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ALTITUDE DEVELOPMENTAL TESTING OF THE J-2 ROCKET ENGINE IN PROPULSION ENGINE TEST CELL (J-4)(TEST J4-1801-03)

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Per AF Letter
dtd 12 July 74
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William O. Cole

C. H. Kunz
ARO, Inc.

December 1967

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LARGE ROCKET FACILITY
ARNOLD ENGINEERING DEVELOPMENT CENTER
AIR FORCE SYSTEMS COMMAND
ARNOLD AIR FORCE STATION, TENNESSEE

NOTICES

ALTITUDE DEVELOPMENTAL TESTING OF THE J-2 ROCKET ENGINE IN PROPULSION ENGINE TEST CELL (J-4) (TEST J4-1801-03)

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FOREWORD

The work reported herein was sponsored by the National Aeronautics and Space Administration (NASA), Marshall Space Flight Center (MSFC), under System 921E, Project 9194.

The results of the tests presented were obtained by ARO, Inc. (a subsidiary of Sverdrup & Parcel and Associates, Inc.), contract operator of the Arnold Engineering Development Center (AEDC), Air Force Systems Command (AFSC), Arnold Air Force Station, Tennessee, under Contract AF40(600)-1200. Program direction was provided by NASA/MSFC; engineering liaison was provided by North American Aviation, Inc., Rocketdyne Division, manufacturer of the J-2 rocket engine and Douglas Aircraft Company, manufacturer of the S-IVB stage. The testing reported herein was conducted on July 26, 1967, in Propulsion Engine Test Cell (J-4) of the Large Rocket Facility (LRF) under ARO Project No. KA1801. The manuscript was submitted for publication on September 9, 1967.

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This technical report has been reviewed and is approved.

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Director of Test

ABSTRACT

Two firings of the Rocketdyne J-2 rocket engine were conducted in Test Cell J-4 of the Large Rocket Facility. The firings were accomplished during test period J4-1801-03 at pressure altitudes of approximately 100,000 ft to evaluate S-IVB/S-V start condition effects on (1) engine start transients, (2) gas generator outlet temperature, (3) augmented spark igniter operation, and (4) fuel pump low level stall margin for J-2 engine J-2052. Engine components were thermally conditioned to temperatures predicted for S-IVB/SV first burn and one orbit restart on firings 03A and 03B, respectively. Excessive gas generator outlet temperature peaks were experienced on these firings, resulting in a gas generator outlet temperature probe failure and premature engine shutdown on firing 03B. The two remaining scheduled firings for the test period were cancelled. Post-test inspection revealed erosion of the fuel turbine first stage, and small cracks were found in the curvic coupling of fuel turbine first-stage wheel. The accumulated engine firing duration was 31.3 sec.

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NOMENCLATURE

A	Area, in.^{2}
ASI	Augmented spark igniter
ES	Engine start, designated as the time that helium control and ignition phase solenoids are energized
GG	Gas generator
MOV	Main oxidizer valve
STDV	Start tank discharge valve
t_0	Defined as the time at which the opening signal is applied to the start tank discharge valve solenoid
VSC	Vibration safety counts, defined as engine vibration in excess of 150 g rms in a 960- to 9000-Hz frequency range

SUBSCRIPTS

f	Force
m	Mass
t	Throat

SECTION I
INTRODUCTION

Testing of the Rocketdyne J-2 rocket engine (S/N J-2052) using a S-IVB battleship stage has been in progress since July 1966 at AEDC in support of the J-2 engine application on the Saturn IB and Saturn V launch vehicles for the NASA Apollo Program. The two firings reported herein were conducted during test period J4-1801-03 on July 26, 1967, in Propulsion Engine Test Cell (J-4) (Figs. 1 and 2, Appendix I) of the Large Rocket Facility (LRF) to evaluate S-IVB/S-V start condition effects on (1) engine start transients, (2) gas generator outlet temperature, (3) augmented spark igniter operation, and (4) fuel pump low level stall margin. These firings were conducted at a pressure altitude of approximately 100, 000 ft, using predicted J-2 engine flight temperatures to simulate first burn and restart after one orbit.

Data collected to accomplish the test objectives are presented herein. Copies of all data obtained during this test have been previously supplied to the sponsor. The results of the previous test period are reported in Ref. 1.

SECTION II
APPARATUS

2.1 TEST ARTICLE

The test article was a J-2 rocket engine (Fig. 3) designed and developed by the Rocketdyne Division of North American Aviation, Inc. The engine uses liquid oxygen and liquid hydrogen as propellants and has a thrust rating of 225, 000 lb_f at an oxidizer-to-fuel mixture ratio of 5.5. A S-IVB battleship stage was used to supply propellants to the engine. A schematic of the battleship stage with the J-2 engine is shown in Fig. 4.

Listings of major engine components and engine orifices for this test period are presented in Tables I and II (Appendix II), respectively. All engine modifications and component replacements performed since the previous test period are presented in Tables III and IV, respectively. The thrust chamber heater blankets were in place during this test period, although they were not utilized.

2.1.1 J-2 Rocket Engine

The J-2 rocket engine (Figs. 3 and 5, Ref. 2) features the following major components:

1. Thrust Chamber - The tubular-walled, bell-shaped thrust chamber consists of an 18.6-in.-diam combustion chamber (8.0 in. long from the injector mounting to the throat inlet) with a characteristic length (L*) of 24.6 in., a 170.4-in.^{2} throat area, and a divergent nozzle with an expansion ratio of 27.1. Thrust chamber length (from the injector flange to the nozzle exit) is 107 in. Cooling is accomplished by the circulation of engine fuel flow downward from the fuel manifold through 180 tubes and then upward through 360 tubes to the injector.
2. Thrust Chamber Injector - The injector is a concentric-orificed (concentric fuel orifices around the oxidizer post orifices), porous-faced injector. Fuel and oxidizer injector orifice areas are 25.0 and 16.0 in.^{2}, respectively. The porous material, forming the injector face, allows approximately 3.5 percent of total fuel flow to transpiration cool the face of the injector.
3. Augmented Spark Igniter - The augmented spark igniter unit is mounted on the thrust chamber injector and supplies the initial energy source to ignite propellants in the main combustion chamber. The augmented spark igniter chamber is an integral part of the thrust chamber injector. Fuel and oxidizer are ignited in the combustion area by two spark plugs.
4. Fuel Turbopump - The turbopump is composed of a two-stage turbine-stator assembly, an inducer, and a seven-stage, axial-flow pump. The pump is self lubricated and nominally produces, at rated conditions, a head rise of 35,517 ft (1225 psia) of liquid hydrogen at a flow rate of 8414 gpm for a rotor speed of 26,702 rpm.
5. Oxidizer Turbopump - The turbopump is composed of a two-stage turbine-stator assembly and a single-stage centrifugal pump. The pump is self lubricated and nominally produces, at rated conditions, a head rise of 2117 ft (1081 psia) of liquid oxygen at a flow rate of 2907 gpm for a rotor speed of 8572 rpm.
6. Gas Generator - The gas generator consists of a combustion chamber containing two spark plugs, a pneumatically operated control valve containing oxidizer and fuel poppets, and an injector assembly. The oxidizer and fuel poppets provide a fuel

lead to the gas generator combustion chamber. The high energy gases produced by the gas generator are directed to the fuel turbine and then to the oxidizer turbine (through the turbine crossover duct), before being exhausted into the thrust chamber at an area ratio (A/A_t) of approximately 11.

7. Propellant Utilization Valve - The motor-driven propellant utilization valve is mounted on the oxidizer turbopump and bypasses liquid oxygen from the discharge to the inlet side of the pump to vary engine mixture ratio.
8. Propellant Bleed Valves - The pneumatically operated fuel and oxidizer bleed valves provide pressure relief for the boiloff of propellants trapped between the static test stage prevalves and main propellant valves at engine shutdown.
9. Integral Hydrogen Start Tank and Helium Tank - The integral tanks consist of a 7258-in.^{3} sphere for hydrogen with a 1000-in.^{3} sphere for helium located within it. Pressurized gaseous hydrogen in the start tank provides the initial energy source for spinning the propellant turbopumps during engine start. The helium tank provides a helium pressure supply to the engine pneumatic control system.
10. Oxidizer Turbine Bypass Valve - The pneumatically actuated oxidizer turbine bypass valve provides control of the fuel turbine exhaust gases directed to the oxidizer turbine in order to control the oxidizer-to-fuel turbine spinup relationship. The fuel turbine exhaust gases which bypass the oxidizer turbine are discharged into the thrust chamber.
11. Main Oxidizer Valve - The main oxidizer valve is a pneumatically actuated, two-stage, butterfly-type valve located in the oxidizer high pressure duct between the turbopump and the main injector. The first-stage actuator positions the main oxidizer valve at the 14-deg position to obtain initial thrust chamber ignition; the second-stage actuator ramps the main oxidizer valve full open to accelerate the engine to main-stage operation.
12. Main Fuel Valve - The main fuel valve is a pneumatically actuated butterfly-type valve located in the fuel high pressure duct between the turbopump and the fuel manifold.
13. Pneumatic Control Package - The pneumatic control package controls all pneumatically operated engine valves and purges.
14. Electrical Control Assembly - The electrical control assembly provides the electrical logic required for proper sequencing of engine components during operation.

15. Primary and Auxiliary Flight Instrumentation Packages - The instrumentation packages contain sensors required to monitor critical engine parameters. The packages provide environmental control for the sensors.

2.1.2 S-IVB Battleship Stage

The S-IVB battleship stage is approximately 22 ft in diameter and 49 ft long and has a maximum propellant capacity of 46, 000 lb of liquid hydrogen and 199, 000 lb of liquid oxygen. The propellant tanks, fuel above oxidizer, are separated by a common bulkhead. Propellant prevalves, in the low pressure ducts (external to the tanks) interfacing the stage and the engine, retain propellant in the stage until being admitted into the engine to the main propellant valves and serve as emergency engine shutoff valves. Propellant recirculation pumps in both fuel and oxidizer tanks are utilized to circulate propellants through the low pressure ducts and turbopumps before engine start to stabilize hardware temperatures near normal operating levels and to prevent propellant temperature stratification. Vent and relief valve systems are provided for both propellant tanks.

Pressurization of the fuel and oxidizer tanks was accomplished by facility systems using hydrogen and helium, respectively, as the pressurizing gases. The engine-supplied gaseous hydrogen for fuel tank pressurization during S-IVB flight was routed to the facility vent system.

2.2 TEST CELL

Test Cell J-4, Fig. 2, is a vertically oriented test unit designed for static testing of liquid-propellant rocket engines and propulsion systems at pressure altitudes of 100, 000 ft. The basic cell construction provides a 1.5-million-lbf-thrust capacity. The cell consists of four major components (1) test capsule, 48 ft in diameter and 82 ft in height, situated at grade level and containing the test article; (2) spray chamber, 100 ft in diameter and 250 ft in depth, located directly beneath the test capsule to provide exhaust gas cooling and dehumidification; (3) coolant water, steam, nitrogen (gaseous and liquid), hydrogen (gaseous and liquid), and liquid oxygen and gaseous helium storage and delivery systems for operation of the cell and test article; and (4) control building, containing test article controls, test cell controls, and data acquisition equipment. Exhaust machinery is connected with the spray chamber and maintains a low pressure before and after the engine firing and exhausts the products of combustion from the engine firing.

Before a firing, the facility steam ejector, in series with the exhaust machinery, provides a pressure altitude of 100,000 ft in the test capsule. A detailed description of the test cell is presented in Ref. 3.

The battleship stage and the J-2 engine were oriented vertically downward on the centerline of the diffuser-steam ejector assembly. This assembly consisted of a diffuser duct (20 ft in diameter by 150 ft in length), a centerbody steam ejector within the diffuser duct, a diffuser insert (13.5 ft in diameter by 30 ft in length) at the inlet to the diffuser duct, and a gaseous nitrogen annular ejector above the diffuser insert. The diffuser insert was provided for dynamic pressure recovery of the engine exhaust gases to maintain engine ambient pressure altitude (attained by the steam ejector) during the engine firing. The annular ejector was provided to suppress steam recirculation into the test capsule during steam ejector shutdown. The test cell was also equipped with (1) a gaseous nitrogen purge system for continuously inerting the normal air in-leakage of the cell; (2) a gaseous nitrogen repressurization system for raising test cell pressure, after engine cutoff, to a level equal to spray chamber pressure and for rapid emergency inerting of the capsule; and (3) a spray chamber liquid nitrogen supply and distribution manifold for initially inerting the spray chamber and exhaust ducting and for increasing the molecular weight of the hydrogen-rich exhaust products.

An engine component conditioning system was provided for temperature conditioning engine components. The conditioning system utilized a liquid hydrogen-helium heat exchanger to provide cold helium gas for component conditioning. Engine components requiring temperature conditioning were the thrust chamber, crossover duct, pneumatic regulator, and main oxidizer valve closing control line and second-stage actuator. Helium was routed internally through the crossover duct and tubular-walled thrust chamber and externally over the pneumatic regulator and main oxidizer valve closing control line and second-stage actuator.

2.3 INSTRUMENTATION

Instrumentation systems were provided to measure engine, stage, and facility parameters. The engine instrumentation was comprised of (1) flight instrumentation for the measurement of critical engine parameters and (2) facility instrumentation which was provided to verify the flight instrumentation and to measure additional engine parameters. The flight instrumentation was provided and calibrated and periodically recalibrated at AEDC. Appendix III contains a list of all measured test parameters and the locations of selected sensing points.

Pressure measurements were made using strain-gage-type pressure transducers. Temperature measurements were made using resistance temperature transducers and thermocouples. Oxidizer and fuel turbopump shaft speeds were sensed by magnetic pickup. Fuel and oxidizer flow rates to the engine were measured by turbine-type flowmeters which are an integral part of the engine. The propellant recirculation flow rates were also monitored with turbine-type flowmeters. Engine side loads were measured with dual-bridge, strain-gage-type load cells which were laboratory calibrated before installation. Vibrations were measured by accelerometers mounted on the oxidizer injector dome and on the turbopumps. Primary engine and stage valves were instrumented with linear potentiometers and limit switches.

