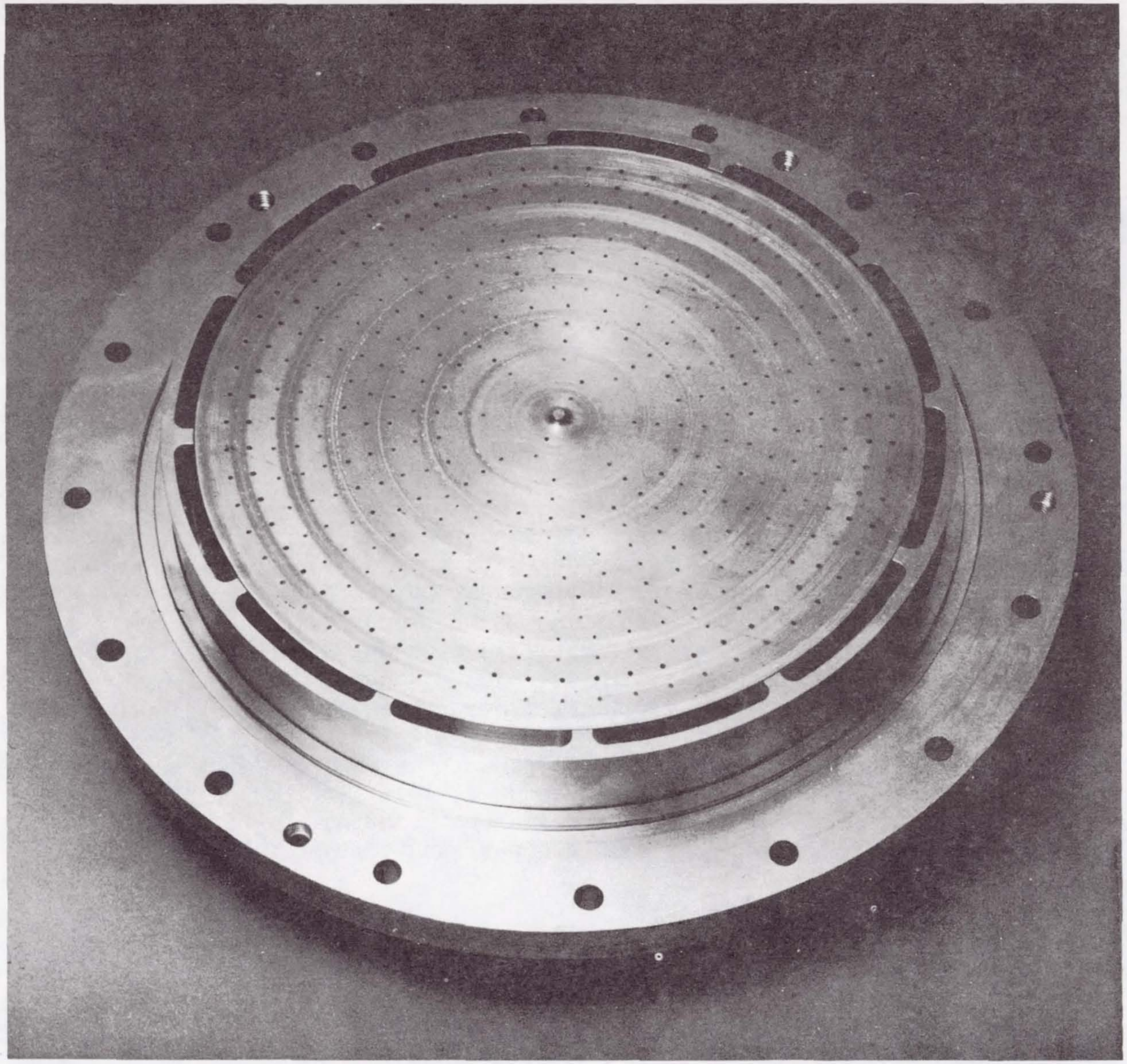

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ACOUSTIC CAVITY PROGRAM

BAFFLELESS INJECTOR WITH AXIAL CAVITY

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The thrust chamber used for these tests was uncooled and designed to accommodate replaceable rings which contained the acoustic cavities to be tested. The chamber is shown in this slide. The chamber dimensions are equivalent to those of the ascent engine, i.e., 7.79 inch diameter, contraction ratio of 2.90 and $L^* = 37.6$ inches. Stability rating was accomplished by disturbing the combustion process with 6.5 grain (RDX) bombs, with two bombs being used for each hot firing.

Detailed measurements of temperatures within the acoustic cavity were made with tungsten/tungsten-rhenium thermocouples located at several depth positions in the cavity.

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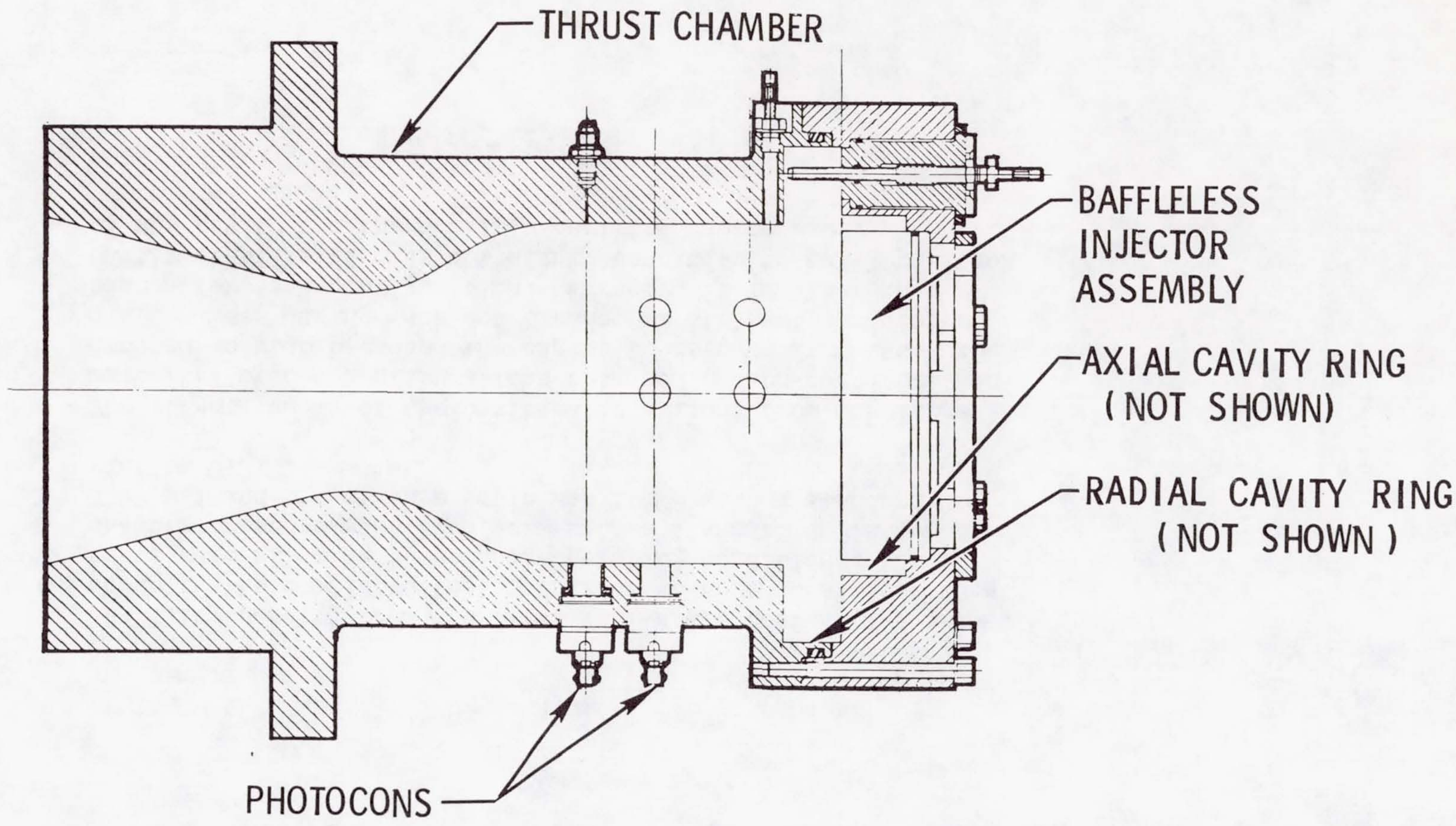
FULL SCALE THRUST CHAMBER



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Stability results from the first full-scale test series are shown in the completed test matrix shown in this slide. These tests were all made at nominal operating conditions; the chamber pressure was 122.5 psia with a standard deviation of 0.97 psi and the mixture ratio was 1.608 with a standard deviation 0.013.

The effectiveness of the cavities is evident from the large number of stable configurations obtained. Each configuration denoted as stable caused the engine to recover from two bomb disturbances during each hot firing, or at least six disturbances for each configuration. Further, it is clear that cavity stabilization is not highly sensitive to axial position of the cavity and the multiple cavities can be used.

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COMPLETED TEST MATRIX FOR PHASE I FULL SCALE FIRINGS

TEST OBJECTIVE	CAVITY CONFIGURATION NUMBER	SLOT WIDTH, INCH	AXIAL POSITION** OF RADIAL SLOT, INCH			AXIAL SLOT	SLOT LENGTH, INCH			PLANNED NUMBER* OF FIRINGS	ACTUAL NUMBER* OF FIRINGS	STABILITY RESULTS
			0	0.5	1.0		1.3	1.5	1.8			
REFERENCE	1	0								3	3	UNSTABLE
	2	0.2				X	X			3	3	NOT VALID
	3	0.3				X	X			3	3	NOT VALID
WIDTH EFFECT	4	0.3	X				X			3	4	STABLE
	5	0.6	X				X			3	4	
	6	1.0	X				X			3	6	
AXIAL POSITION EFFECT	7	0.4		X				X		3	6	
	8	0.6		X				X		3	3	
	9	0.3			X			X		3	3	UNSTABLE
	10	0.4			X			X		3	3	STABLE
	11	0.6			X			X		3	3	
	12	0.4		X			X			3	3	
	13	0.4		X					X	3	6	
MULTIPLE CAVITIES	14	0.2, 0.2			X	X	X			3	3	
	15	0.2, 0.2, 0.2		X	X	X	X			3	3	
TOTAL										44	56	

*TWO BOMB DISTURBANCES FOR EACH FIRING
**EXCEPT FOR ZERO POSITION, AXIAL POSITION REFERENCES TO DISTANCE OF SLOT MID-WIDTH FROM INJECTOR FACE

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The influence of operating conditions on cavity stabilization was investigated during the off nominal test series. Fourteen tests were made with a 0.3" x 1.3" axially directed cavity. Chamber pressure and mixture ratio were varied from nominal conditions, as indicated by the completed test matrix shown in this slide. The test results exhibit a gradual and consistent variation of stability with mixture ratio and chamber pressure. These results probably reflect simply a variation in the instability driving processes rather than a cavity effect. The stabilization ability of a cavity does not appear highly sensitive to engine operating conditions.

The influence of the film coolant flow rate on cavity stabilization was also investigated. These tests were made, in part, because detrimental effects of this cooling on stability have been observed with acoustic liners. Twenty-two full-scale tests were made with two different cavity configurations at each film-coolant flow-rate. Film-coolant flow-rates of 50, 100 and 200 percent of nominal were employed, where the film coolant flow-rate used on previous tests was regarded as 50 percent rather than 100 percent of nominal. A test matrix for the completed tests is shown in this slide. The stability results indicate little influence, if any, of film coolant flowrate over the range investigated. The cavity temperatures were affected to some degree but a slight change of stability with the 0.3-inch wide cavity appears compatible with the temperature change. The observed weak dependence of cavity stabilization on film coolant flowrate removes concern for a degradation of stability accompanying increased film coolant flowrate.