The data acquisition systems were calibrated by (1) precision electrical shunt resistance substitution for the pressure transducers, load cells, and resistance temperature transducer units; (2) voltage substitution for the thermocouples; (3) frequency substitution for shaft speeds and flowmeters; and (4) frequency-voltage substitution for accelerometers.

The types of data acquisition and recording systems used during this test period were (1) a multiple-input digital data acquisition system (MicroSADIC®) scanning each parameter at 40 samples per second and recording on magnetic tape, (2) single-input, continuous-recording FM systems recording on magnetic tape, (3) photographically recording galvanometer oscillographs, (4) direct-inking, null-balance potentiometer-type X-Y plotters and strip charts, and (5) optical data recorders. Applicable systems were calibrated before each test (atmospheric and altitude calibrations). Television cameras, in conjunction with video tape recorders, were used to provide visual coverage during an engine firing, as well as for replay capability for immediate examination of unexpected events.

2.4 CONTROLS

Control of the J-2 engine, battleship stage, and test cell systems during the terminal countdown was provided from the test cell control room. A facility control logic network was provided to interconnect the engine control system, major stage systems, the engine safety cutoff system, the observer cutoff circuits, and the countdown sequencer. A schematic of the engine start control logic is presented in Fig. 6. The sequence of engine events for a normal start and shutdown is presented in Figs. 7a and b. Two control logics for sequencing the stage prevalves and recirculation systems with engine start for simulating engine flight start sequences are presented in Figs. 7c and d.

SECTION III
PROCEDURE

Preoperational procedures were begun several hours before the test period. All consumable storage systems were replenished, and engine inspections, leak checks, and drying procedures were conducted. Propellant tank pressurants and engine pneumatic and purge gas samples were taken to ensure that specification requirements were met. Chemical analysis of propellants was provided by the propellant suppliers. Facility sequence, engine sequence, and engine abort checks were conducted within a 24-hr time period before an engine firing to verify the proper sequence of events. Facility and engine sequence checks consisted of verifying the timing of valves and events to be within specified limits; the abort checks consisted of electrically simulating engine malfunctions to verify the occurrence of an automatic engine cutoff signal. A final engine sequence check was conducted immediately preceding the test period.

Oxidizer dome, gas generator oxidizer injector, and thrust chamber jacket purges were initiated before evacuating the test cell. After completion of instrumentation calibrations at atmospheric conditions, the test cell was evacuated to approximately 0.5 psia with the exhaust machinery, and instrumentation calibrations at altitude conditions were conducted. Immediately before loading propellants on board the vehicle, the cell and exhaust-ducting atmosphere was inerted. At this same time, the cell nitrogen purge was initiated for the duration of the test period, except for the engine firing. The vehicle propellant tanks were then loaded, and the remainder of the terminal countdown was conducted. Table V presents the engine purges and thermal conditioning operations during the terminal countdown and immediately following the engine firing.

Temperature conditioning of the various engine components was accomplished as required, using the facility-supplied engine component conditioning system. Engine components which required temperature conditioning were the thrust chamber, the crossover duct, pneumatic regulator, main oxidizer valve closing control line, and main oxidizer valve second-stage actuator. The engine component conditioning system utilized a liquid hydrogen-helium heat exchanger to provide the chilled helium for component conditioning.

SECTION IV
RESULTS AND DISCUSSION

4.1 TEST SUMMARY

Two firings of the J-2 rocket engine were conducted during test J4-1801-03 on July 26, 1967, for a total firing duration of 31.3 sec. These firings were in support of the S-IVB/S-V J-2 engine developmental program. Engine components were thermally conditioned to temperatures predicted for S-IVB/S-V first burn and one orbit restart on firings 03A and 03B, respectively. A propellant utilization valve excursion from null to the full-closed position at t_0 + 10 sec was accomplished during firing 03A, effectively changing the engine mixture ratio from 5.0 to 5.5. Firing 03B was conducted with the propellant utilization valve fully open. Test requirements and specific test results are summarized in Table VI. Start and shutdown times of selected engine valves are presented in Table VII. The pump inlets, start tank, and helium tank pressure and temperature conditions at engine start are shown in Fig. 8.

Excessive gas generator outlet temperature peaks were experienced on these firings. This resulted in a gas generator outlet temperature probe failure which produced a premature cutoff on firing 03B. The two remaining scheduled firings for the test period were cancelled. Specific test objectives and a brief summary of results obtained for each firing are presented as follows:

Firing	Test Objectives	Results
03A	S-IVB/S-V first burn; evaluate the effect of thrust chamber resistance on gas generator outlet temperature and augmented spark igniter operation during the start transient at maximum start tank energy	The gas generator outlet initial peak temperature was 2490°F with no second peak. Augmented spark ignition was detected 236 msec after engine start. Post-test inspection revealed that no augmented spark igniter erosion had occurred during this test period.
03B	S-IVB/S-V restart; evaluation of the effect of minimum model specification fuel pump inlet pressure and maximum first orbit starting energy on engine start transient and fuel pump low level stall margin	The gas generator outlet temperature attained 2480°F before failure of the temperature probe, prematurely terminating the firing at t_0 + 1.25 sec. A conservative fuel pump stall margin was maintained during the start transient.

The presentation of the test results in the following sections will consist of a discussion of each engine firing with pertinent comparisons. The data presented will be those recorded on the digital data acquisition system, except as noted.

4.2 TEST RESULTS

4.2.1 Firing J4-1801-03A

The programmed 30-sec engine firing was successfully accomplished. Engine start and shutdown transients are presented in Fig. 9. Table VII presents selected engine valve operating times for engine start and shutdown. Test cell pressure and combustion chamber pressure during the firing are presented in Fig. 10. Pressure altitude at engine start was 101, 000 ft and attained a maximum of 97, 000 ft (geometric pressure altitude, Ref. 4) during main-stage operation. Combustion chamber pressure reflects the propellant utilization valve excursion at about t_0 + 10 sec, which effectively changed the engine mixture ratio from 5. 0 to 5. 5. Thermal conditions of selected engine components are shown in Fig. 11.

Test conditions for firing 03A were selected to repeat conditions for firing 02A, except for thrust chamber resistance to fuel flow (the test variable). These test conditions are compared in Table VIII. The engine was reorificed between tests 02 and 03. The orifice diameters effective for these tests are presented in Table II. Figure 12 presents a plot of the resistance to fuel flow during the fuel lead and gas generator ignition on these firings. It can be seen that firing 03A had lower fuel system resistance as a result of its colder thrust chamber prechill. Both firings (02A and 03A) had 3-sec fuel leads. A comparison of the gas generator ignition transients for these firings (Fig. 13) indicates the gas generator developed higher power and experienced a faster start on firing 02A. The gas generator shutdown transient on firing 03A (Fig. 14) was normal.

The gas generator ignition occurred at t_0 + 0. 669 sec on firing 03A, as compared to t_0 + 0. 642 sec on firing 02A. The lower fuel system resistance on firing 03A produced lower gas generator fuel injector pressure during the start transient. The lower fuel injector pressure and delayed gas generator ignition on firing 03A produced lower gas generator power during the bootstrap transient period. This lower power produced a slower buildup rate of the fuel and oxidizer turbine speeds (Fig. 15). Therefore, a longer time was required for thrust chamber ignition and for combustion chamber pressure to attain main-stage operation on firing 03A (indicated by the time required for combustion chamber pressure to attain 550 psia; this time was t_0 + 1.925 sec on firing 03A, as compared to t_0 + 1.854 sec on firing 02A). The gas generator outlet temperature peaked at 2490°F on firing 03A, as compared to 2080°F on firing 02A. The gas generator oxidizer supply line temperature probably contributed to this excessive temperature peak on firing 03A. The line was colder than desired during this test period, as indicated by a comparison of these line temperatures for firings 02A and 03A in Fig. 16. The main oxidizer valve began its second-stage ramp during oxidizer dome prime (t_0 + 1.001 sec); therefore, no gas generator outlet second peak temperature was experienced.

Augmented spark ignition was detected 236 msec after helium control solenoid "on" for firing 03A, as compared to 220 msec on firing 02A. Vibration safety counts (vibration measured in excess of 150 g) were recorded for 35 msec during main chamber ignition. Fuel pump performance (Fig. 17) indicates a conservative stall margin was maintained during the engine start transient.

Engine steady-state performance data are presented in Table IX. The data presented were for a 1-sec data average of test measurements from 29 to 30 sec and were computed using the Rocketdyne PAST 640 modification zero performance computer program. Engine test measurements required by the program and the program computations are presented in Appendix IV. Engine performance for this test was higher than nominal. Normalized data, to standard engine performance values, computed by the program revealed that the gas generator oxidizer supply orifice, the gas generator fuel supply orifice, and the oxidizer turbine bypass orifice were all slightly undersized.

4.2.2 Firing J4-1801-03B

This firing was terminated at t_0 + 1.25 sec by the engine safety cutoff system because the failure of the gas generator outlet temperature probe. This was a new probe which had been installed before test 03. The failure of this probe is attributed to the excessive gas generator temperature peaks experienced during this test period. Engine start and shutdown transients of selected parameters are presented in Fig. 18. Table VII presents selected engine valve operating times for engine start and shutdown. Test cell pressure and combustion chamber pressure during the firing are presented in Fig. 19. Pressure altitude at engine start was 105,000 ft. The thermal conditions of engine components are shown in Fig. 20.

The gas generator ignition and shutdown transients of selected parameters are presented in Fig. 21. The energy added to the start tank gas by the warm turbine components in addition to the power developed by the gas generator was sufficient to produce a fast buildup rate of the oxidizer turbine speed (Fig. 18). This produced hydraulic torque sufficient to delay the beginning of the second-stage ramp of the main oxidizer valve until t_0 + 1.195 sec (194 msec slower than for firing 03A).

The gas generator oxidizer supply line temperature (Fig. 16), similar to the temperature experienced on firing 03A, probably contributed to the excessive gas generator temperature peak and temperature probe failure encountered on this firing.

Fuel pump performance data (Fig. 22) indicate a conservative stall margin was maintained during the start transient. Vibration safety counts were recorded for 9 msec during main chamber ignition.

4.3 POST-TEST INSPECTION

Engine inspection after this test period revealed that no apparent augmented spark igniter erosion had occurred. However, the first stage of the fuel turbine, both stator and wheel blades, was eroded. Also, small cracks were found in the curvic coupling of the fuel turbine first-stage wheel. Rocketdyne approved delaying replacement of the turbopump assembly until after the next test period.

SECTION V
SUMMARY OF RESULTS

The results of the two Rocketdyne J-2 rocket engine firings conducted on July 26, 1967, in Test Cell J-4 are summarized as follows:

1. Increased thrust chamber resistance to fuel flow results in a faster buildup rate of gas generator power and thrust chamber pressure during the bootstrap transient period.
2. Test data indicate the temperature of the gas generator oxidizer supply line was probably the primary factor influencing the excessive initial peak temperatures (above 2400°F) experienced on these firings.
3. For the S-IVB/S-V restart with maximum first orbit start energy (firing 03B), the beginning of the main oxidizer valve

second-stage ramp was 194 msec slower than for the S-IVB/S-V first burn with maximum start energy (firing 03A).

4. No gas generator outlet second peak temperature was experienced for the S-IVB/S-V first burn firing with maximum starting energy (firing 03A).
5. A conservative fuel pump stall margin was maintained during the engine start transients of this test period.

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APPENDIXES

I. ILLUSTRATIONS
II. TABLES
III. INSTRUMENTATION
IV. METHODS OF CALCULATIONS (PERFORMANCE PROGRAM)

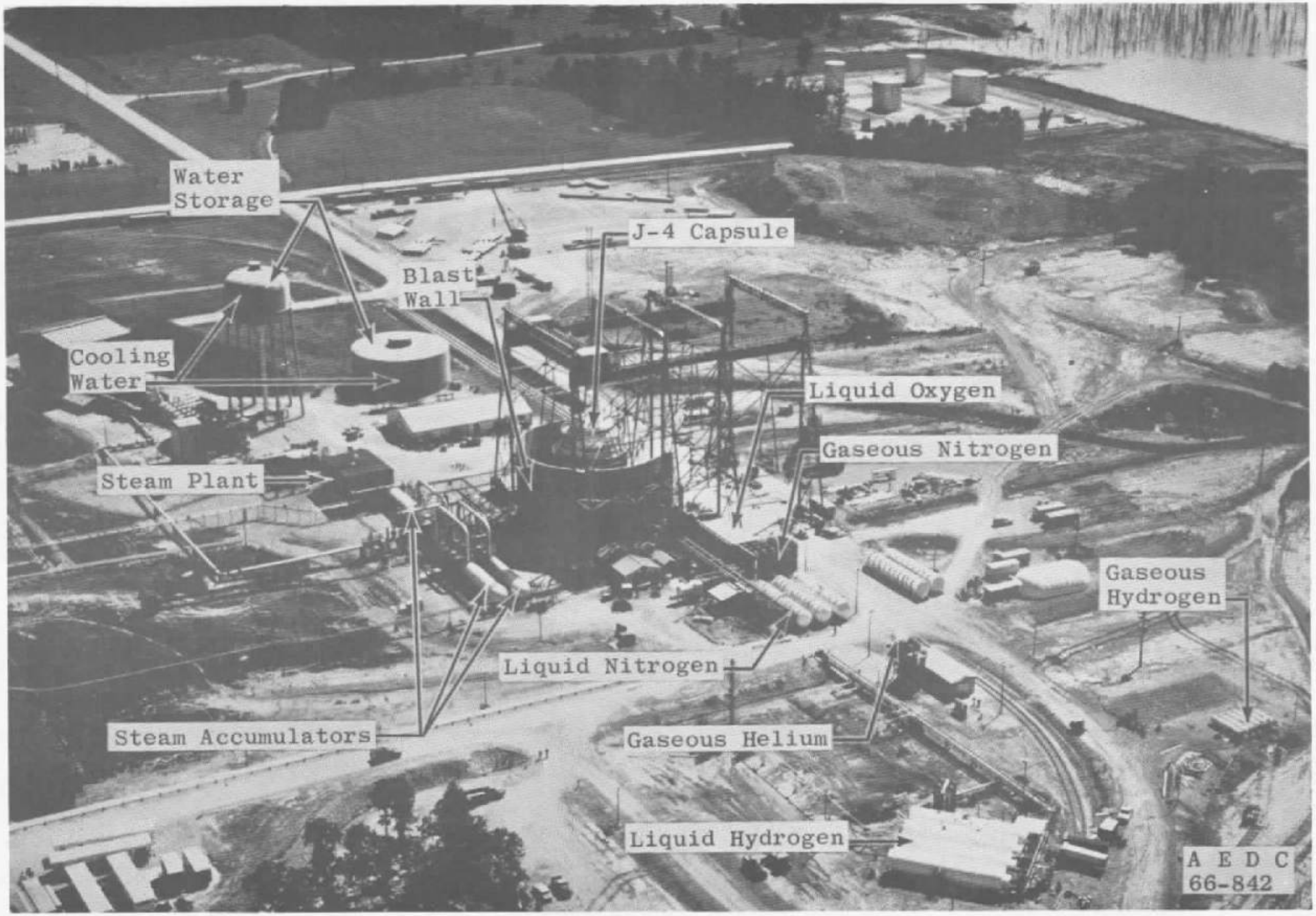


Fig. 1 Test Cell J-4 Complex

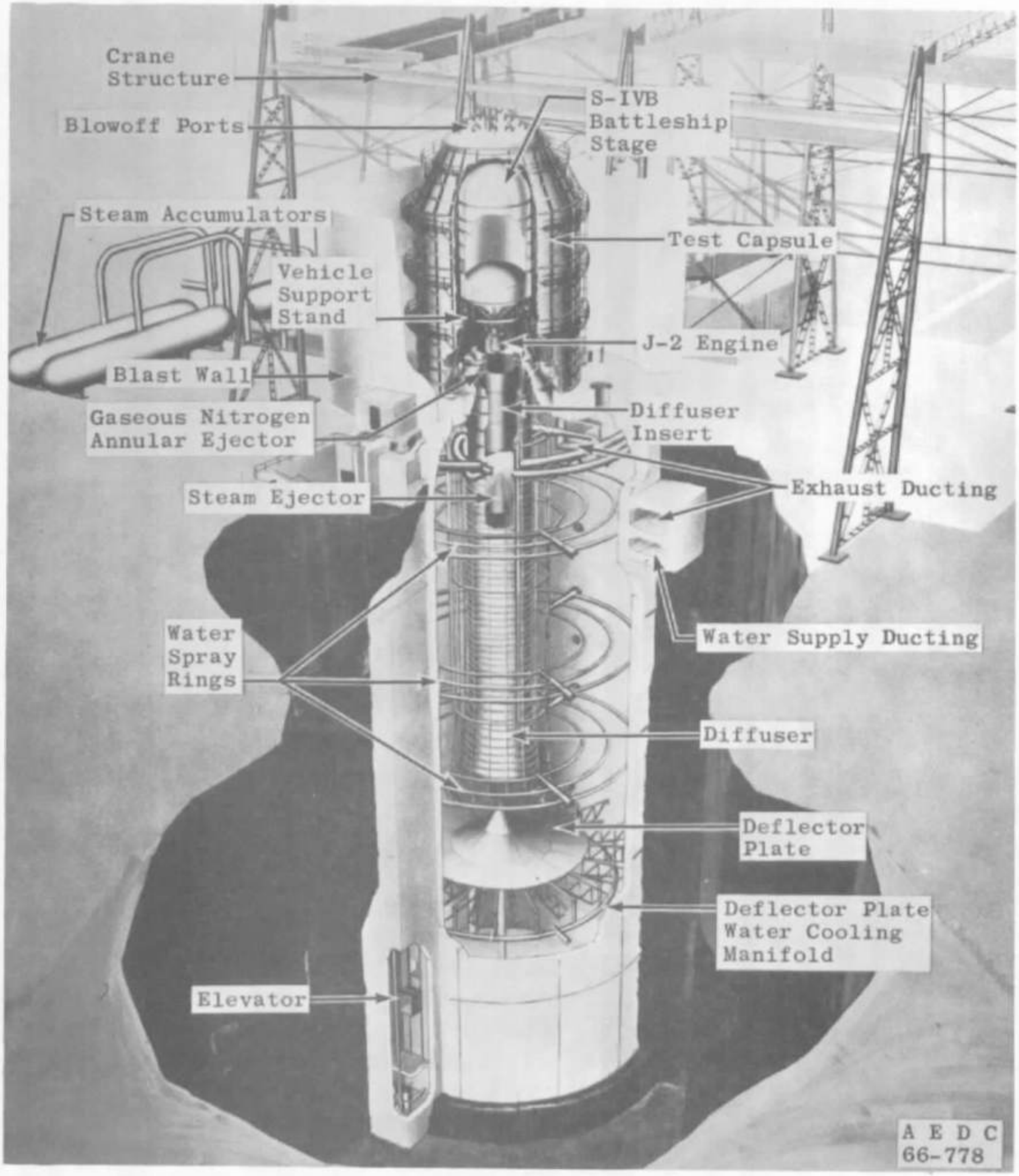


Fig. 2 Test Cell J-4, Artist's Conception

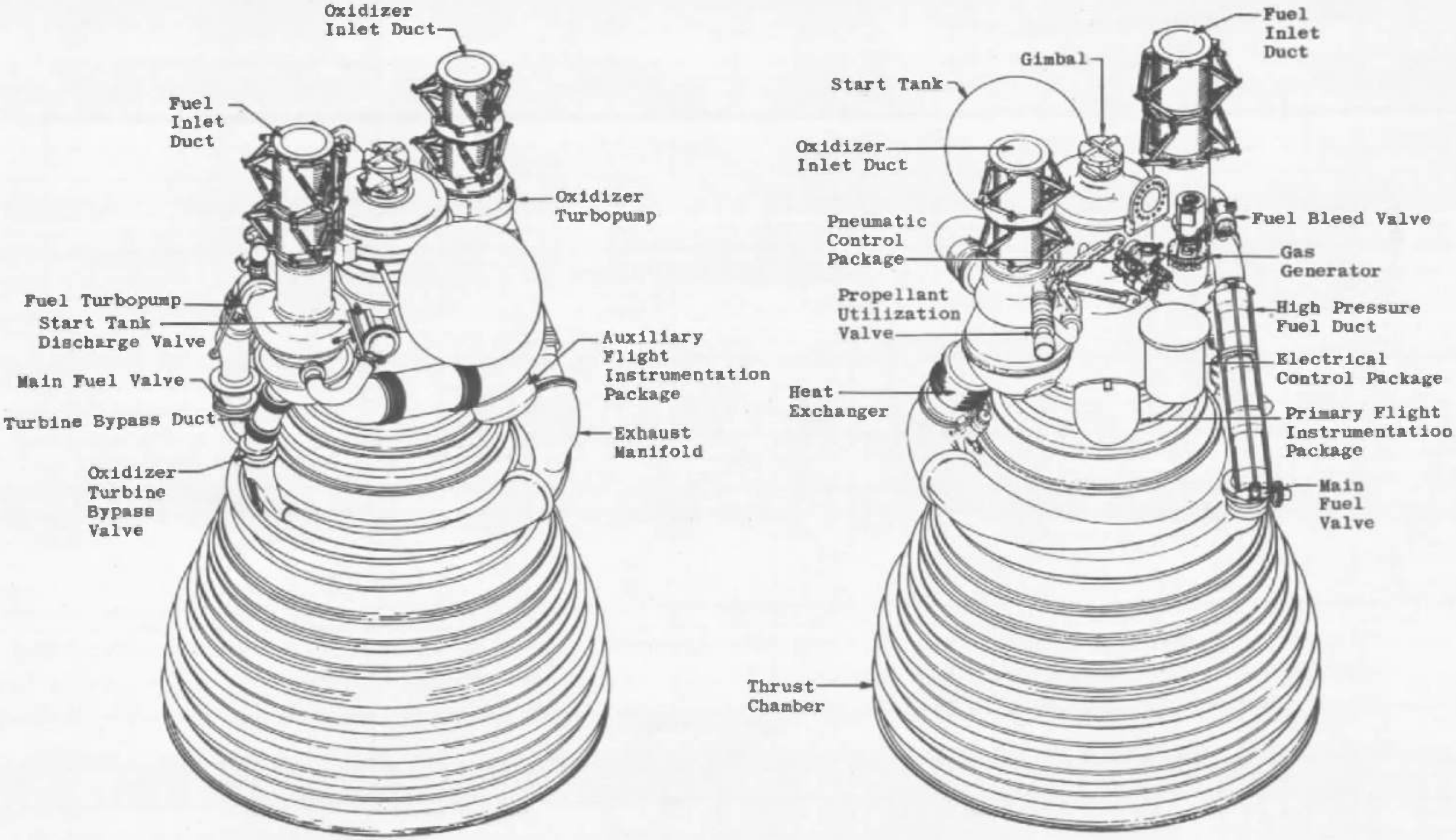


Fig. 3 Engine Details

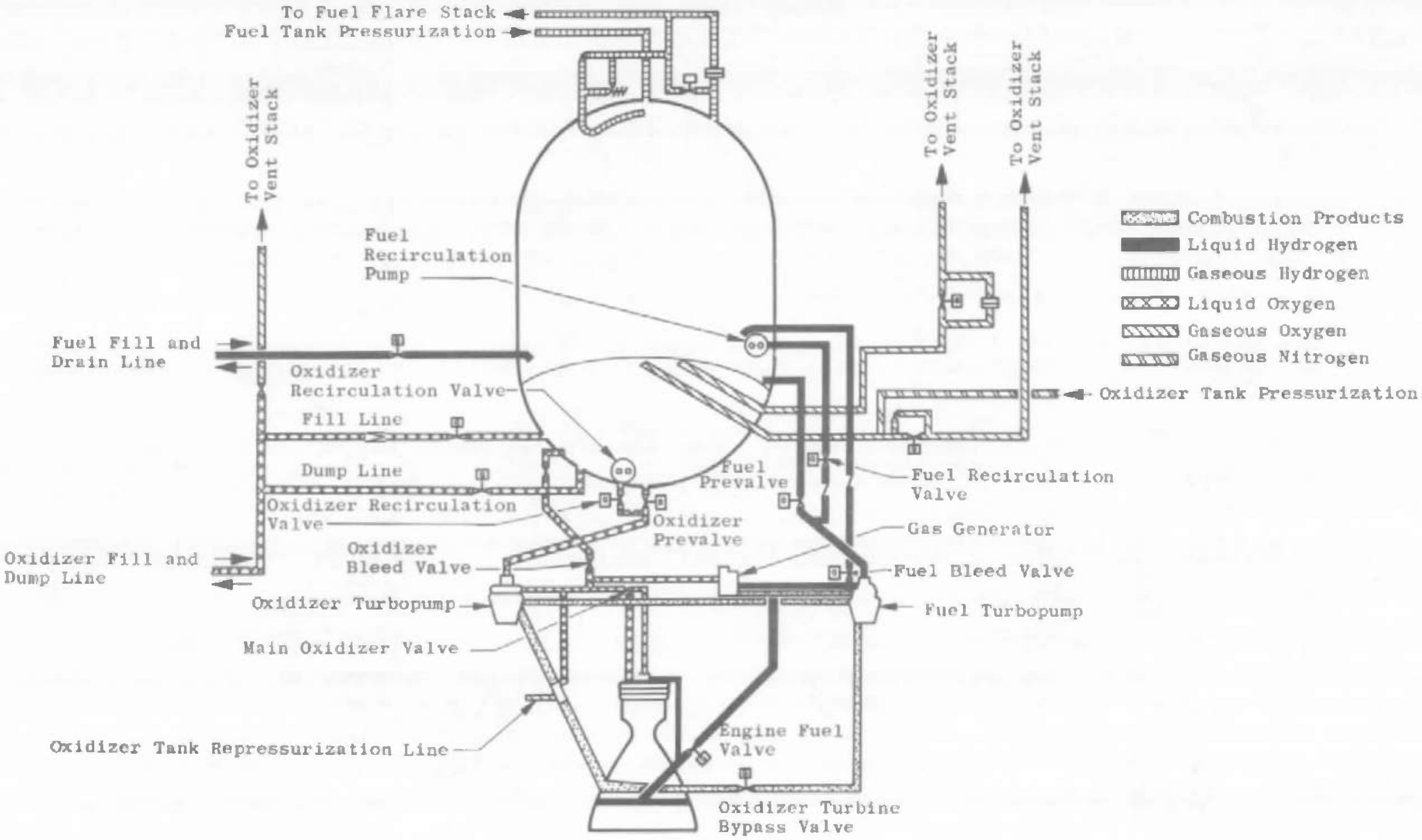


Fig. 4 S-IVB Battleship Stage/J-2 Engine Schematic

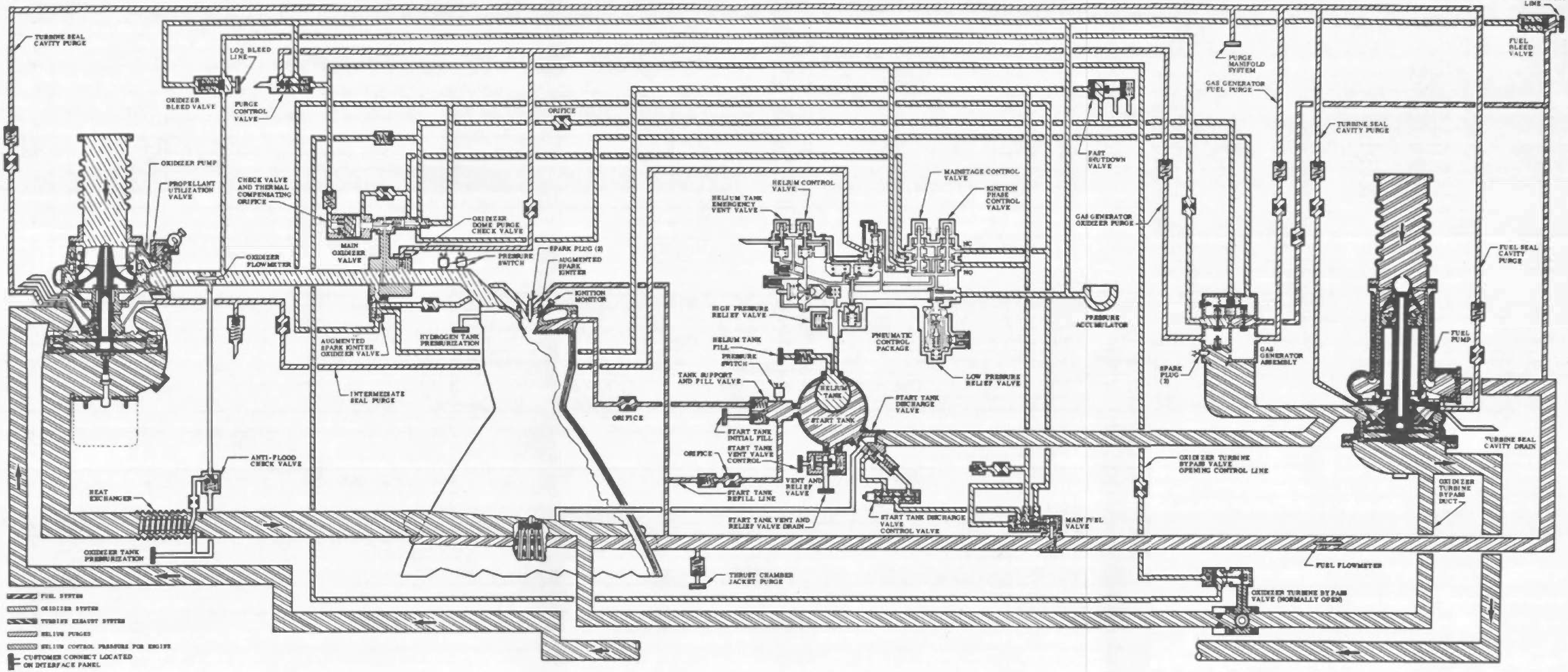