Also shown is a completed test matrix from a series of tests to evaluate the stabilization ability of four different unconventional cavity configurations (two "L" shaped and two intermediate configurations). All four produced dynamic stability. This demonstrated effectiveness allows greater design flexibility to be introduced and more efficient utilization of available space. Moreover, the unconventional cavities were designed for two widely separated cavity temperatures and, thereby, resonant frequencies. Nevertheless, all of them stabilized the engine, a result which suggests a substantial stability margin.

COMPLETED TEST MATRICES

OFF-NOMINAL FULL-SCALE TEST SERIES

TEST OBJECTIVES	MIXTURE RATIO			CHAMBER PRESSURE, psia			NUMBER* OF FIRINGS	STABILITY
	1.4	1.6	1.8	90	122	200		
STUDY OFF-NOMINAL EFFECTS		X			X		3	M.S.**
		X		X			3	STABLE
		X				X	3	UNSTABLE
	X				X		3	M.S.
			X		X		2	STABLE
							14	

FILM-COOLING FULL-SCALE TEST SERIES

TEST OBJECTIVES	CAVITY CONFIGURATION		COOLANT FLOWRATE, PERCENT OF NOMINAL			STABILITY
	0.3 x 1.3 AXIAL	0.4 x 1.5 RADIAL	50	100	200	
FLOWRATE	X		X			M.S.**
	X			X		M.S.
	X				X	M.S.
		X	X			STABLE
		X		X		STABLE
		X			X	STABLE

UNCONVENTIONAL-CAVITY FULL-SCALE TEST SERIES

TEST OBJECTIVE	APERTURE		CAVITY			STABILITY
	WIDTH (IN.)	LENGTH (IN.)	WIDTH (IN.)		LENGTH (IN.)	
	0.3	0.2	0.9	1.3	0.7	
EVALUATE INTERMEDIATE CAVITY	X	X	X		X	STABLE
	X	X		X	X	STABLE
	0.4	0.5	0.9	1.3	0.4	
EVALUATE "L" SHAPED CAVITY	X	X	X		X	STABLE
	X	X		X	X	STABLE

* TWO PLANNED BOMBS FOR EACH FIRING; HOWEVER, THE SECOND BOMB WAS NOT FIRED IF THE FIRST INITIATED AN INSTABILITY

** M.S. = MARGINALLY STABLE

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This slide shows a graphical summary of the stability results from the full-scale cavity tests at nominal operating conditions. It is clear that a relatively large region of stable operation, in terms of cavity depth and width, exists for this engine but the boundaries of that region have not yet been defined, except for the smaller cavity width. It is likely that separate plots should be made for each axial position rather than combined as shown here. The indicated size of this stable region suggests the ability to stabilize more unstable engines than the one tested, which was, nonetheless, quite unstable without the cavity.

Also shown on this plot are predicted curves of constant damping coefficient. The curves were calculated from the radial slot model with an assumed average cavity temperature of 900 F. The curves agree quite well with the measured results, but this average cavity temperature is somewhat lower than measured.

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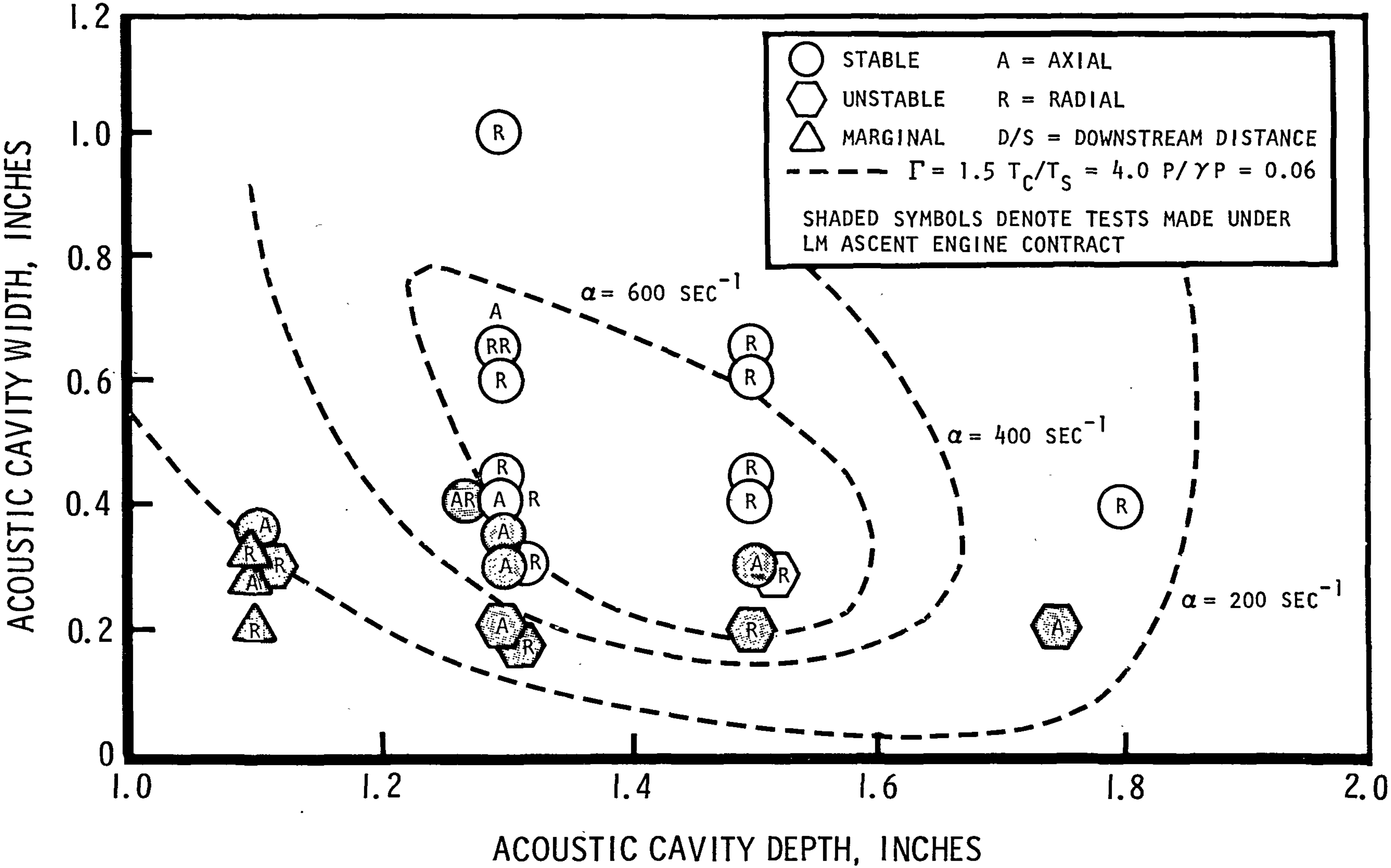
COMPARISON OF PREDICTED TRENDS WITH OBSERVED FULL SCALE STABILITY



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CONCLUSIONS

- THE EFFECTIVENESS AND UTILITY OF ACOUSTIC CAVITIES FOR SUPPRESSING ACOUSTIC MODES OF COMBUSTION INSTABILITY HAS BEEN CLEARLY DEMONSTRATED
- CAVITY STABILIZATION IS NOT HIGHLY SENSITIVE TO VARIATIONS IN ENGINE OPERATING CONDITIONS OR FILM COOLANT FLOWRATE
- THE ABILITY TO USE UNCONVENTIONAL CAVITIES HAS BEEN DEMONSTRATED
- A LARGELY SATISFACTORY ANALYTICAL MODEL FOR CAVITY DESIGN HAS BEEN DEVELOPED
- A SOUND TECHNOLOGICAL BASIS FOR CAVITY DESIGN HAS BEEN DEVELOPED

"NONCIRCULAR INJECTOR ORIFICES AND ADVANCED FABRICATION TECHNIQUES"

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TECHNICAL MANAGER

M. LAUSTEN

MANNED SPACECRAFT CENTER

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TITLE: NONCIRCULAR ORIFICE HOLES AND ADVANCED FABRICATION TECHNIQUES FOR LIQUID ROCKET INJECTORS

CONTRACT: NAS9-9528

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PRESENTER: R. M. McHALE

COMPANY: ROCKETDYNE, DIVISION OF NORTH AMERICAN ROCKWELL CORP.

NASA PROJECT MANAGER: M. LAUSTEN

INTRODUCTION

Most rocket engine injector designs in existence today employ circular orifices. Historically, circular orifices have been employed because of manufacturing limitations. With the advent of new fabrication techniques, injector orifices can now be fabricated by means other than conventional twist drilling. Noncircular orifices can be produced with relative ease. Because of these fabrication advances, it is appropriate to re-evaluate injector design practices to determine whether noncircular orifice designs can offer superior qualities either in terms of greater flexibility, lower costs, and better reproducibility, or in terms of equivalent or improved performance and injector-thrust chamber design compatibility.

Potential advantages of noncircular orifices can be seen from an examination of current problem areas associated with circular orifices. Circular orifice design is particularly sensitive to tolerances and machining practices which control such parameters as: impingement point, orifice diameter, entrance conditions, and free-stream L/D. Minor misimpingement of two circular streams (in terms of a few thousandths of an inch for a 0.030-inch-diameter orifice) causes gross distortion of spray fan shapes and distributions. In spite of the specification of very close tolerances, engine-to-engine performance and thrust chamber wall erosion variations occur which can be traced, in part, to slight misimpingement. Furthermore, the flowrate through a circular orifice is proportional to the diameter squared, and thus local flowrates are particularly sensitive to drill diameter. The diameter ratio used in the design of circular orifice unlike-impinging-stream patterns, is dictated by consideration of the propellant momentum and diameter ratio required to maximize mixing. Almost without exception, unequal sizes result. Although this condition may produce the best mixing, it is highly probably that atomization is impaired (the fan shape is also influenced).

Many of these disadvantages can be overcome utilizing noncircular orifice shapes. For instance, tolerance control relaxation can be exercised on rectangular orifices on the small side (which is more difficult to control) because flow variation is less sensitive to this single dimension. In addition, the use of noncircular orifices certainly lends itself to the possibility of equalizing contact dimension (impingement of a square on the smaller side of a rectangle, for instance).

To assess the possible advantages of noncircular orifices, an applied research program was conducted containing analysis, design, and experiment to determine whether or not noncircular orifices can provide greater flexibility and/or a significant improvement in injector designs as opposed to the conventional circular orifices.

There is an apparent dichotomy which has developed in present day technology. On the one hand, there is an increasing demand for excellence, while on the other hand, there is a demand for lower costs. This is especially true of the technology associated with the aerospace industry. It is becoming more and more obvious that yesterdays techniques and approaches to research and development efforts are not suited to the accomplishment of both these primary objectives.

In this program, the objective was to evaluate and characterize new and different rocket engine injector elements to broaden the foundation of injector design. The requirements for these new elements include such considerations as ultra-high performance, reduced fabrication costs, and increased design flexibility. If the original development technique of cut and try with full-scale hardware were selected to meet this objective, the time and money expenditures required, would prove to be intolerable.

At Rocketdyne, a new and advanced approach to rocket engine injector characterization has been developed. The major objectives of this new approach are to reduce overall costs and to provide, at the same time, greater insight into the actual mechanisms which affect injector performance. Rather than attempting to analyze a complete injector on a hot fire basis, study is initiated with single injector elements on a cold flow basis using non-reactive propellant simulants. Furthermore, the overall performance limiting processes associated with combustion in a rocket chamber are grouped into two major classes: (1) mixing processes, and (2) atomization processes.