Fig. 5 Engine Schematic

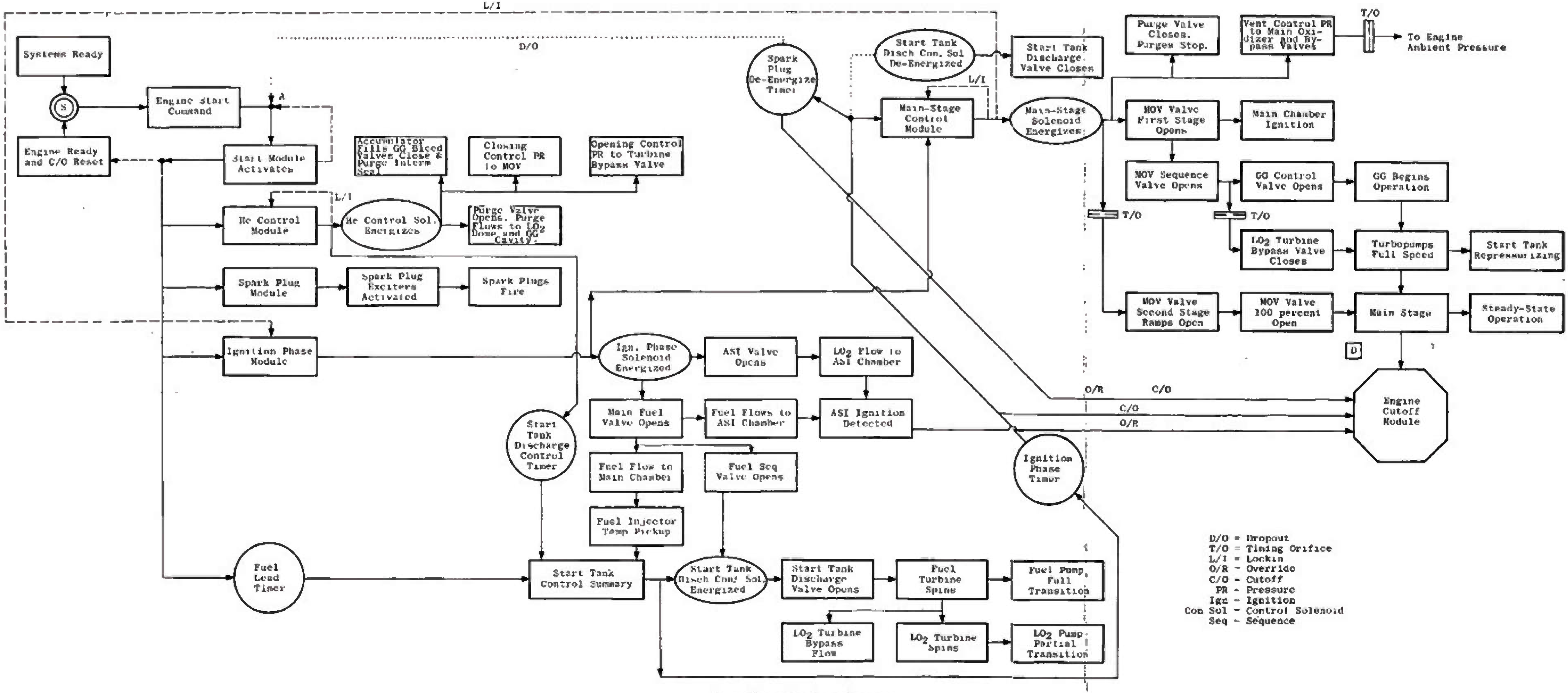
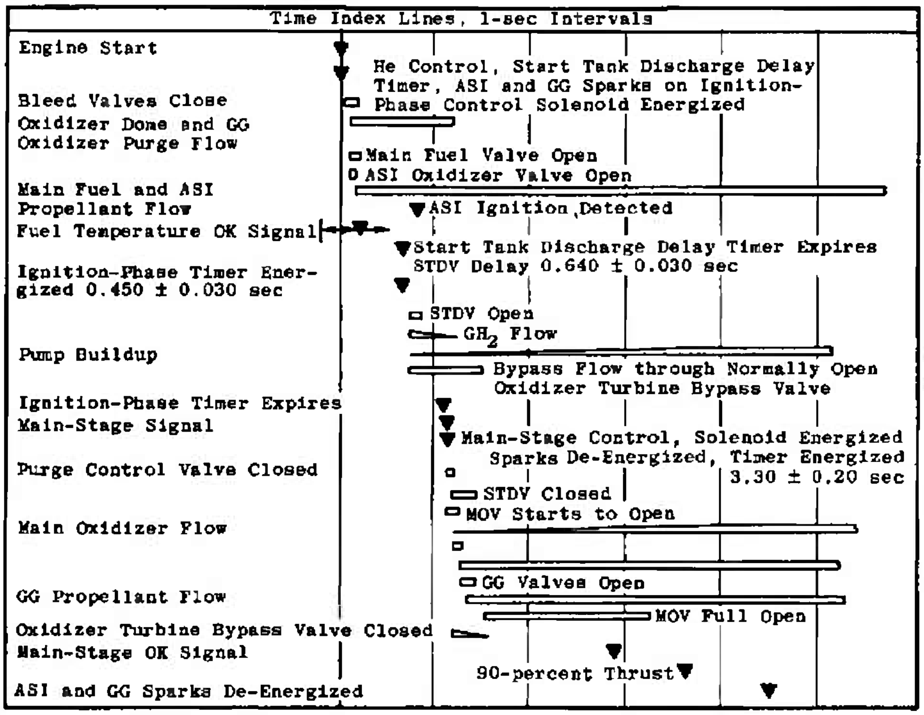
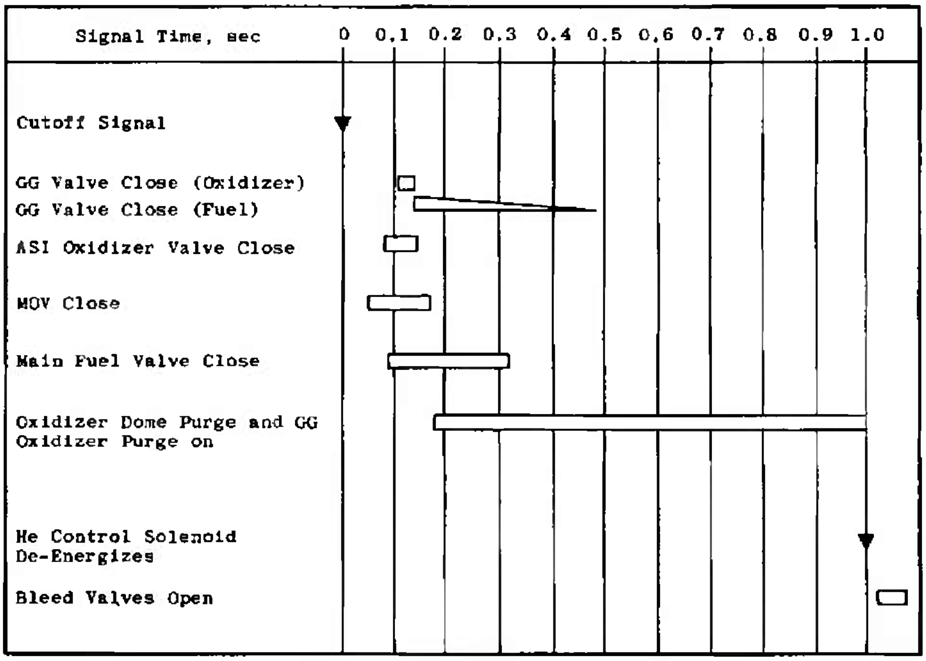


Fig. 6 Engine Start Logic Schematic

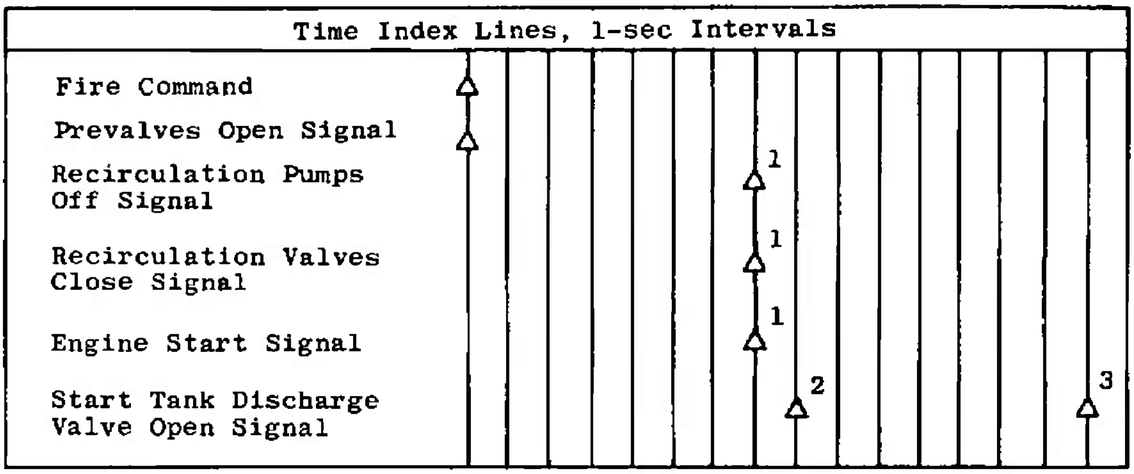


a. Start Sequence



b. Shutdown Sequence

Fig. 7 Engine Start and Shutdown Sequence

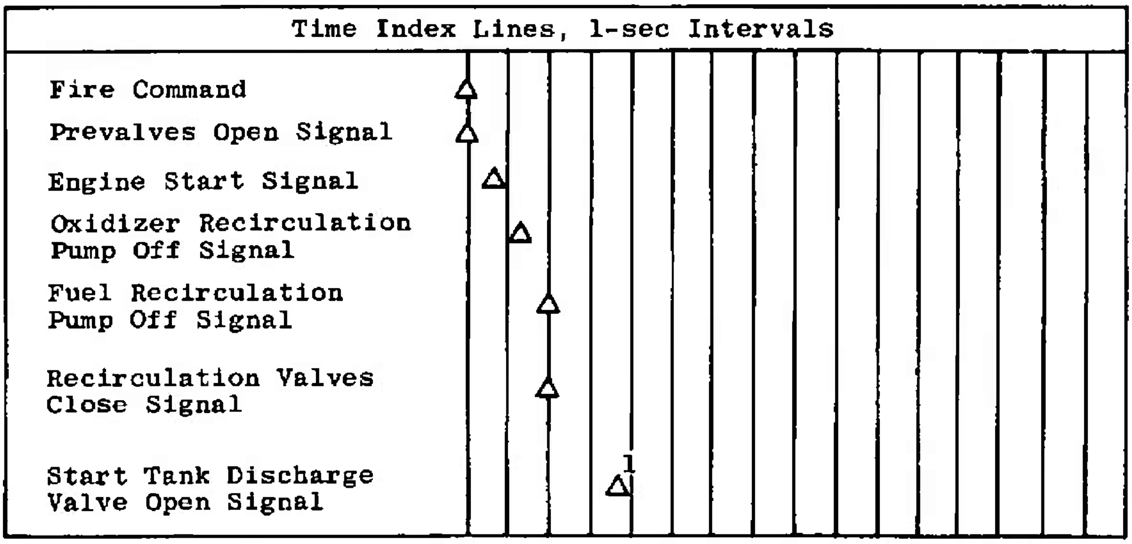


[1] Nominal Occurrence Time (Function of Prevalves Opening Time)

[2] One-sec Fuel Lead (S-II/S-V and S-IVB/S-IB)

[3] Eight-sec Fuel Lead (S-IVB/S-V and S-IB Orbital Restart)