These processes are investigated independently with cold flow modeling techniques which have been developed for each. Mixing characteristics are defined by the direct measurement of mass and mixture ratio distribution profiles employing appropriate propellant simulants. These data are used to compute a mixing uniformity parameter, E_m, and also $\eta_{c^*_{mix}}$, which is obtained by combustion model analysis of the mass and mixture ratio profile data. The atomization process is investigated with the molten wax technique wherein molten wax is injected through the element and the frozen particles collected to determine the mass median droplet diameter as well as the drop size distribution about the median size. A vaporization rate limited combustion model is employed to estimate the contribution of the vaporization process to the overall performance in the form of the vaporization limited C* efficiency $\eta_{c*_{vap}}$. The two independent limited performance estimates are then combined to estimate the overall efficiency through the first order approximation of their product, $\eta_{c*_{pred}} = \eta_{c*_{mix}} \times \eta_{c*_{vap}}$. This method produces design and analysis information pertaining to the performance of many different elements and modifications of these elements at a cost far less than that incurred in hot firing.

Following single element cold flow analysis, the usual program plan includes single element hot firing studies of those elements and their various configurations which were shown to be of interest by the cold flow tests. Single element hot fire tests provide additional information about the mixing and atomization mechanisms at a cost which is also substantially below full scale injector firings.

The final step in the research investigation is usually the design of a full scale or multi-element injector, the design having been dictated by the information obtained in the single element cold flow and hot fire programs. The full scale injector design dictated by this approach is usually quite close to the final configuration and will not require costly major redesign. The overall cost of the development program is well below that of the cut and try approach with the added advantage that detailed information is available concerning the role of operating and design variables in the performance of the injector. In other words, the injector will be high performing and the investigators will know why it is and will be able to extend their knowledge to the design of related hardware without the necessity of starting from scratch.

Information pertaining to chamber compatibility is also made available in the results of a program which contains single element cold flow studies. The mass and mixture ratio profiles offer a direct picture of the flow field which can be expected from a given element. Superposition of these pictures and the geometry of the chamber wall yields an estimate of the interaction of zones of known temperature and the wall surface. Without cold flow results, information of this nature must be obtained through direct hot fire testing.

The validity of this overall approach has been documented in the past by many programs. Two such programs of particular interest are the space storable propellant injector study, NAS3-12051, and the gas augmented injector study, NAS3-12001. In these programs, excellent agreement was obtained between hot fire test results and cold flow estimates of these results.

The overall program was divided into two phases as shown on the accompanying page: Phase I, analytical and experimental study of noncircular orifices and elements, liquid/liquid propellants, and Phase II, analytical and experimental study of noncircular orifices and elements, gas/liquid propellants

PHASE I

The first step in the liquid/liquid study of Phase I was the investigation of the hydraulic characteristics of single, noncircular orifices. To reduce the number of candidates to a reasonable value, a preliminary analysis and evaluation of orifice shapes was conducted. Several noncircular orifice shapes were selected for cold flow testing. Cold flow tests were performed on these orifices along with a circular orifice as a standard for comparison. Based upon the cold flow results, rectangular and triangular orifices were selected for study, along with circular orifices, in the context of unlike doublet injector elements.

The single element study consisted of both cold flow and hot fire evaluation. The mixing and atomization characteristics of the unlike doublets were determined through the utilization of specialized cold flow techniques. The elements were subsequently hot fired and the performance characteristics of the various elements were compared on the basis of both cold flow and hot fire results.

Unlike-doublet elements with noncircular orifices appeared to provide significant advantages in the area of liquid/liquid propellant mixing.

PHASE II

Phase II was initiated with a preliminary analysis of candidate injector elements incorporating noncircular orifices for gas/liquid applications. A conventional, circular concentric tube injector element was carried throughout the study as a reference for performance. Evaluation of the preliminary analysis lead to the selection of a noncircular (rectangular) concentric tube element for further study. Two concentric tube elements, one circular and one noncircular, were fabricated. The mixing uniformity and atomization characteristics of these elements were determined employing special, two-phase cold flow techniques. The cold flow data were employed to predict C* efficiencies for the two elements. Comparison of the elements was based upon these predicted performance values.

The noncircular concentric tube element produced significantly higher predicted combustion efficiencies than the circular element.

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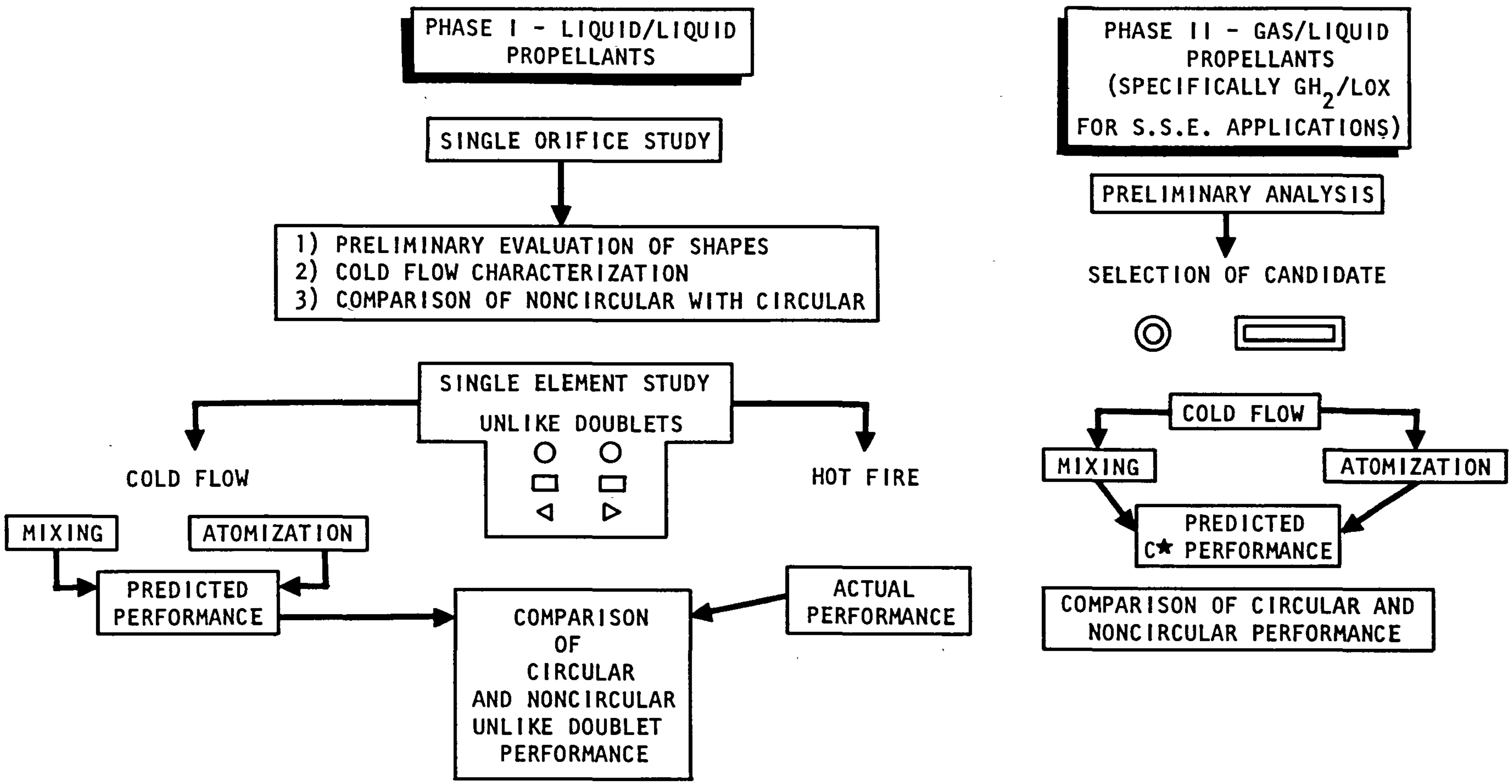
PROGRAM LOGIC DIAGRAM



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NONCIRCULAR ORIFICE PROGRAM RESULTS

PHASE I

Preliminary Analysis and Evaluation

The overall program was initiated with a preliminary analysis and evaluation of orifice shapes. This was employed as a means of reducing the number of candidate orifice shapes to a reasonable value prior to cold flow characterization studies. Existing experimental data and analytical techniques were employed in the preliminary analysis to estimate the orifice coefficients, C_D, of the various shapes along with the effects of manifold cross velocity, back pressure, Δp, and total flowrate on C_D and free jet stability. An attempt was also made in the preliminary analysis to define correlating parameters and techniques for evaluation of the cold flow data which was to be taken.

The various orifice shapes were compared on the basis of the results of the preliminary analysis and a preliminary evaluation of the shapes was conducted. Six orifice shapes were selected on the basis of their performance in the preliminary evaluation to undergo cold flow analysis along with the standard, circular orifice shape. The shapes which were selected are shown on the chart opposite.

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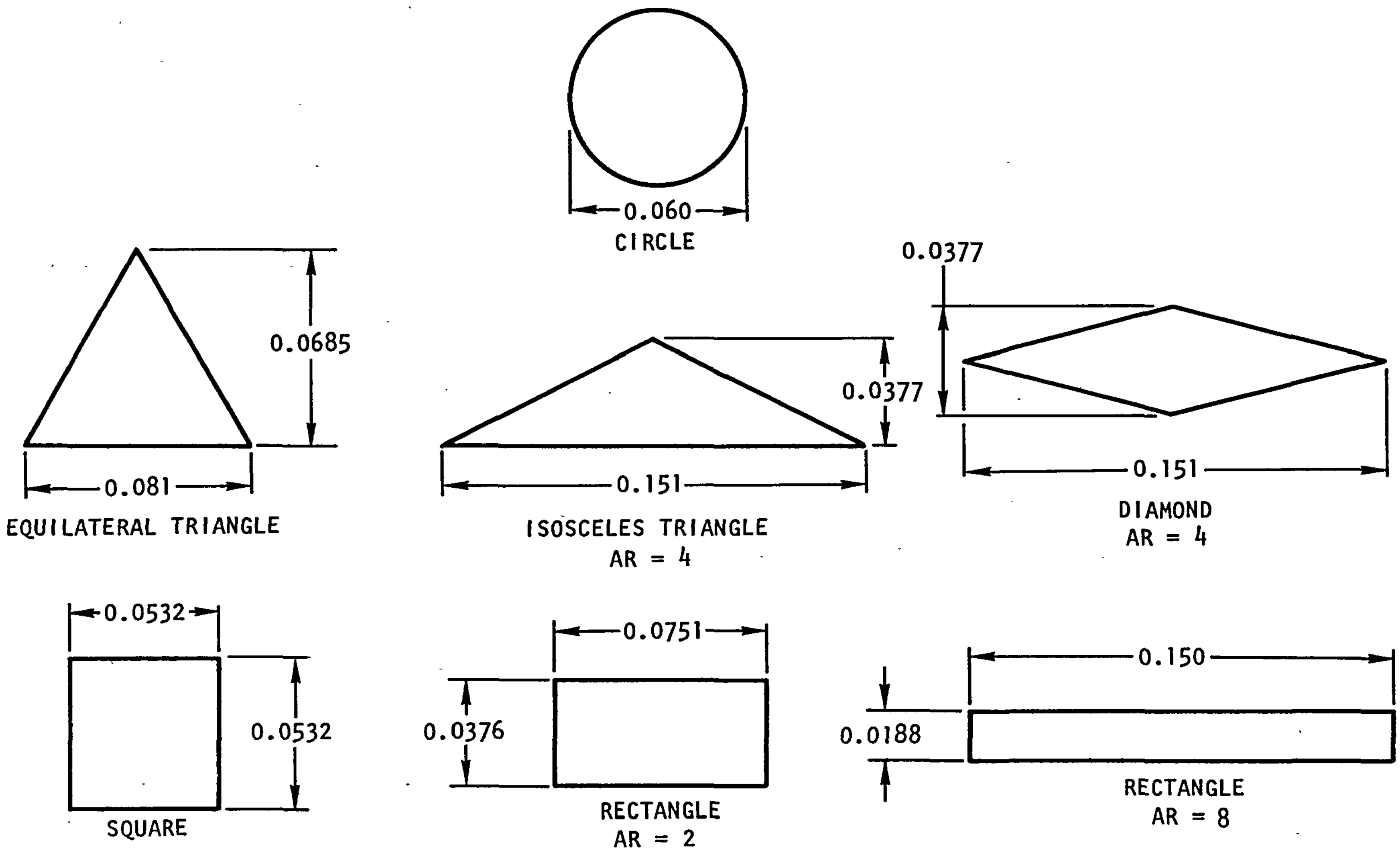
ORIFICES SELECTED FOR SINGLE ORIFICE COLD-FLOW EVALUATION



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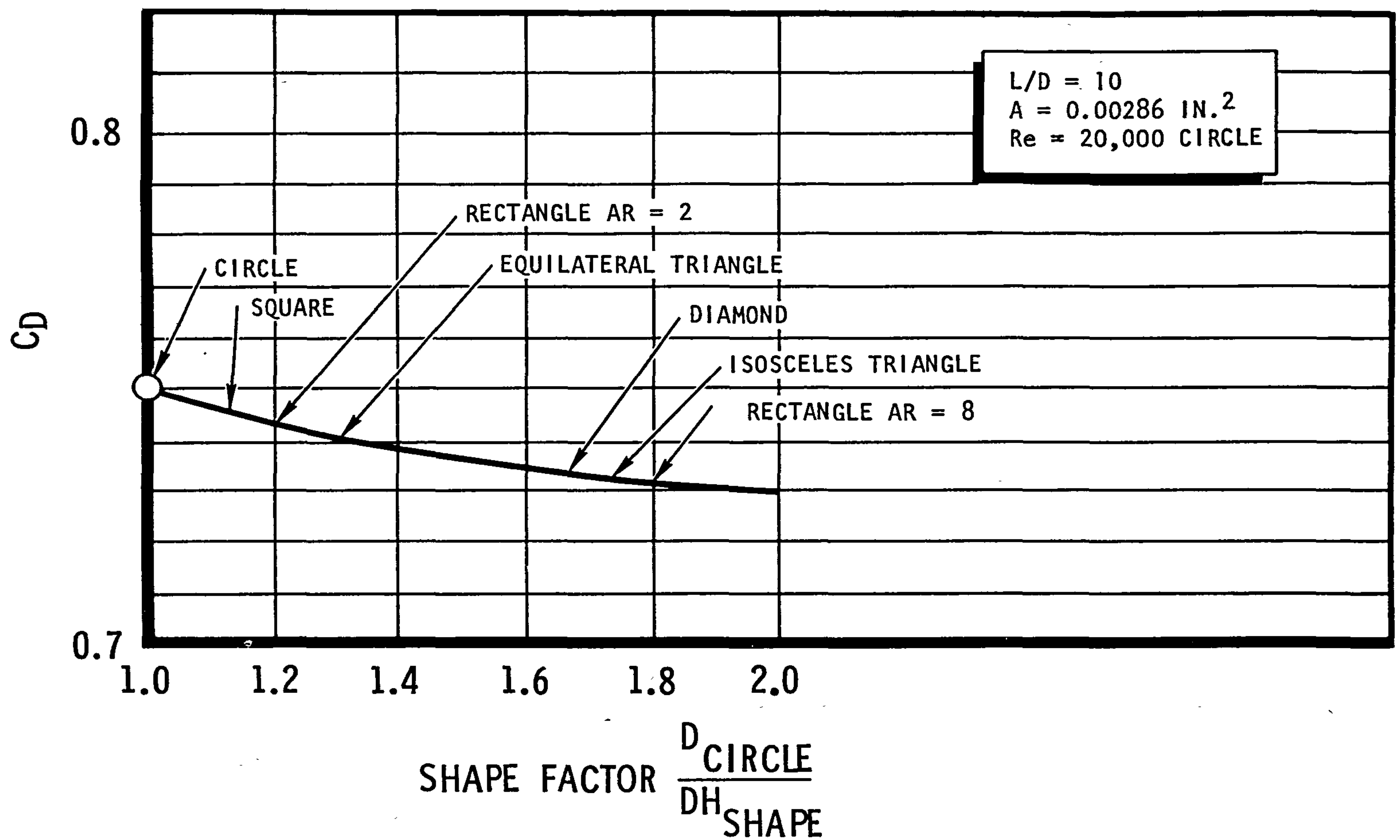
Single Orifice Cold Flow Results

Following the preliminary analysis and evaluation, cold flow models of each of the orifice shapes selected (see previous page) were designed and fabricated; all with the same cross-sectional area. Several orifice length-to-hydraulic diameter ratio configurations were fabricated for each shape (L/D_H = 2, 4, 6 and 20).

The significant results of the orifice cold flow studies are summarized on the facing chart. Each orifice was tested over a range of total flow rate, Δp, backpressure, and length. It was discovered that the results of these various experiments, expressed in terms of the orifice coefficient, C_D, could be correlated with standard techniques available from circular orifice technology if the hydraulic diameter was employed as the characteristic length to describe a given shape and if a proper description of the initial boundary layer development within the orifice was employed. The curve shown was generated from the correlations which were developed. The orifice coefficient is shown as a function of the shape factor (diameter of a circle/hydraulic diameter of a noncircular shape having the same cross-sectional area). The ranking of the various shapes which were tested is also shown. It can be seen that the orifice coefficient is not a strong function of orifice shape.

SUMMARY OF ORIFICE COEFFICIENT RESULTS



Single Element Cold Flow Results

Based upon an evaluation of the results of the single orifice cold flow studies, the triangle and the rectangle were selected as the candidate shapes for further evaluation in a single injector element cold flow study. The unlike-doublet element type was selected for these studies due to its relative simplicity. Several doublet elements were designed and fabricated which incorporated the orifice shapes which had been selected. These various patterns are shown schematically on the facing chart. An unlike-doublet with circular orifices was also fabricated to provide a standard of comparison for element performance. (Each configuration is given a letter for reference purposes.)

These elements were designed specifically for application with the NTO/50-50 propellant combination based upon optimization criteria which were available at the time for circular orifices.

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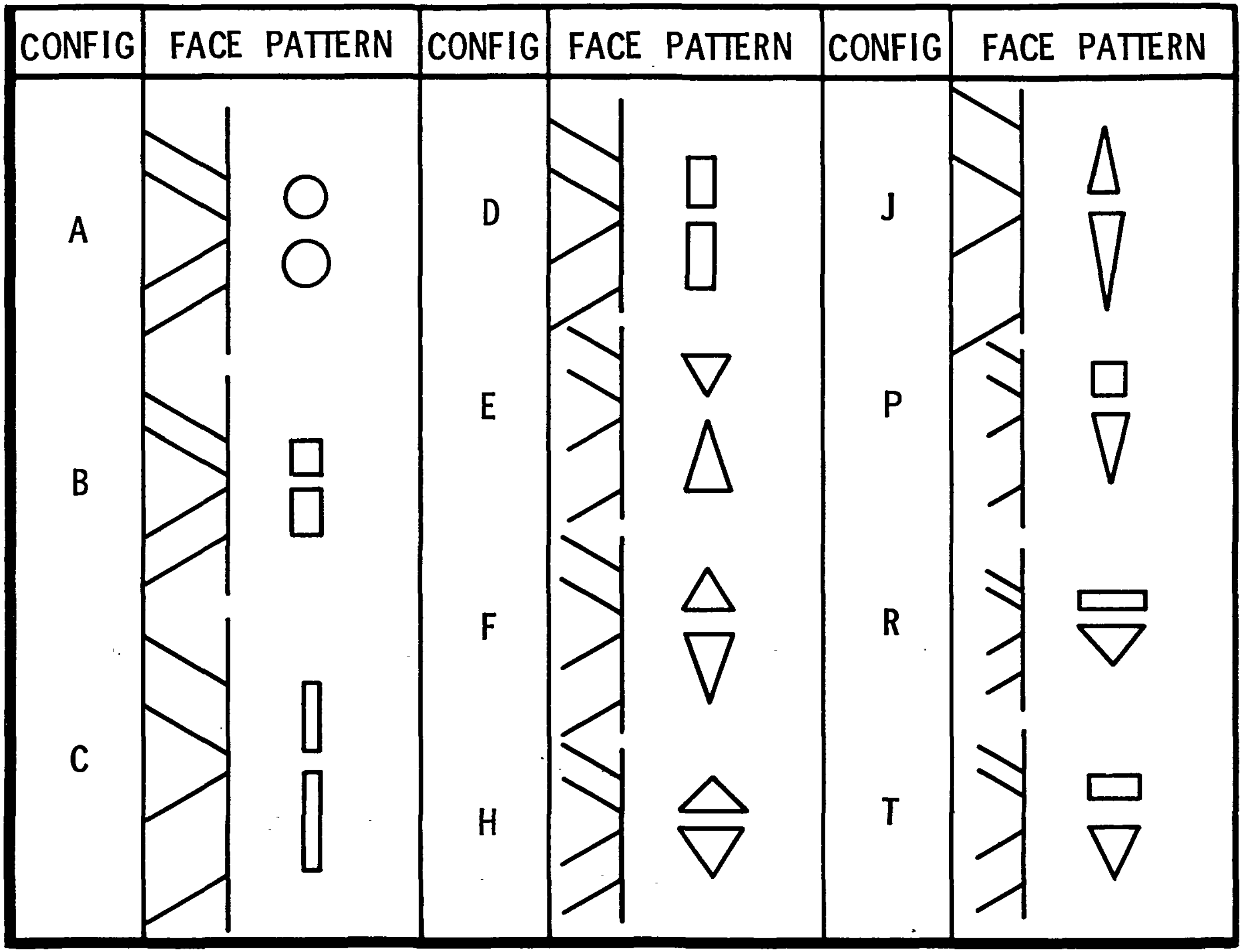
SUMMARY OF UNLIKE-DOUBLET ELEMENT DESIGNS



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Mixing uniformity tests were conducted with a multi-tube collection matrix employing trichloroethylene and water as oxidizer and fuel simulants respectively. The element spray field was directed at the tube matrix. The mass collected and the mixture ratio in each tube were recorded. These data were interpreted in terms of E_m according to the formulation shown.

Mixing uniformity was correlated in terms of a momentum ratio parameter, N.

DEFINITIONS

$$E_m = \left[1 - \sum_{i}^{N} \frac{\dot{w}_i}{\dot{w}_T} \frac{(R - r_i)}{R} - \sum_{i}^{N} \frac{w_i}{\dot{w}_T} \frac{(R - r_i)}{R - 1} \right] 100$$

WHERE

E_m = MIXING UNIFORMITY PARAMETER

$\dot{w}_i/\dot{w}_T$ = MASS FRACTION IN THE STREAM TUBE

R = RATIO OF TOTAL OXIDIZER MASS TO TOTAL OXIDIZER AND FUEL MASS

r_i = RATIO OF OXIDIZER MASS TO TOTAL OXIDIZER AND FUEL MASS IN AN INDIVIDUAL STREAM TUBE FOR $r_i < R$

$\bar{r}_i$ = RATIO OF OXIDIZER MASS TO TOTAL OXIDIZER AND FUEL MASS IN AN INDIVIDUAL STREAM TUBE FOR $r_i > R$

$$N = \frac{1}{1 + \frac{M_f}{M_o} \frac{D_o}{D_f}}$$

WHERE

N = MOMENTUM RATIO PARAMETER

M_f/M_o = FUEL-TO-OXIDIZER MOMENTUM RATIO

D_o/D_f = OXIDIZER-TO-FUEL ORIFICE HYDRAULIC DIAMETER RATIO

$D_{o,f}$ = 4 X AREA/PERIMETER = HYDRAULIC DIAMETER

Significant results of the mixing uniformity tests are presented on the facing page wherein E_m is shown as a function of N. The element configurations represented were the highest performing elements from each element group. Mixing results for a spray fan element which was tested are also shown. This element was composed of two unlike impinging nozzle orifices which produced pre-atomized spray fans. It may be noted that the noncircular elements optimize at the same value of N at which the optimum value for the circular element is located. The level of mixing is significantly higher for the noncircular elements.