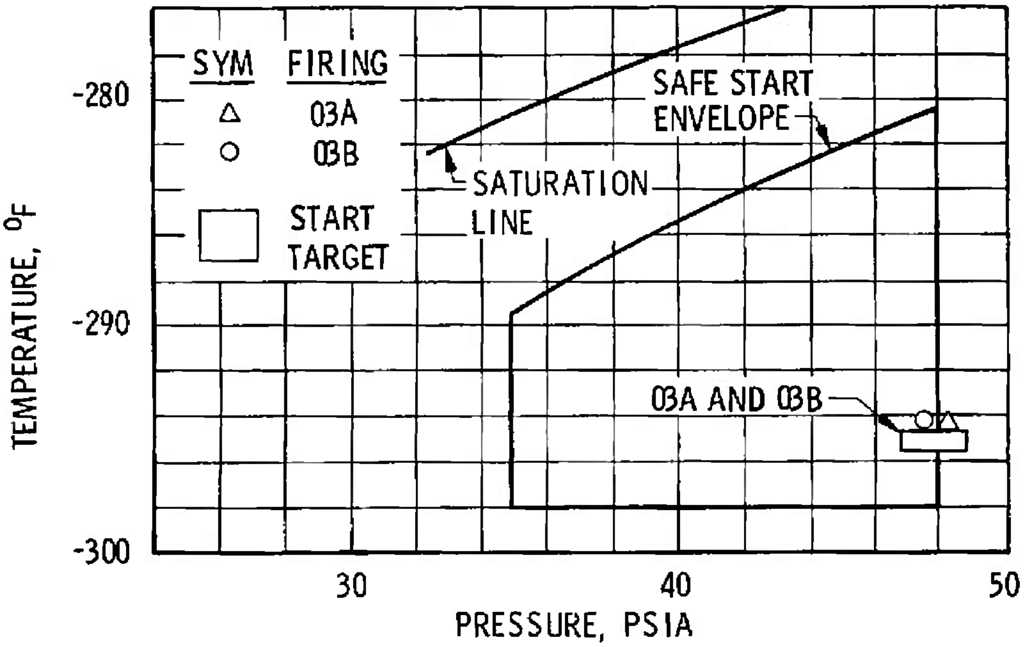
c. Normal Logic Start Sequence



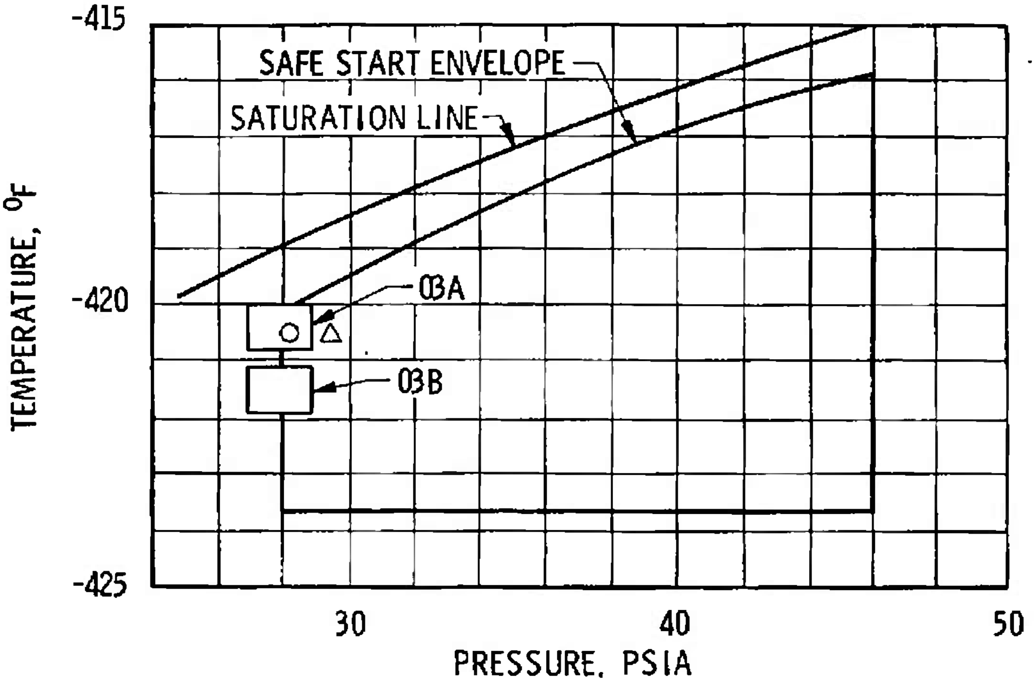
[1] Three-sec Fuel Lead (S-IVB/S-V First Burn)