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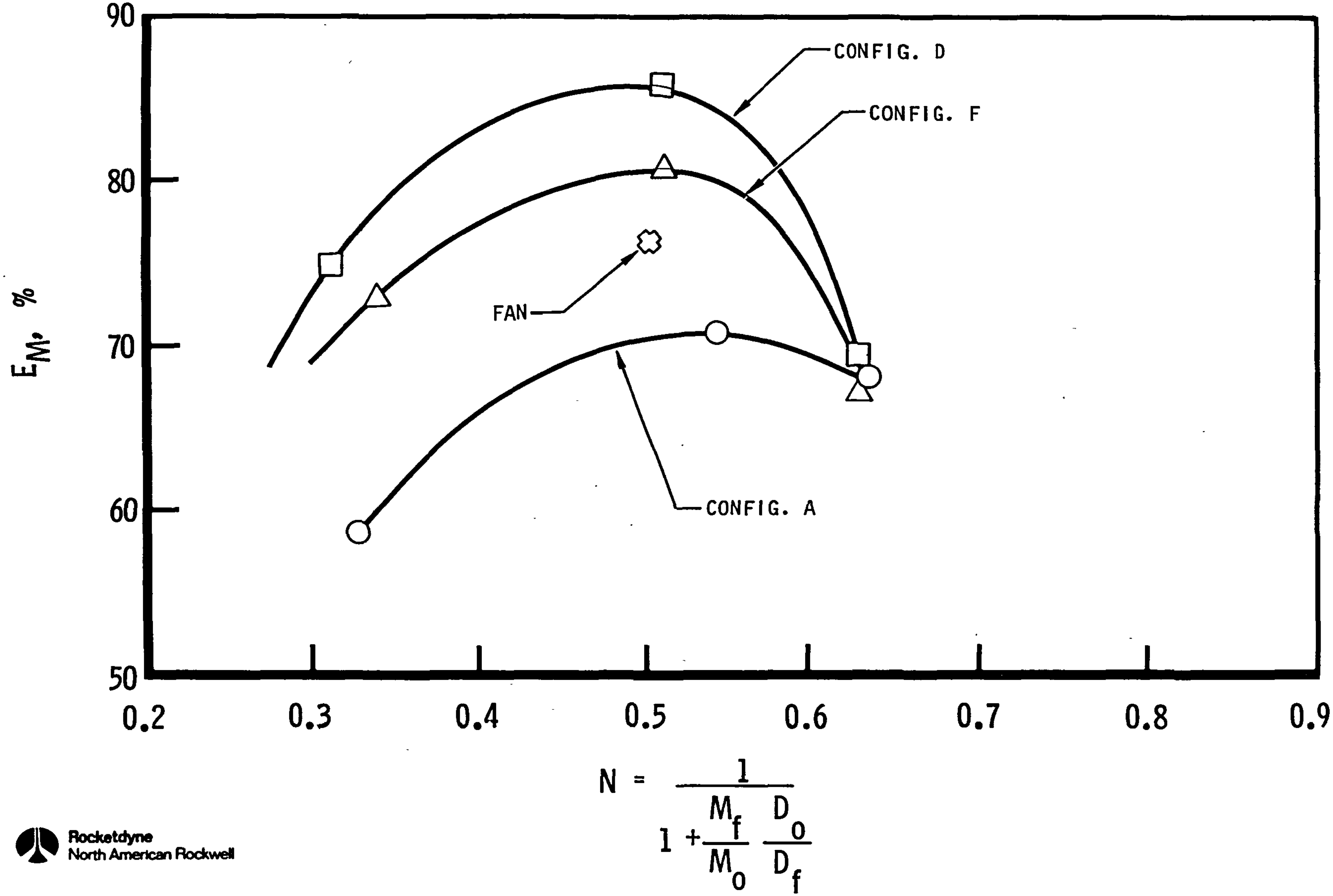
MIXING UNIFORMITY RESULTS FOR UNLIKE DOUBLET ELEMENTS



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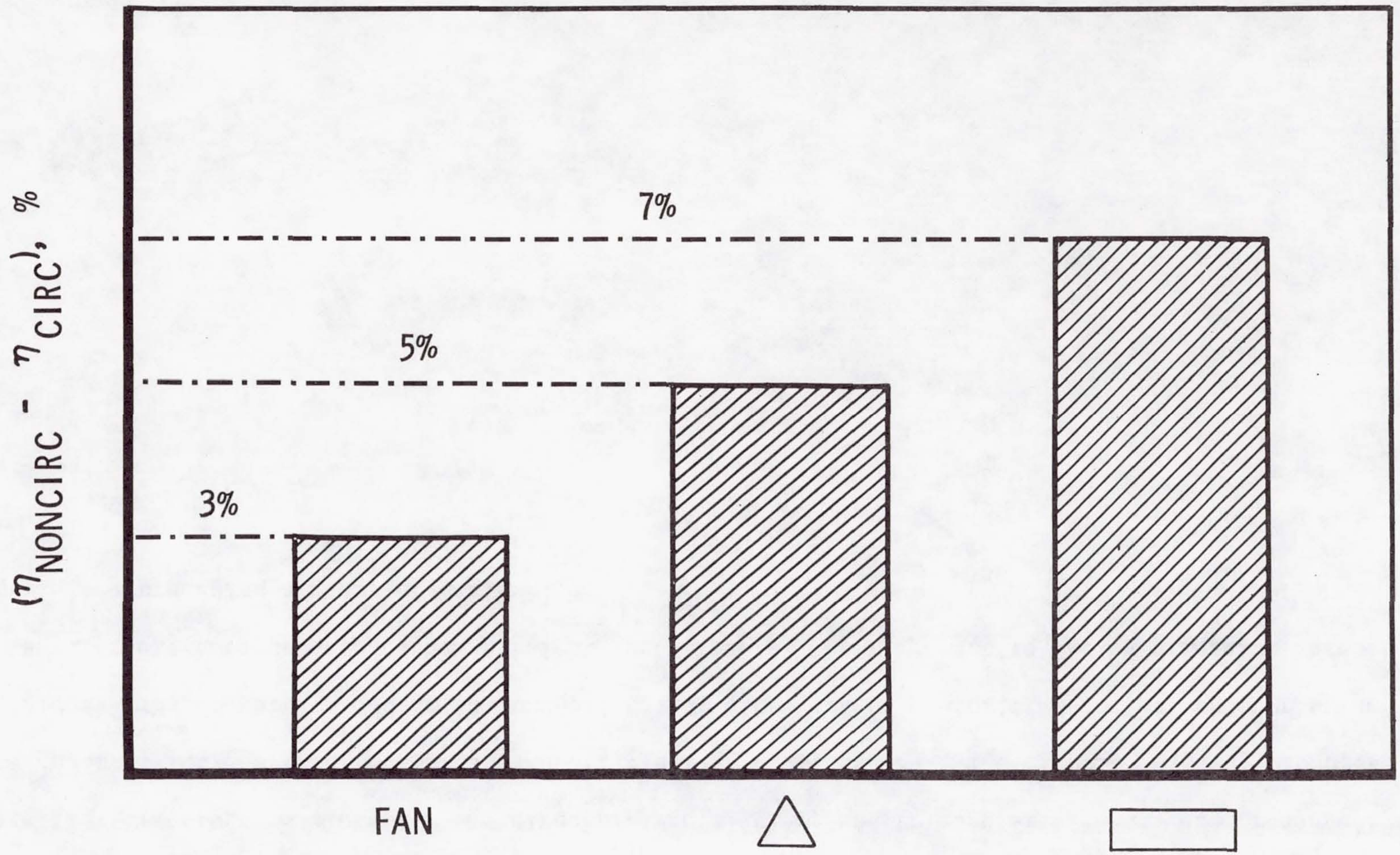
The mixing results were employed to estimate a mixing limited C* efficiency for the various element types through theoretical combustion modeling. Results of this study are presented opposite wherein the difference between the mixing limited C* efficiency of a given element and the circular element is represented for each configuration.

COMPARISON OF MIXING LIMITED C* EFFICIENCY IMPROVEMENT PROVIDED BY NONCIRCULAR ORIFICE ELEMENTS



Mass flux distribution plots for the circular, rectangular, and triangular elements are presented on the accompanying chart for comparison (configurations A, D, and F). The circular element produces a "kidney" shaped pattern due to the unequal facing diameters of the orifices. The rectangular and triangular elements produce rather straight patterns, as their respective facing dimensions are equal (see preceding summary of designs).

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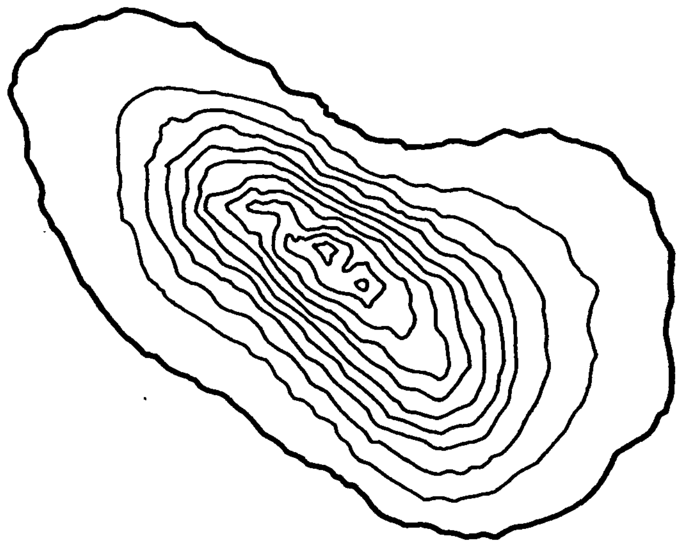
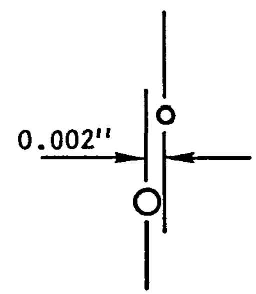
MASS DISTRIBUTION MAPS FOR UNLIKE-DOUBLET ELEMENTS





CONFIGURATION A

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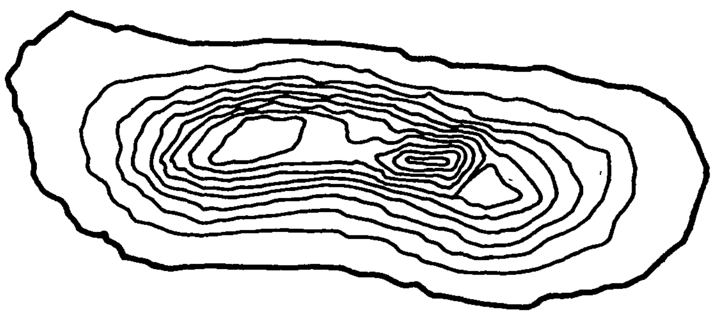

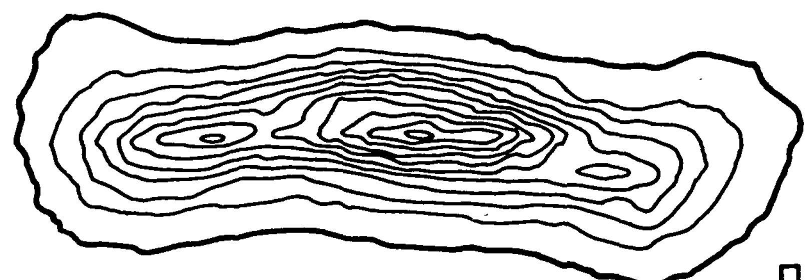

CONFIGURATION F

CONFIGURATION D

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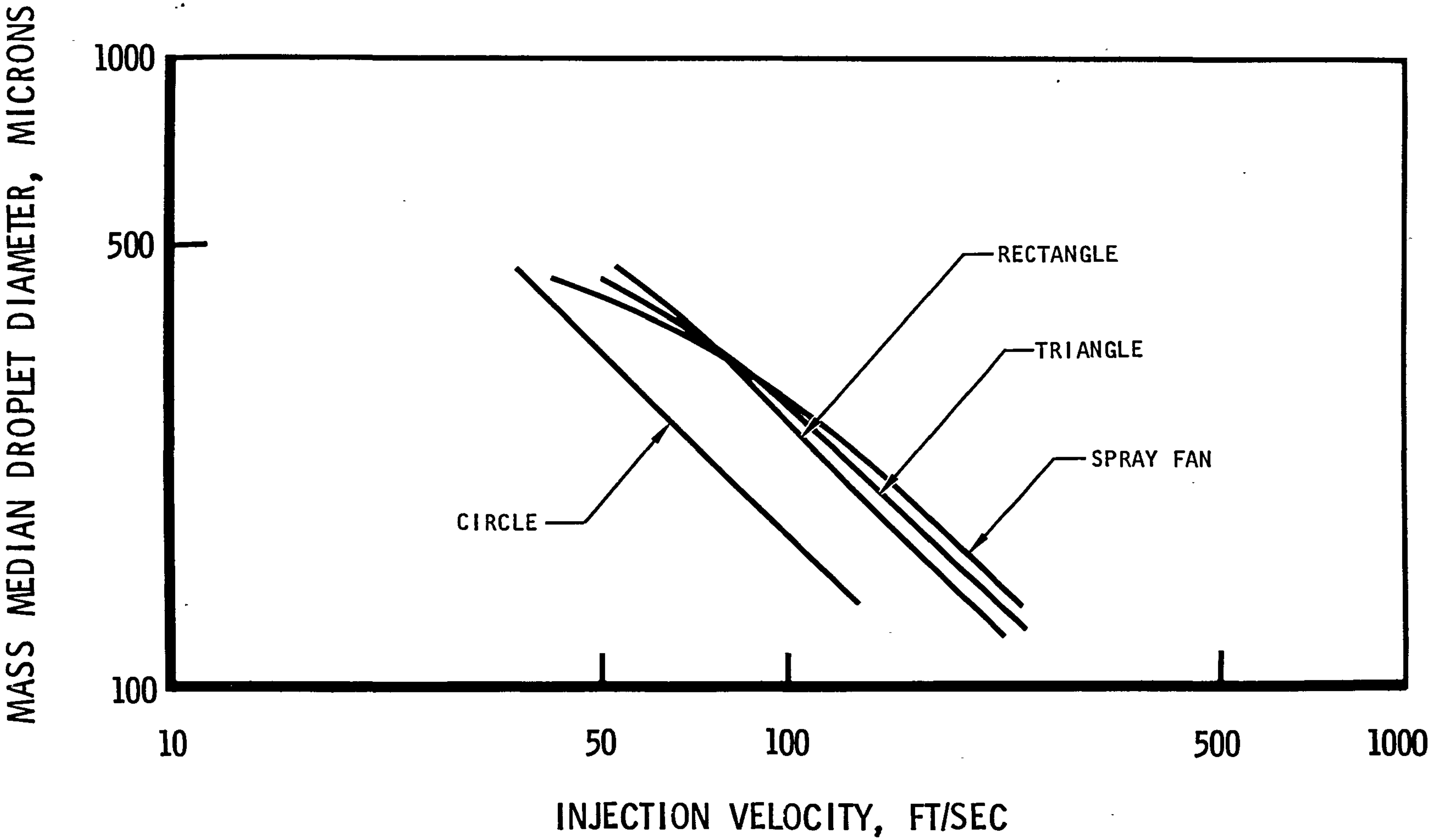
Atomization testing was conducted with the molten wax technique. The results were expressed in terms of the mass median droplet diameter, $\overline{D}$, of a given droplet distribution.

Results of the atomization testing are summarized wherein the value of $\overline{D}$ for the various elements is shown as a function of injection velocity. Data shown were obtained with the elements operating at a value of the momentum ratio parameter, $N = 0.5$. Dropsizes shown apply to both the fuel and oxidizer species for a given element. It is evident that the circular unlike-doublet demonstrates the superior atomization characteristics.*

*Data presently being taken under Contract NAS7-726 suggest that the circular orifices in the noncircular program may have been operating in a separated mode. Thus, the jet diameters and the droplets were artificially small. Under hot fire conditions, the orifices will flow full and the dropsizes of the various elements may be very nearly equal.

COMPARISON OF WAX DROPLET DIAMETERS FOR NONCIRCULAR AND CIRCULAR ORIFICE DESIGNS

(1 MICRON = 10^{-6} METERS)



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Single Element Hot Fire Results

Single element hot fire injectors were fabricated for each of the optimum noncircular unlike-doublet patterns and for the circular orifice pattern. Configurations A, D, and F and a spray fan unlike-doublet were fabricated.

Hot fire tests were conducted with these elements with the NTO/50-50 propellant combination. Unfortunately, "blowapart" occurred for the unlike doublet jet type elements and no comparison could be made between circular and noncircular orifices based upon hot fire. However, results predicted from cold flow show the noncircular orifices have superior mixing capabilities.

The spray fan element, however, was not subject to "blowapart". As a result, the hot fire results from this element agreed quite well with the results which were predicted. A comparison between the spray fan element actual and predicted overall C* efficiency is shown on opposite page. The fact that these data do correlate well with cold flow results substantiates the use of cold flow data to compare the relative performance of the various impinging jet elements for which hot fire correlations could not be obtained.

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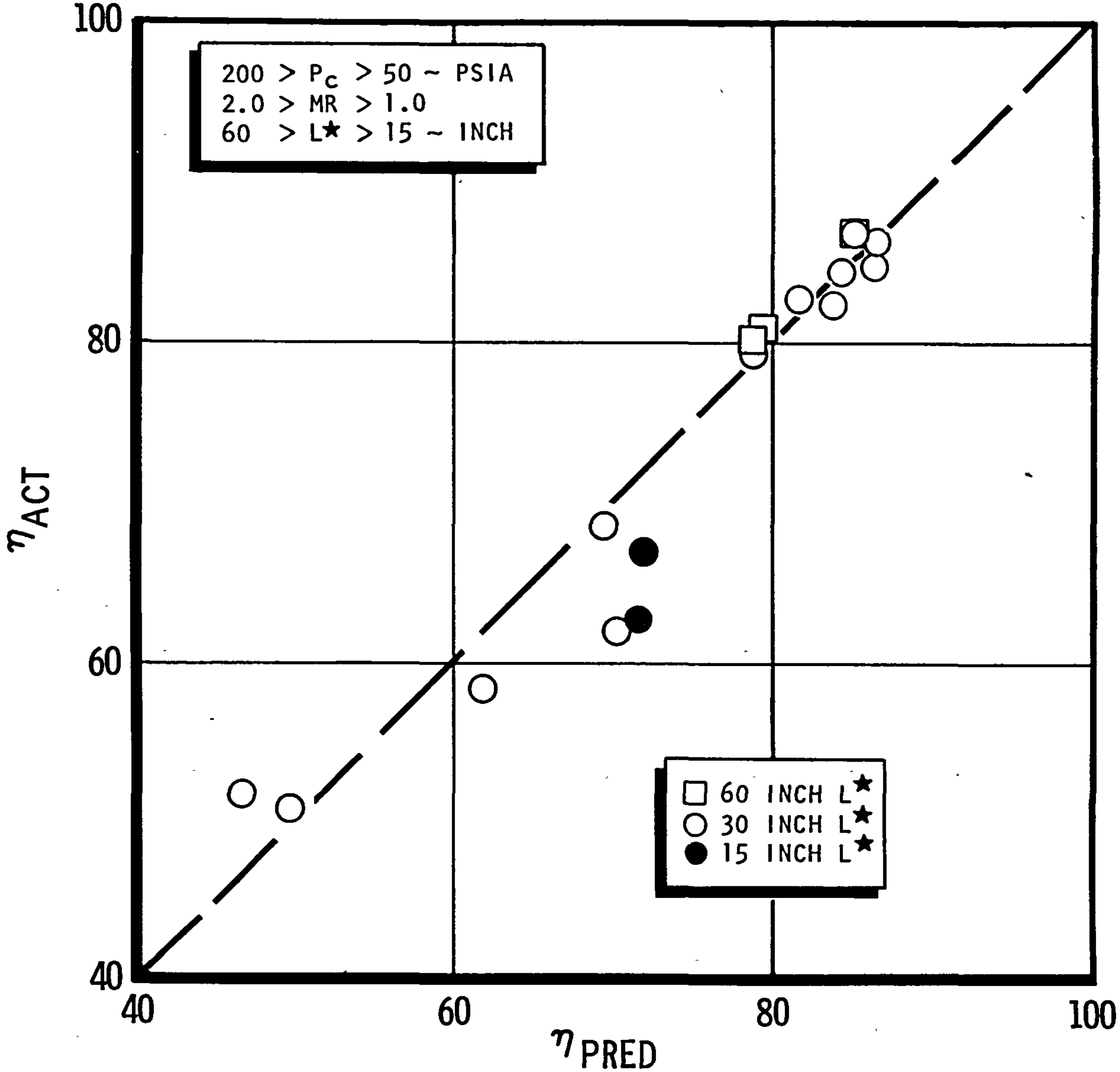
COMPARISON OF PREDICTED AND ACTUAL C* EFFICIENCY FOR THE SELF-ATOMIZING NOZZLE



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PHASE II

Preliminary Analysis and Evaluation

The Phase II effort was initiated with a preliminary analysis and evaluation of candidate element types for application with the gaseous hydrogen/liquid oxygen propellant combination. The analysis and evaluation guidelines were dictated by the design requirements for a conceptual Orbit Maneuvering System (OMS) Engine for the Space Shuttle System. These guidelines are summarized in tabular form on the opposite page.

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ENVISIONED OPERATIONAL CHARACTERISTICS OF THE ORBIT MANEUVERING SYSTEM ENGINE

• PROPELLANTS	LOX/GH_2
• THRUST	8000 LBF
• CHAMBER PRESSURE	800 PSIA
• MIXTURE RATIO	6:1
• EXPANSION RATIO	200:1

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The injector element types selected for analysis (incorporating noncircular orifices) are shown opposite along with a standard, circular concentric tube injector element which was selected as a standard for comparison.

Existing experimental data and analytical techniques were employed to select design configurations of each element shown opposite for the conditions listed in preceding Table. The design study was limited to the 200 lb_f thrust per element level. Both mixing and atomization (expressed in terms of E_m and $\bar{D}$ respectively) characteristics were estimated for the various elements.

A preliminary evaluation of element types was conducted based upon the results of the preliminary analysis. The evaluation results suggested that the noncircular (in this case, rectangular) concentric tube element might possess the greatest potential for providing superior mixing and atomization characteristics for gas/liquids applications. This element, was, therefore, selected for experimental evaluation along with the circular concentric tube element for comparison.

CANDIDATE ELEMENT CONCEPTS FOR GAS/LIQUID APPLICATIONS

BASIC CONCEPT TYPE		
CONCENTRIC TUBE	IMPINGING SPRAY FANS	IMPINGING JETS
STANDARD CIRCULAR CONCENTRIC TUBE	TRIPLET SELF-ATOMIZING NOZZLES	UNLIKE DOUBLET GAS SHOWERHEAD
NONCIRCULAR CONCENTRIC TUBE INCREASED SURFACE AREA	UNLIKE DOUBLET ONE SPRAY NOZZLE	4-ON-ONE GAS SHOWERHEAD
	4 SPRAY NOZZLES ON ONE GAS SHOWERHEAD	4-ON-ONE GAS SHOWERHEAD

SHADED AREAS INDICATE LIQUID INJECTION ORIFICES

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Single Element Cold Flow Evaluation

Cold flow model elements were designed and fabricated for both the circular and noncircular concentric tube configurations. The cold flow model designs were not the same as those obtained in the preliminary analysis. The models were changed to facilitate cold flow testing in existing facilities. The gas velocities were somewhat lower in the model hardware, (i.e., lower than the velocities estimated in the preliminary analysis). The model hardware is shown in the photographs. The respective fuel and oxidizer areas of the two elements were made equal to allow for equal injection velocities at equivalent operating conditions. The length-to-width ratio, or aspect ratio, of the rectangular element oxidizer orifice was equal to 6:1.