d. Auxiliary Logic Start Sequence

Fig. 7 Concluded

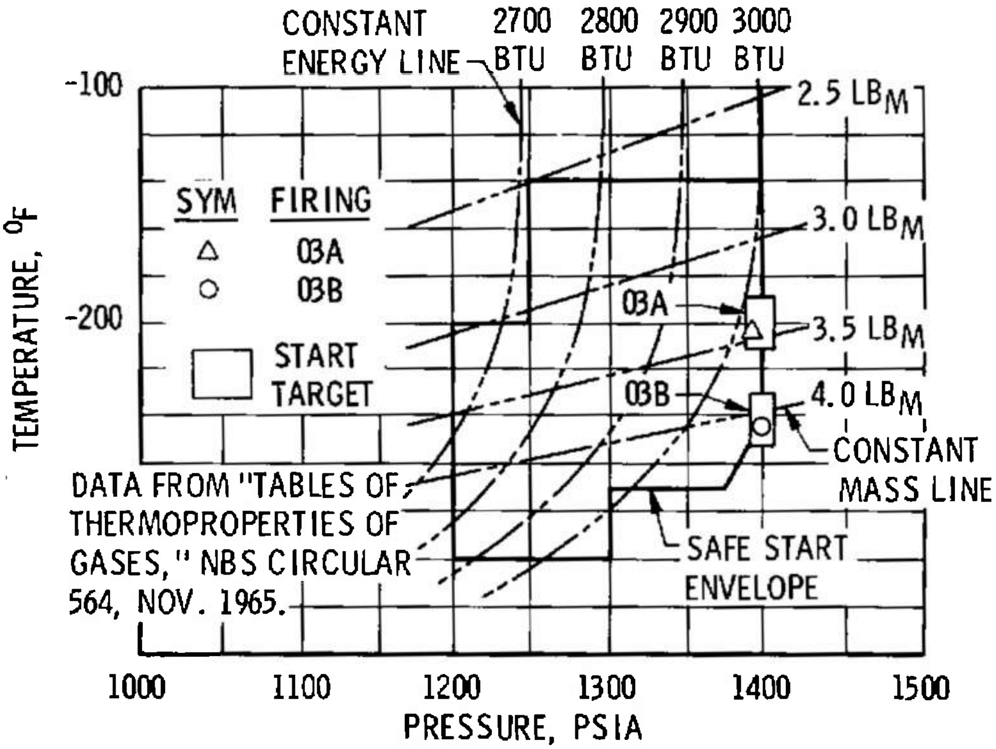


a. Oxidizer Pump Inlet

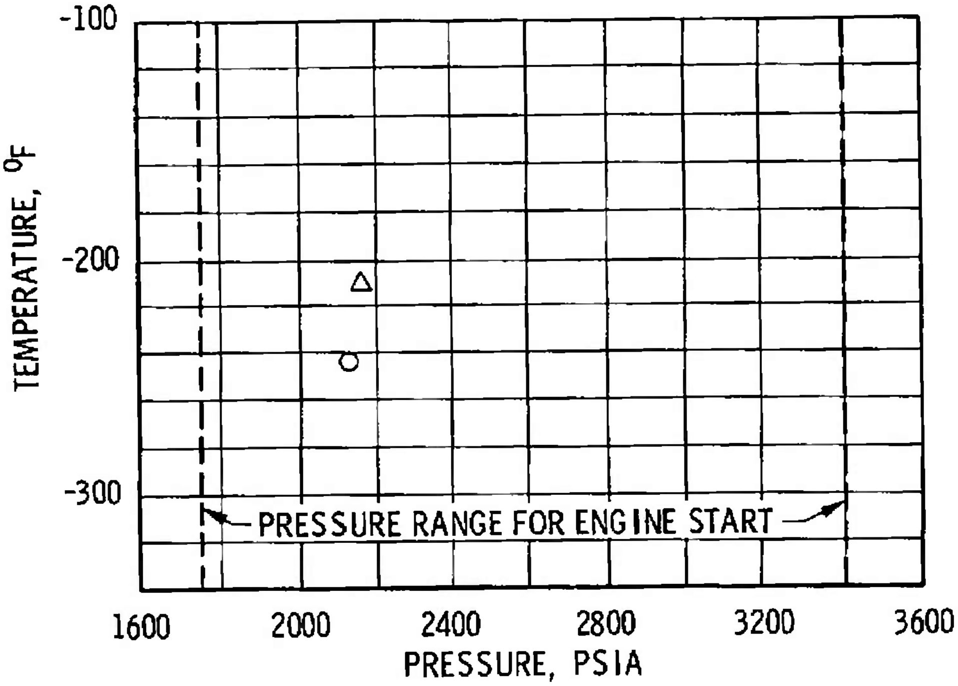


b. Fuel Pump Inlet

Fig. 8 Engine Start Conditions for Pump Inlets, Start Tank, and Helium Tank

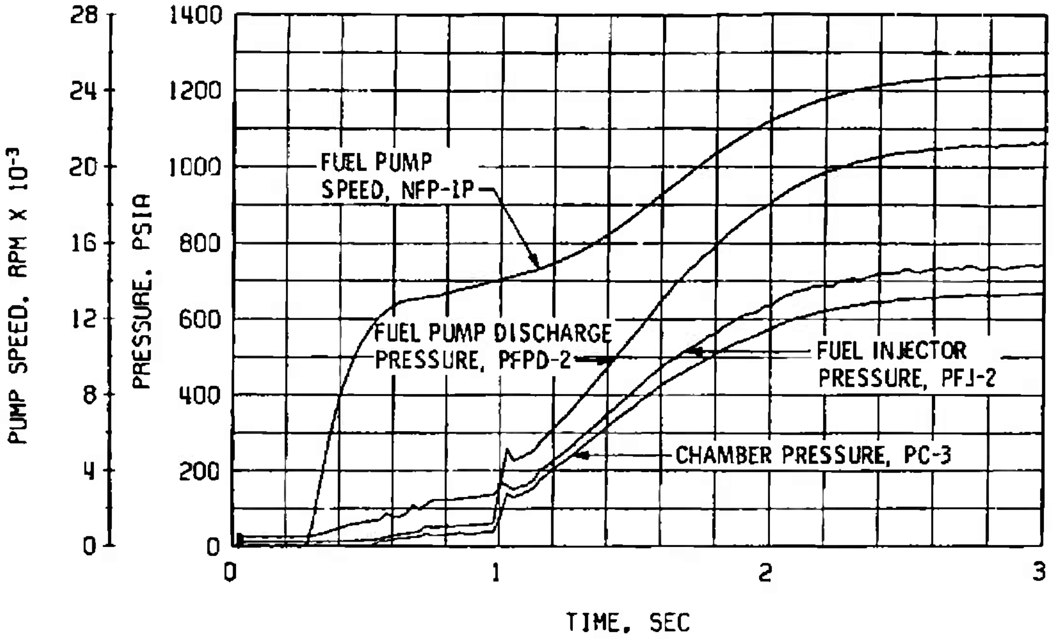


c. Start Tank

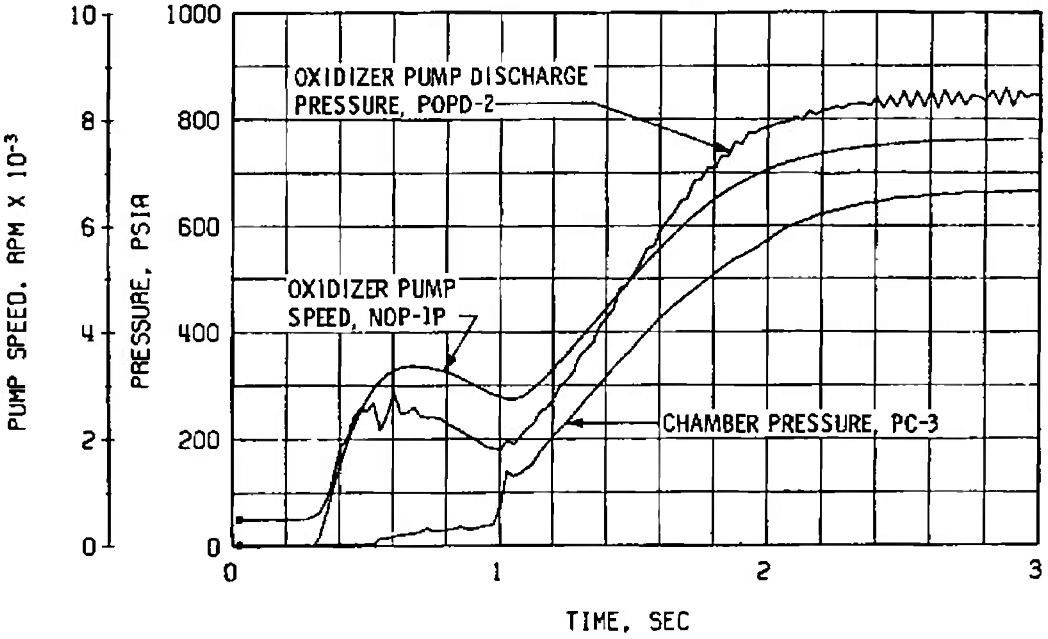


d. Helium Tank

Fig. 8 Concluded



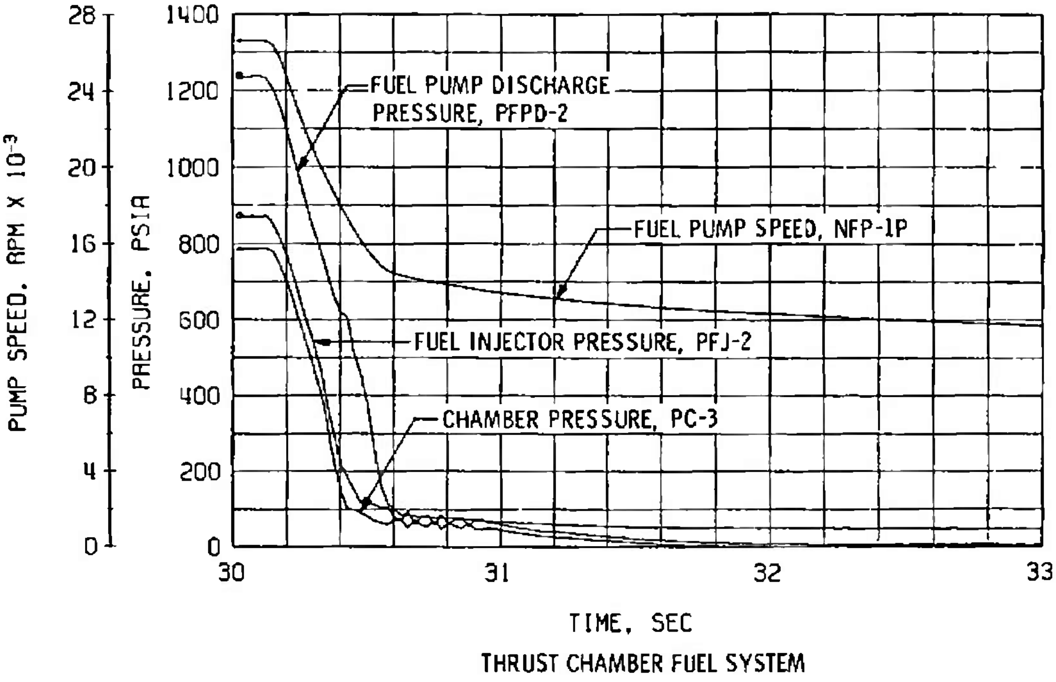
THRUST CHAMBER FUEL SYSTEM

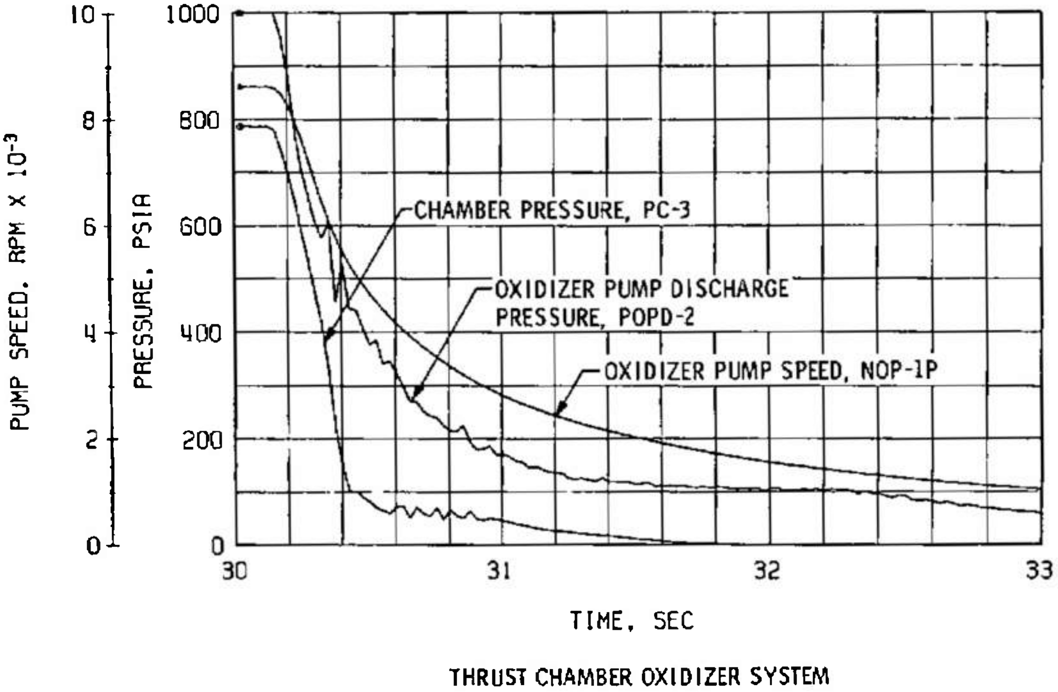


THRUST CHAMBER OXIDIZER SYSTEM

a. Start Transient

Fig. 9 Engine Transient Operation, Firing 03A





b. Shutdown Transient

Fig. 9 Concluded

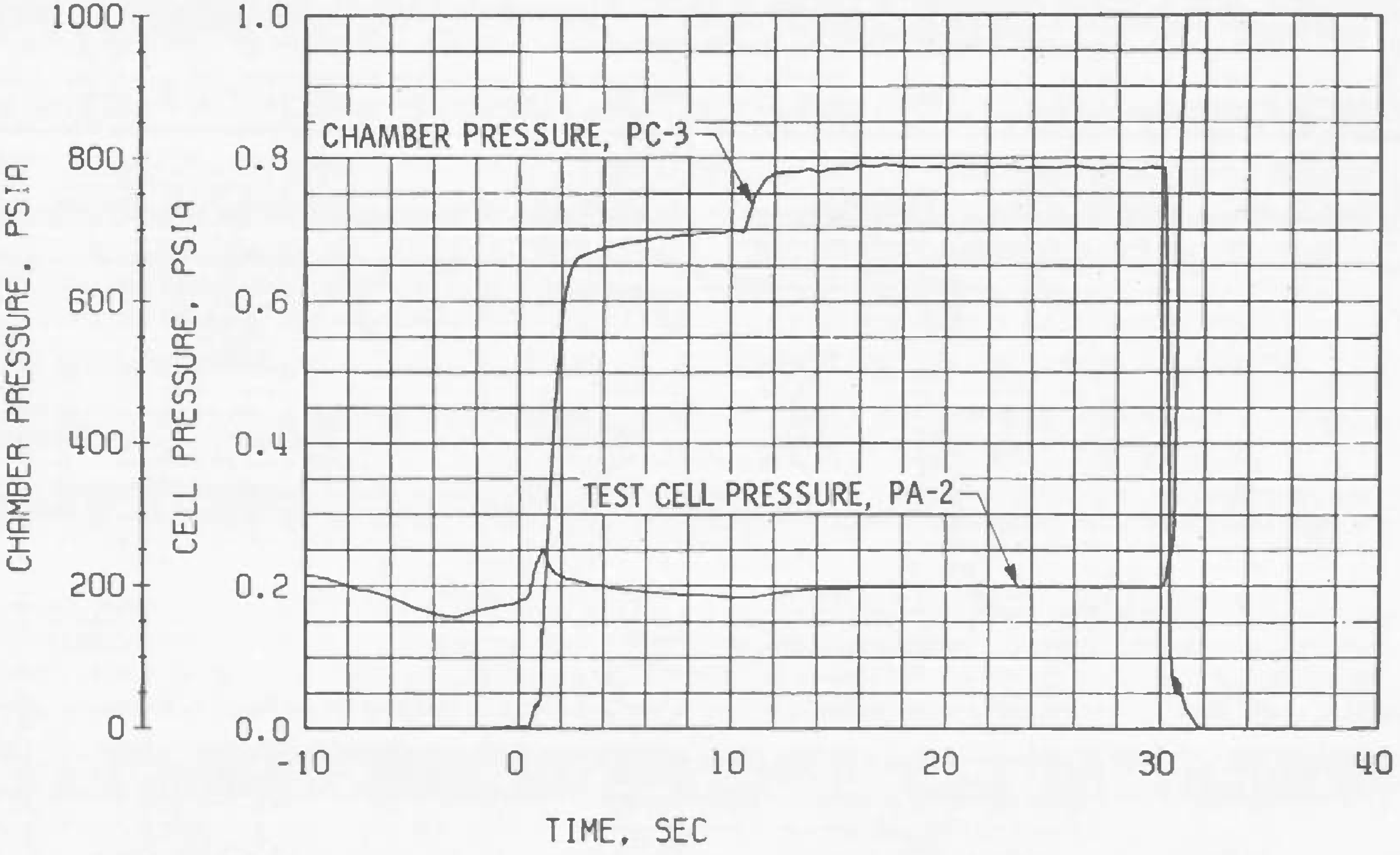
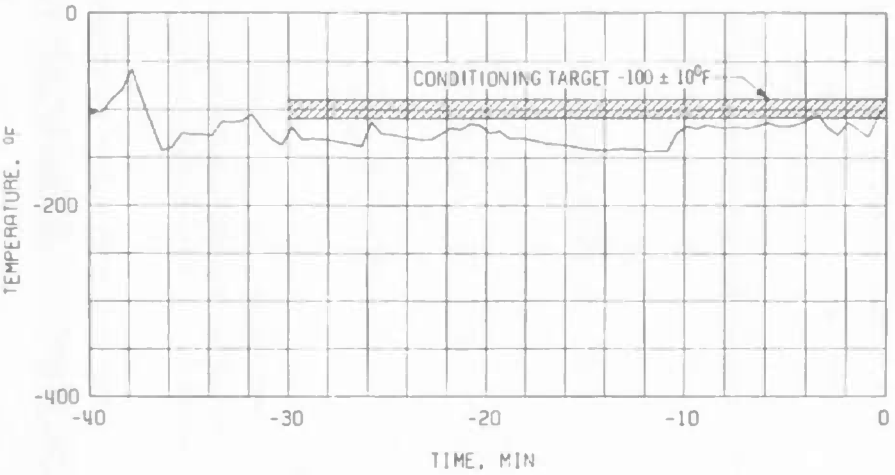
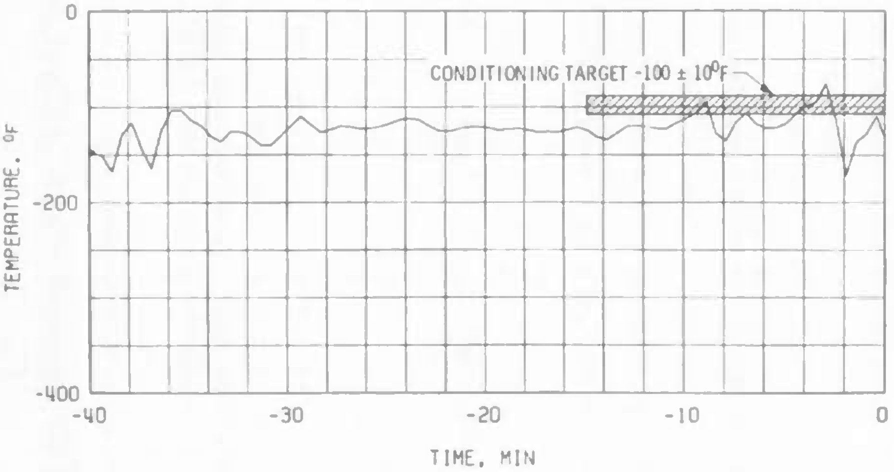


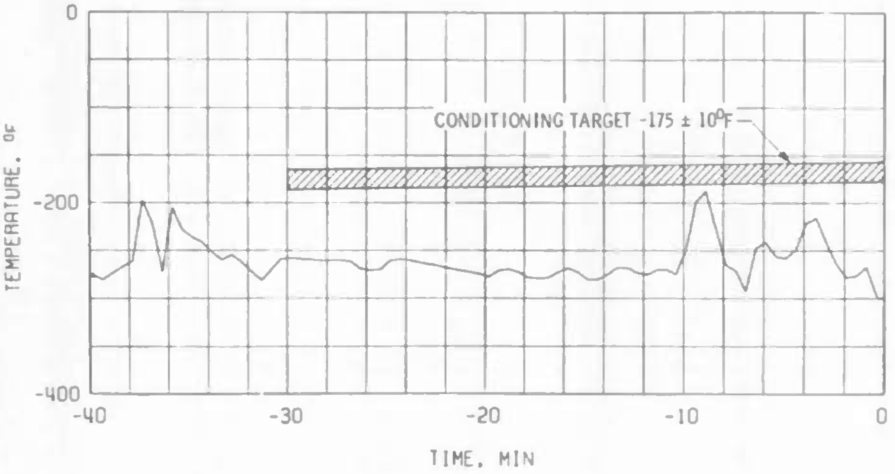
Fig. 10 Engine Ambient and Combustion Chamber Pressures, Firing 03A