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COLD FLOW MODEL HARDWARE

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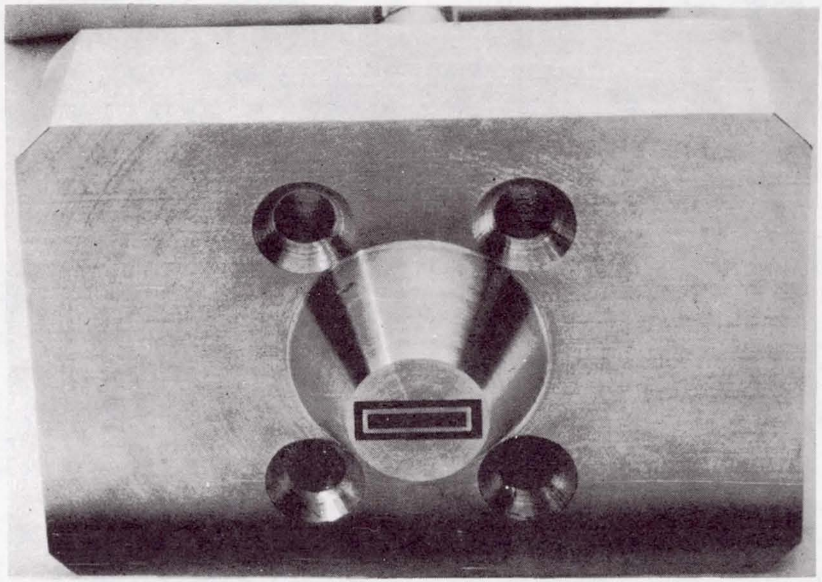

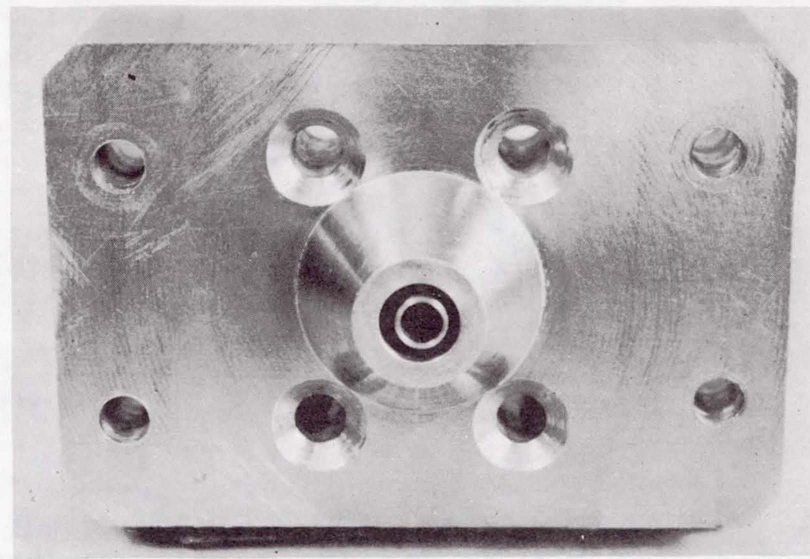

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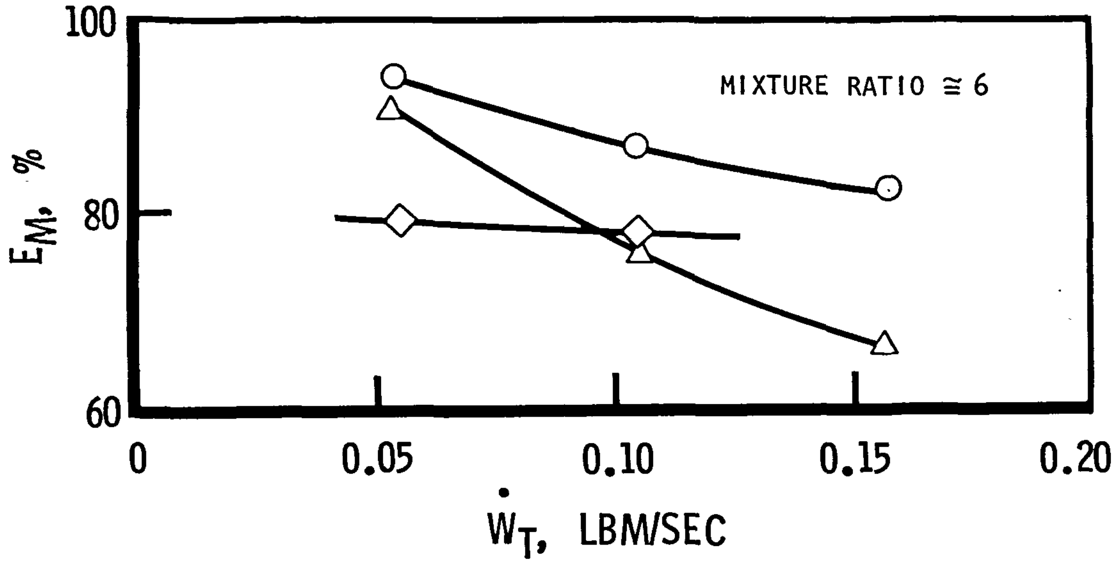
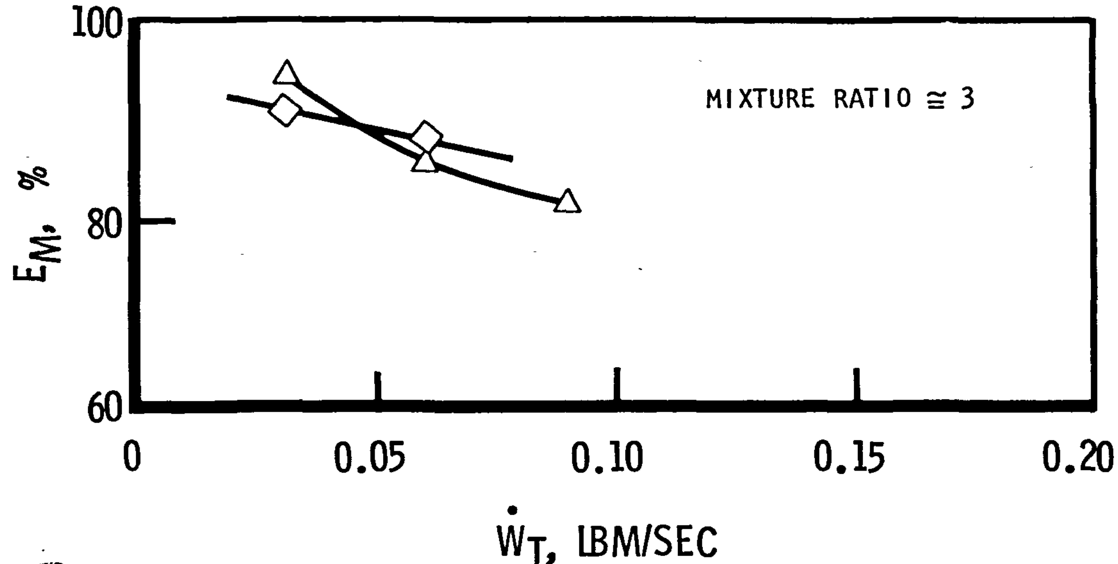
Mixing uniformity tests conducted with a special two-phase impact probe designed to distinguish liquid and gas species flow rates at a given point in a two-phase flow field. The data obtained were reduced by the same method as that described in the Phase I section of this report.

Results of the mixing uniformity experimental study are presented on the facing page in terms of the mixing unformity parameter, E_m, and total element flow rate, $\dot{w}_T$. Results are presented for both the circular and the noncircular concentric tube elements for liquid-to-gas weight flow rate mixture ratios of 3:1 and 6:1. Data from the noncircular element were obtained with the oxidizer center post tip flush with the injector face. Data from the circular element were obtained with both flush and recessed centerpost (the recessed configuration employed a recess of one liquid orifice diameter, $R/D_L=1$).

In general, the mixing levels of the two elements were similar, with the noncircular element being much less sensitive to total element flow rate. Mixing levels were higher at mixture ratio 3:1 than at mixture ratio 6:1 for both elements for fixed values of fuel injection velocity. The smaller flowrates, for which the circular element shows a potential mixing advantage, are too low for practical applications due to the low injection velocities produced.

The flow patterns produced by the two elements showed that the characteristics of the two were quite different, even at equivalent values of E_m. The circular element was typified by an oxidizer-rich core on the centerline of the element while the noncircular element was typified by a fuel-rich core on the centerline. These results suggested that the aspect ratio of 6:1 selected for the noncircular element was too large and the liquid species, being spread very thin, was aspirated too rapidly from the center to the outside of the flow field. This implies that an optimum aspect ratio may exist which is less than 6:1.





MIXING UNIFORMITY RESULTS

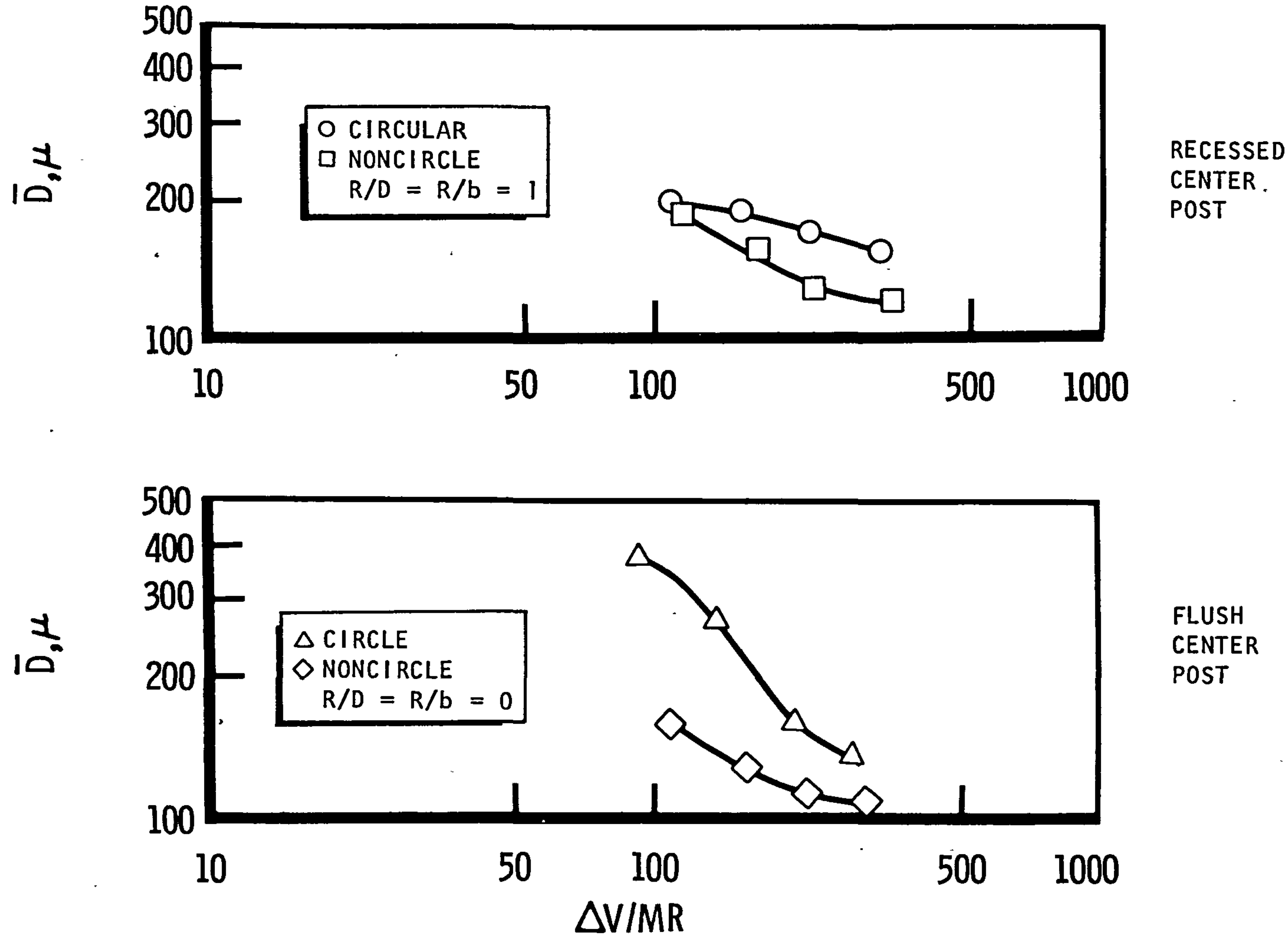
(CIRCULAR AND NONCIRCULAR CONCENTRIC TUBE ELEMENTS AS A FUNCTION OF TOTAL ELEMENT FLOW RATE)

◇ NONCIRCULAR, RECESS = 0
△ CIRCULAR, RECESS = 0
○ CIRCULAR, RECESS = ONE D_L

Atomization experiments were conducted with these elements in the same manner as that described for Phase I with the exception of the fuel simulant which was hot gaseous nitrogen in this case. Results of these tests are shown on accompanying page where mass median droplet diameter, $\bar{D}$, is shown as a function of a modeling parameter $\Delta V/MR$ (gas velocity minus liquid velocity divided by mixture ratio). Both the flush and recessed centerpost* data include results obtained at mixture ratios of 3:1 and 6:1. It is evident from the plots that the noncircular element produced droplets which were significantly smaller than those produced by the circular element in both the flush and recessed configurations.

*Recess depth for the noncircular element was equal to the smaller dimension on the oxidizer orifice, b.

COMPARISON OF CIRCULAR AND NONCIRCULAR CONCENTRIC TUBE MASS MEDIAN DROPLET* SIZES



* DROP SIZES ARE EXPRESSED IN MICRONS (ONE MICRONS=10^{-6} METERS)

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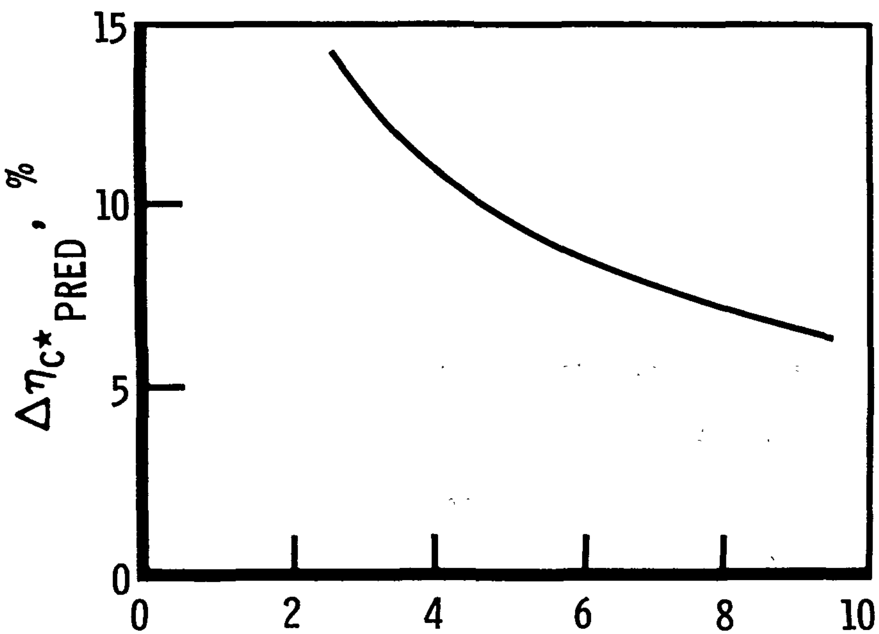
Performance Analysis

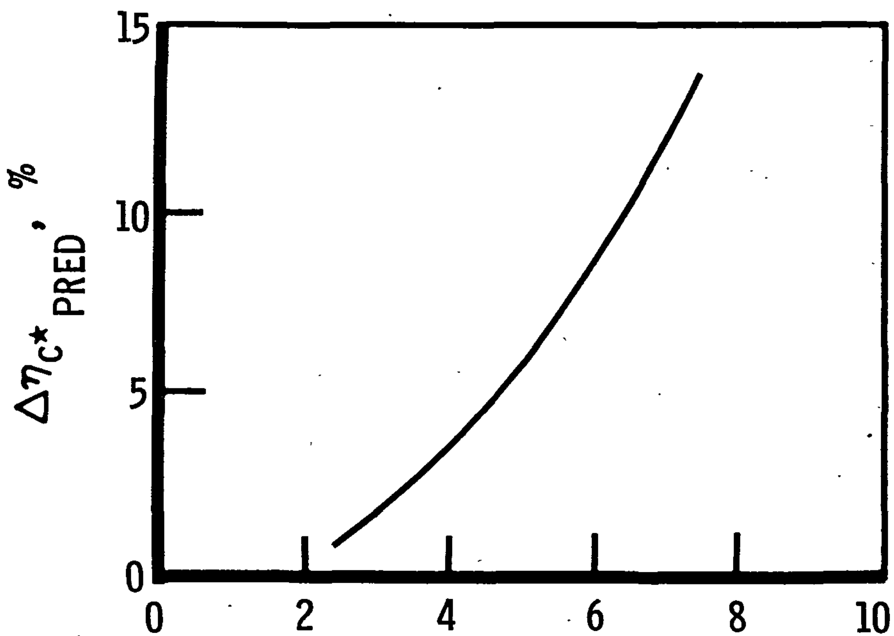
A performance analysis was conducted to estimate the overall C* efficiencies which could be expected from the two elements based upon their individual contributions to mixing and atomization processes. Mixing and vaporization rate limited combustion models were employed in this analysis.

A chamber pressure of 200 psia was selected at which to model the performance characteristics of the two elements due to the fact that the density of gaseous hydrogen at this pressure is equal to the density of the gaseous nitrogen at ambient pressure which was employed as a fuel simulant. Performance was predicted over a range of mixture ratio and chamber length for the two elements. Results of these analyses are shown wherein the difference between the predicted performance for the noncircular and circular concentric tube elements is shown as a function of mixture ratio and chamber length. The results suggest that the noncircular element possesses distinct advantages at all conditions evaluated and that its greatest potential lies in applications to higher mixture ratio and shorter chamber configurations.

DIFFERENCE IN PREDICTED C* EFFICIENCY BETWEEN NONCIRCULAR AND CIRCULAR CONCENTRIC TUBE ELEMENTS AS A FUNCTION OF MIXTURE RATIO AND CHAMBER LENGTH

(P_c 200 PSIA, LOX GH_2)





FUTURE EFFORT

The noncircular orifice program is continuing with a proposed Phase III and Phase IV effort. The additional effort will be devoted to the development and characterization of injector elements for application with the LOX/GH_2 propellant combination. Two element types will be considered: the rectangular concentric tube element and the one other candidate which is to be selected early in the Phase III effort. Phase III will be directed toward developing optimization design criteria for the elements based upon cold flow experimental results. Hot-fire experimentation will be undertaken in Phase IV with both single and multi-element hardware. The results of the program will be employed to design candidate noncircular elements for the various engines for the Space Shuttle System.

CONCLUDING REMARKS

EVALUATION OF NONCIRCULAR ORIFICE LIQUID/LIQUID INJECTOR ELEMENTS - PHASE I

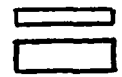
In the overall view, for liquid/liquid applications the results show that the selection on noncircular or circular orifice designs would be based solely on the specific application. For instance, if mixing were the primary consideration, then noncircular injectors would be selected. In addition, for some propellant combinations such as $FLOX/CH_2$, non-circular unlike-doublet injector elements can be designed with reasonable orifice dimensions, while circular element designs would be unrealistic. This difficulty for circular orifice designs can be overcome by utilizing a four-on-one pattern, rather than the unlike-doublet. This, however, carries a penalty in design complexity and impairs the ability to pack a large number of elements into the cross-sectional area of the given injector. Clearly, the availability of noncircular element designs provides the designer with an extended capability and with more flexibility in his selection of element types.

For the specified designs considered, the following conclusions can be made:

1. Orifice flow and spray characteristics are predictable for noncircular orifices using techniques identical to those used for circular orifices.

2. Noncircular elements produce significantly better mixing than a circular unlike doublet at equivalent design conditions.

3. Circular unlike-doublet elements produce smaller dropsizes than the specific noncircular elements evaluated. This conclusion could be different if the elements were designed to impinge in the following manner.



rather than

4. Noncircular element spray characteristics are less sensitive to orifice flow characteristics than circular orifices.

5. Self-atomizing fan elements are not subject to blowapart because they provide mixing in the atomized state rather than impingement of solid jets.

6. EDM provides an ideal fabrication method for producing noncircular elements.

EVALUATION ON NONCIRCULAR ORIFICE GAS/LIQUID INJECTOR ELEMENT - PHASE II

The most significant conclusion which may be made concerning the results of this program is that the noncircular concentric tube element does possess inherent performance advantages for gas/liquid propellant applications. The noncircular element has demonstrated performance characteristics which are superior to those of a conventional circular concentric tube element for the same operating conditions. This is indeed an outstanding accomplishment in the light of the fact that the circular concentric tube element has a long established record as one of the best candidate element types for gas/liquid propellant combinations.

It is in the area of propellant atomization that the noncircular concentric tube element has demonstrated the greatest promise. Dropsizes produced by this element were considerably smaller than those produced by the circular concentric tube element.

Significant conclusions can also be reached concerning the mixing characteristics of the noncircular concentric tube element. At the outset of the program, it was predicted that both the mixing and the atomization performance of the noncircular orifice concentric tube element were related directly to the aspect ratio of the element, and that elements with higher and higher aspect ratios would be higher and higher performing. As far as atomization is concerned, this appears to be the case. However, the data which have been evaluated suggest that there may be an optimum aspect ratio for mixing. This conclusion was reached from examination of the mass and mixture ratio distribution data which show clearly that the maldistribution evinced by the noncircular element was characterized by relatively high amounts of the liquid species in the outer zones. This suggests that the injected gas phase momentum was so effectively transferred through shear along the large gas/liquid contact periphery that the liquid phase was aspirated to a high degree and thrown to the outside of the flow field. On the other hand, the liquid phase remained near the center of the flow field for the circular concentric tube element with approximately the same level of gas species momentum. This high mixture ratio core is principally responsible for the incomplete mixing of these elements. It appears that the liquid was actually spread out somewhat too far with a rectangular concentric tube orifice aspect ratio of 6:1, in other words the problem of liquid "coring" was apparently over-corrected. It is thus probable that the distribution of liquid throughout the gas flow field could be made highly uniform with the proper selection of the aspect ratio for the noncircular element. This "optimum" aspect ratio would be expected to fall between 6:1 and 1:1. The selection of aspect ratio would have to be based on an optimization study which would include consideration of both mixing and atomization effects.

Aside from the performance aspects, the variable aspect ratio feature of the noncircular concentric tube element also provides a powerful new design capability for tailoring the character of the flow distribution pattern produced by the elements. This feature should be quite valuable from the standpoint of enhancing injector/chamber compatibility and controlling chamber heat flux.

Further investigation of the characteristics of the noncircular concentric tube element is more than warranted by the results of this program. In an optimized configuration, this element can provide significant contributions in the field of gas/liquid injector technology.