a. Pneumatic Regulator, TBHR-2

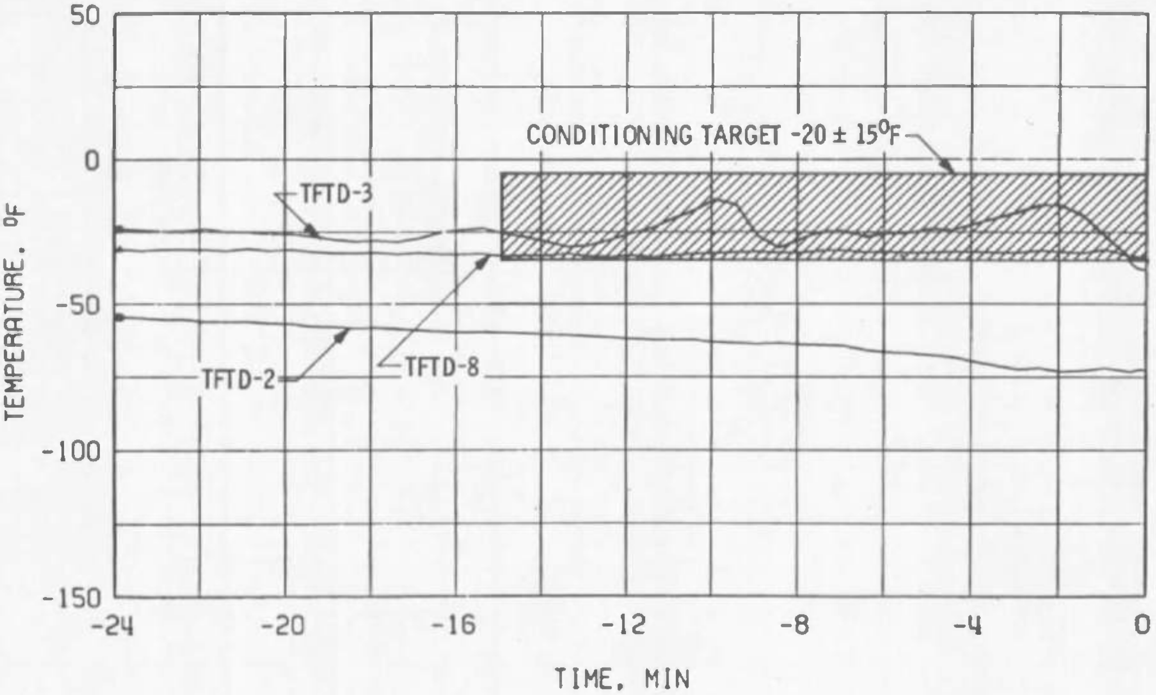


b. Main Oxidizer Closing Control Line Temperature, TSOVAL-2

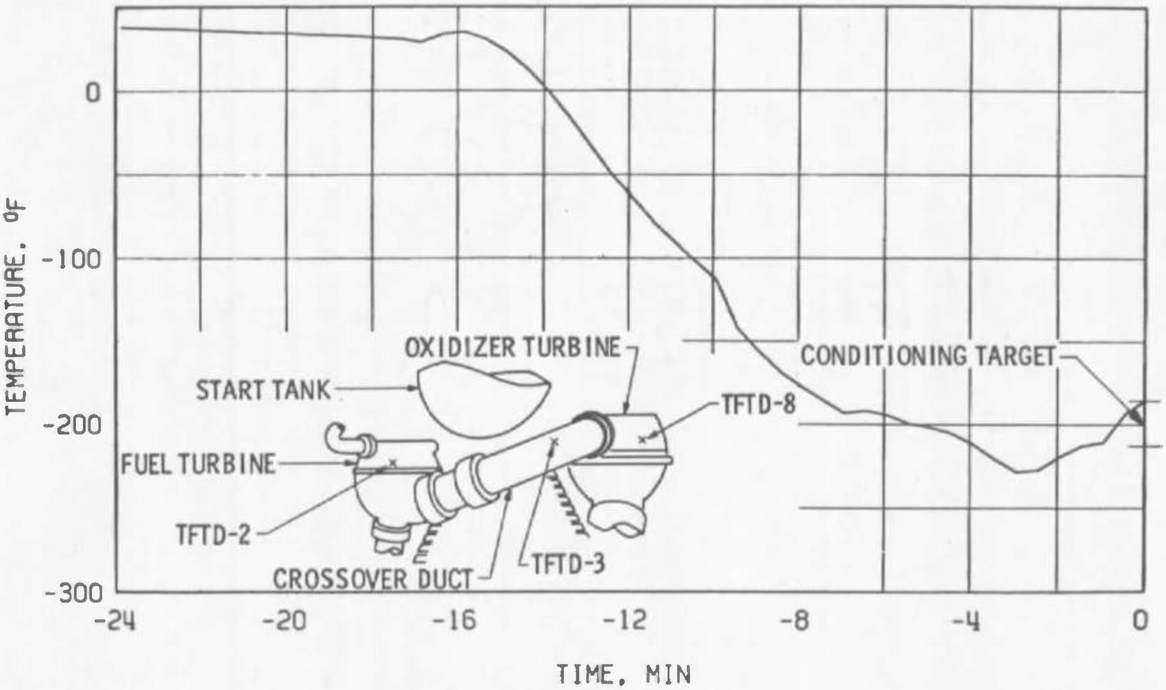


c. Main Oxidizer Valve Second-Stage Actuator, TSOVC-2

Fig. 11 Thermal Conditioning History of Engine Components, Pre-Fire 03A



d. Crossover Duct, TFTD



e. Thrust Chamber Throat, TSC2-19

Fig. 11 Concluded

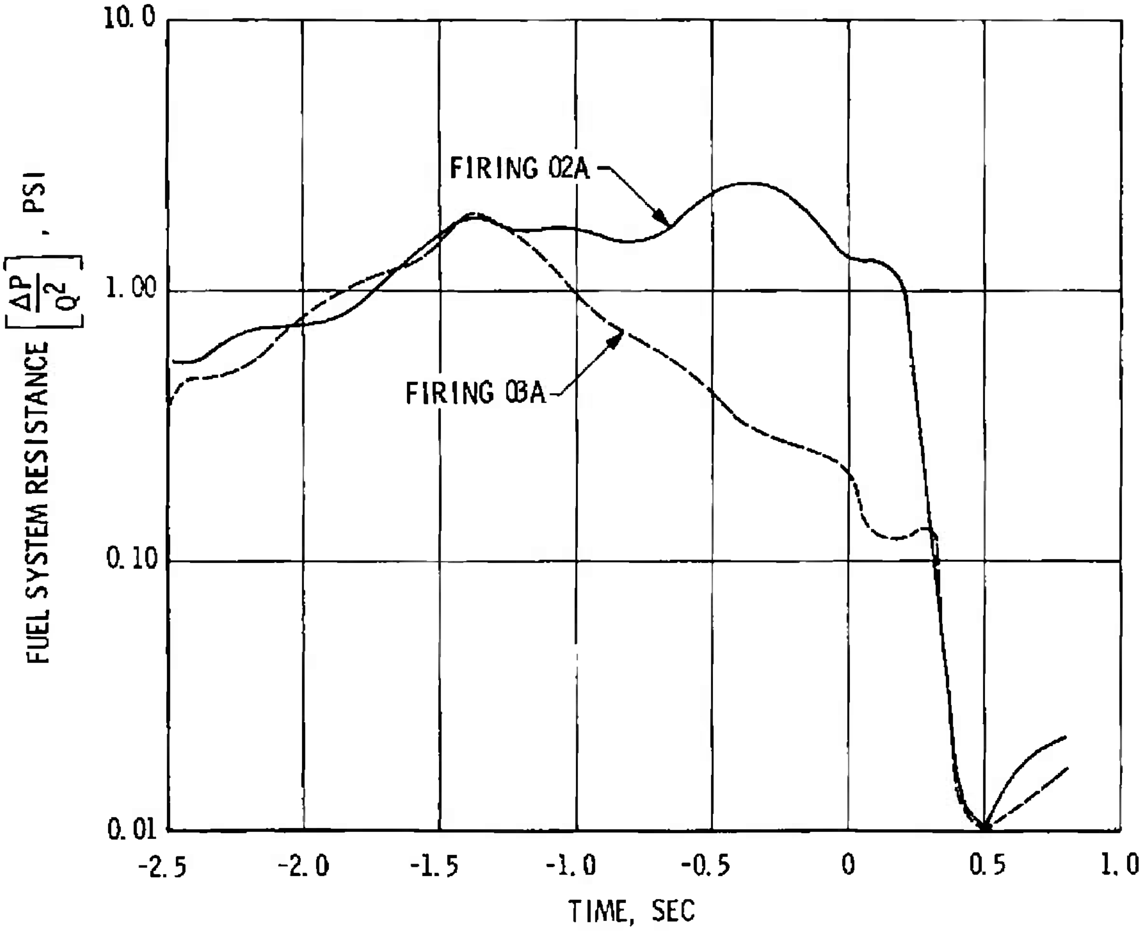
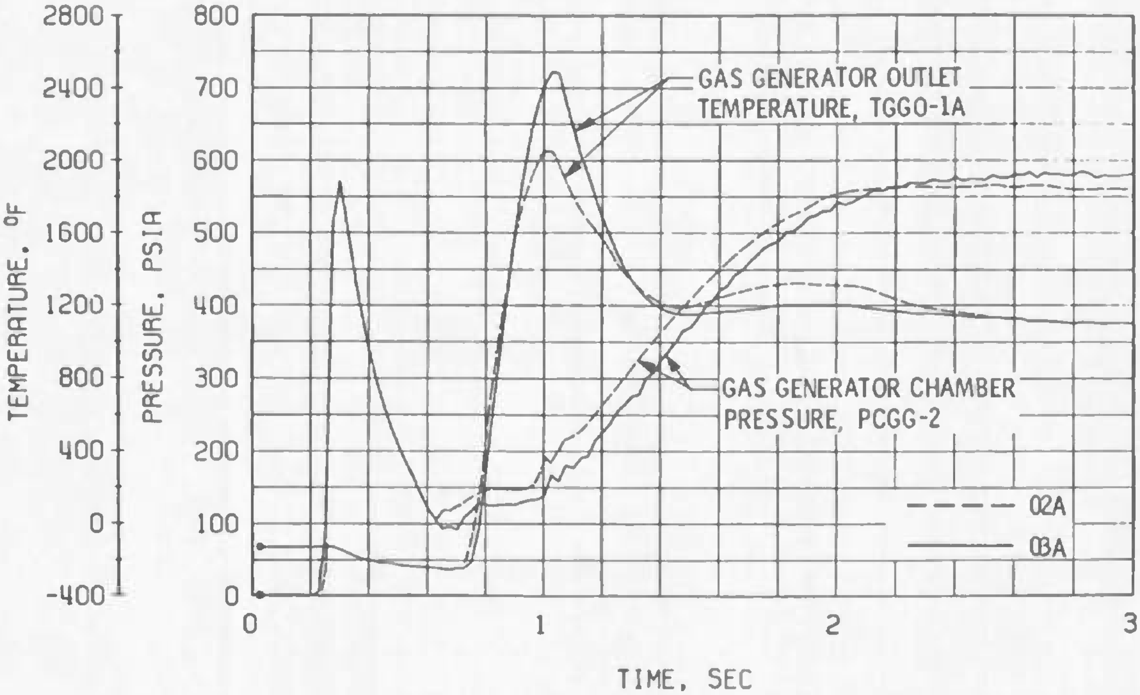
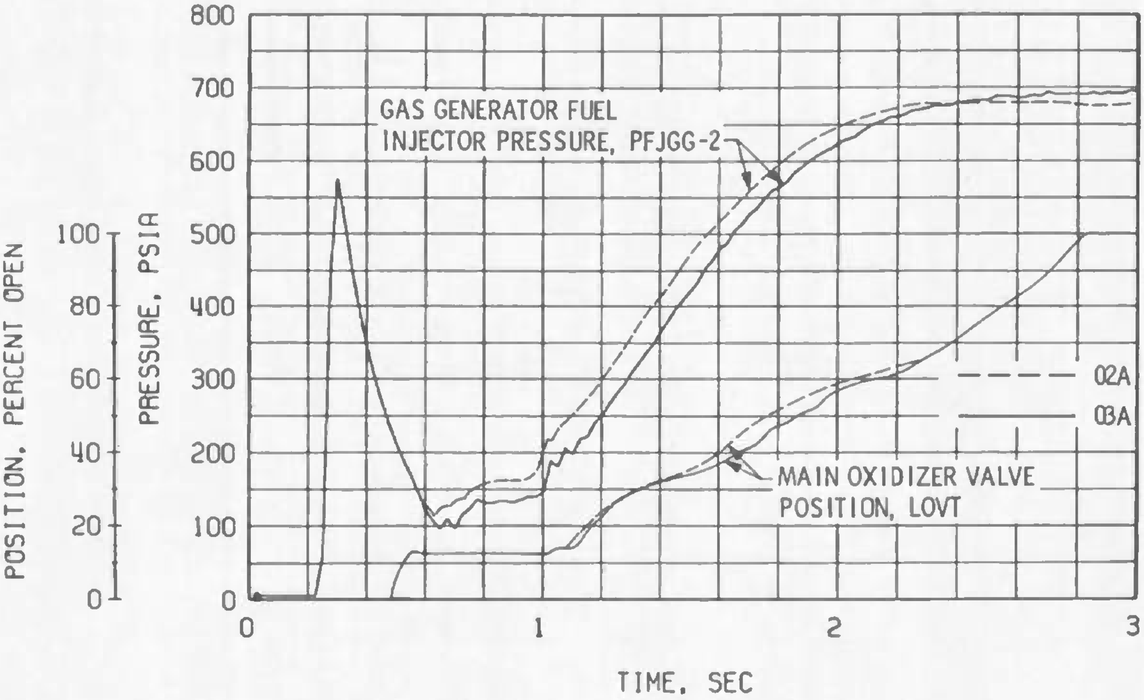


Fig. 12 Fuel System Resistance during Fuel Lead and Gas Generator Ignition, Firings 02A and 03A



a. Gas Generator Outlet Temperature and Chamber Pressure



b. Gas Generator Fuel Injector Pressure and Main Oxidizer Valve Position

Fig. 13 Gas Generator Start Transient Comparison of Firings 02A and 03A

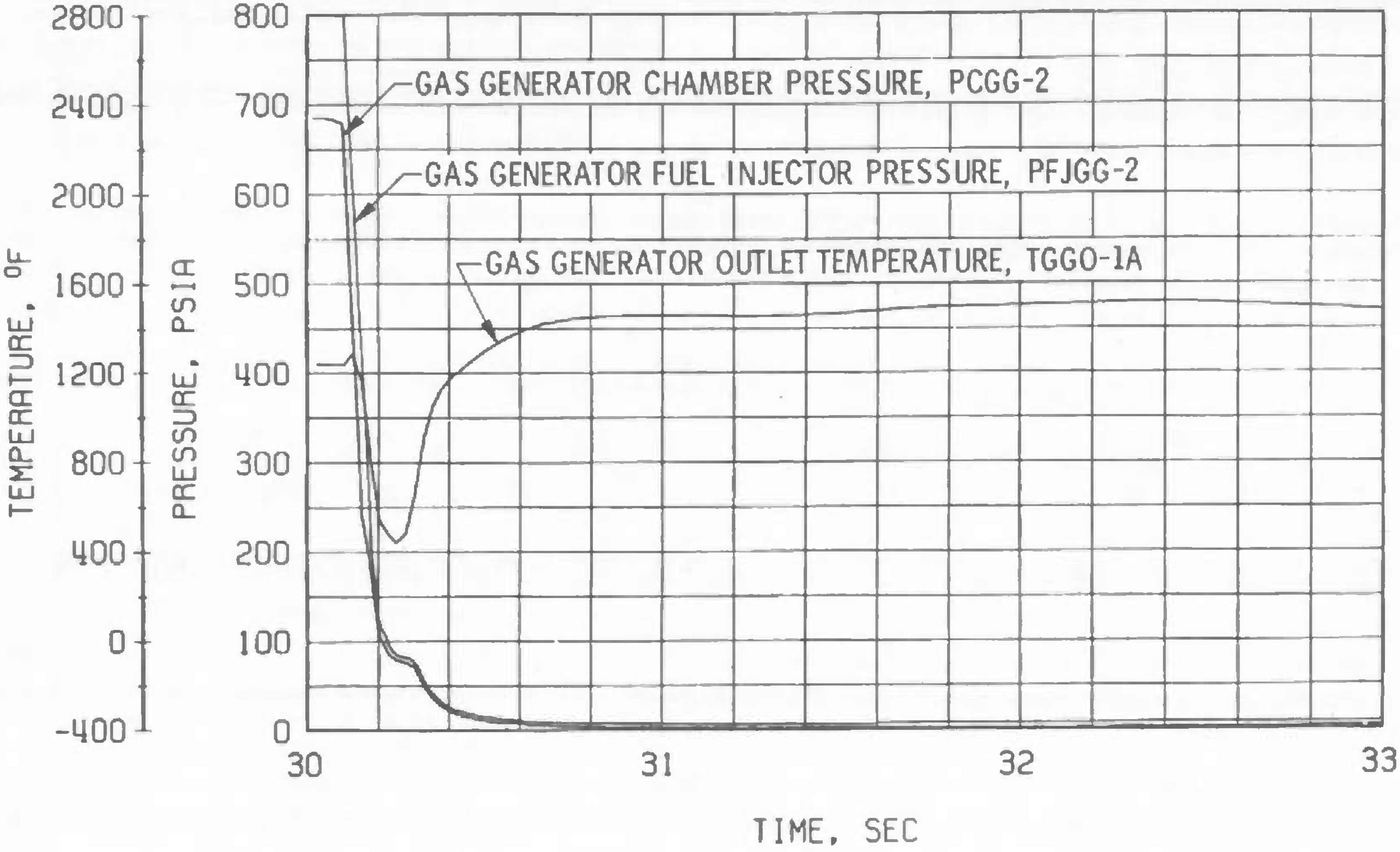
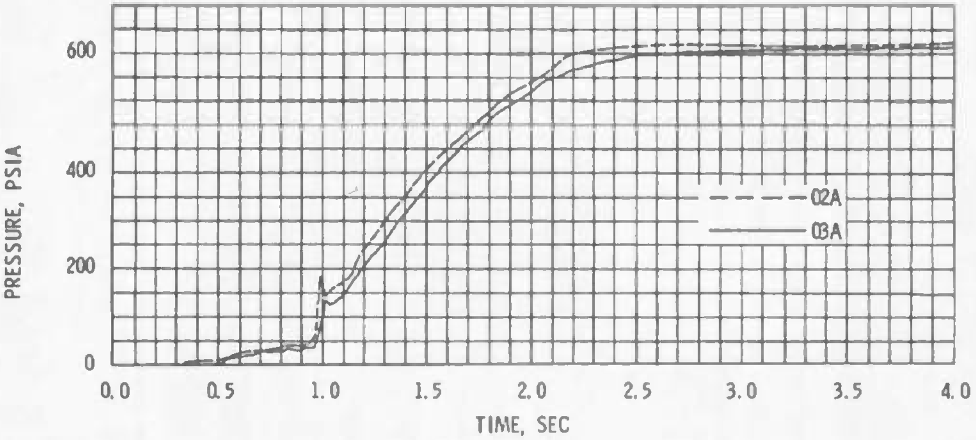
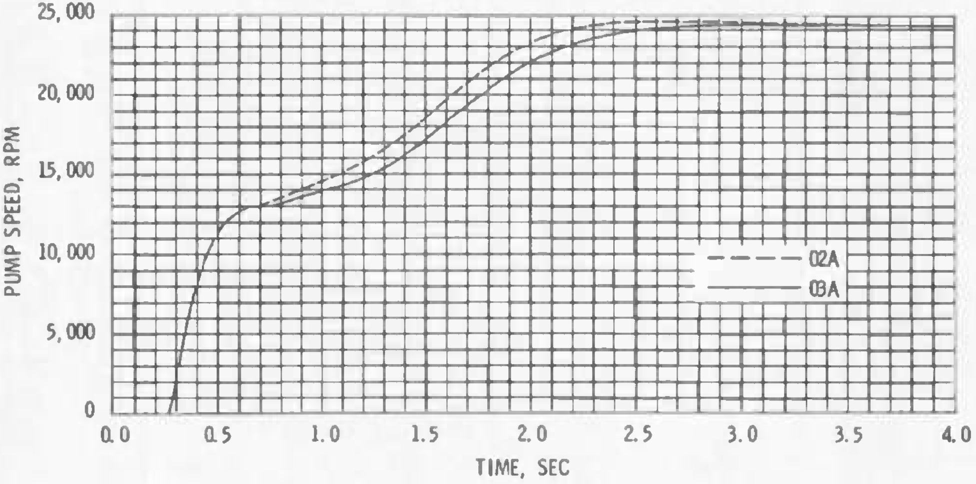


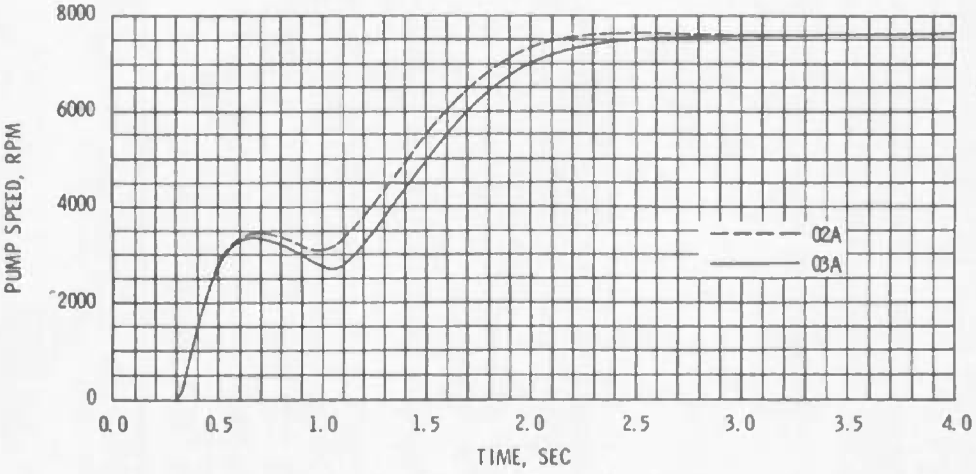
Fig. 14 Gas Generator Shutdown Transient, Firing 03A



a. Combustion Chamber Pressure, PC-3



b. Fuel Turbine Speed, NFP-1P



c. Oxidizer Turbine Speed, NOP-1P

Fig. 15 Comparison of Fuel and Oxidizer Turbine Speed and Chamber Pressure Buildup on Firings 02A and 03A

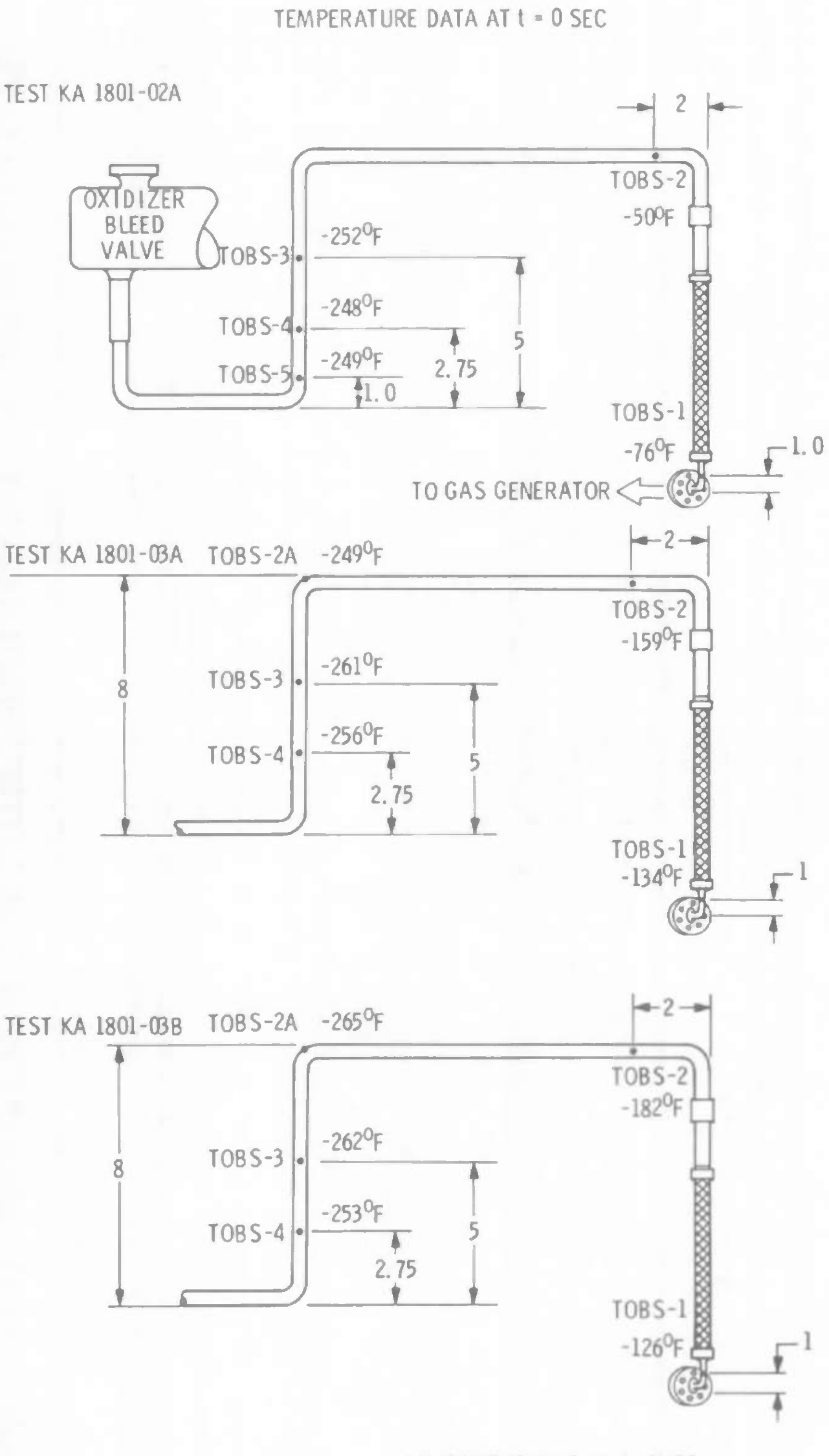


Fig. 16 Gas Generator Oxidizer Supply Line Temperatures, Firings 02A, 03A, and 03B

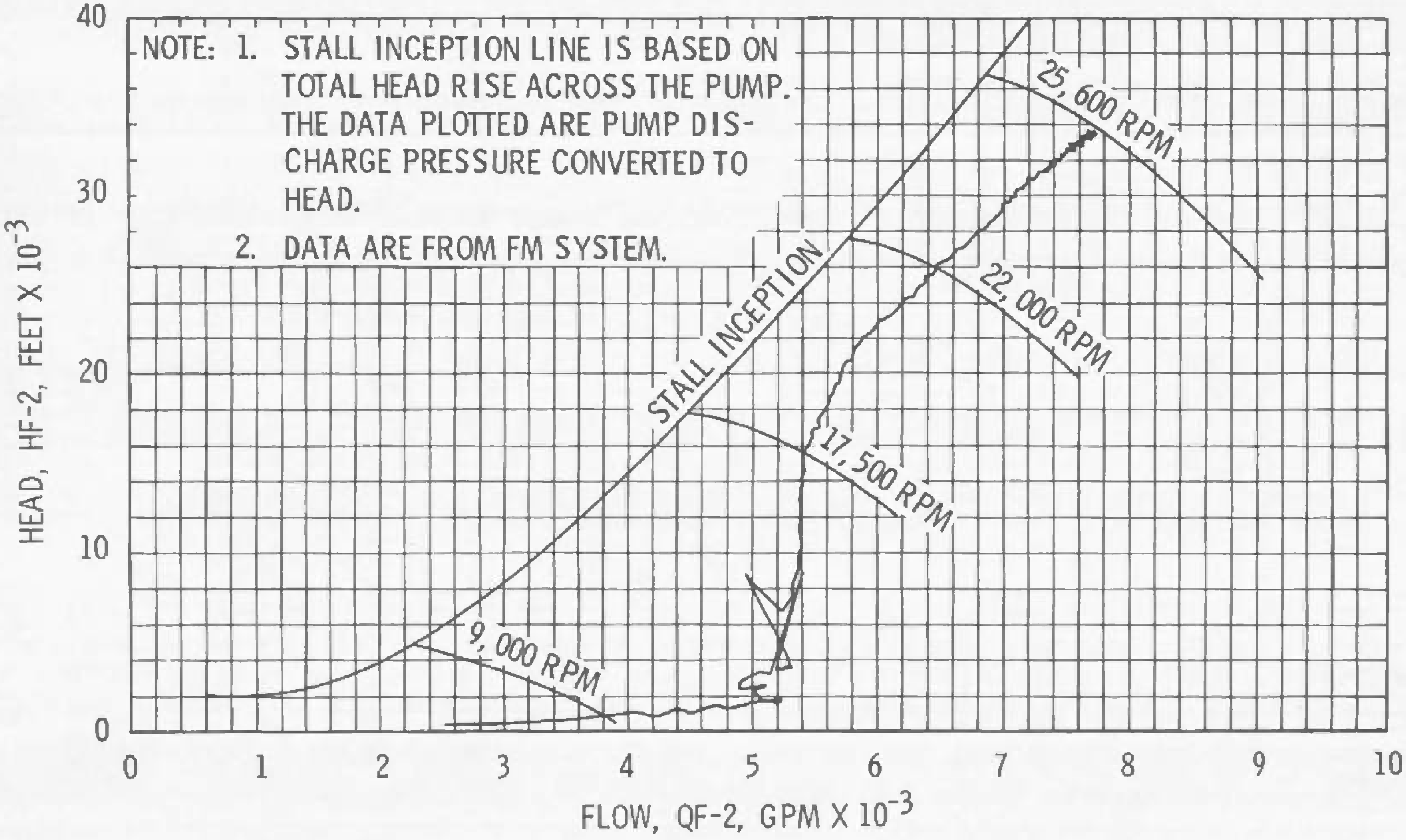
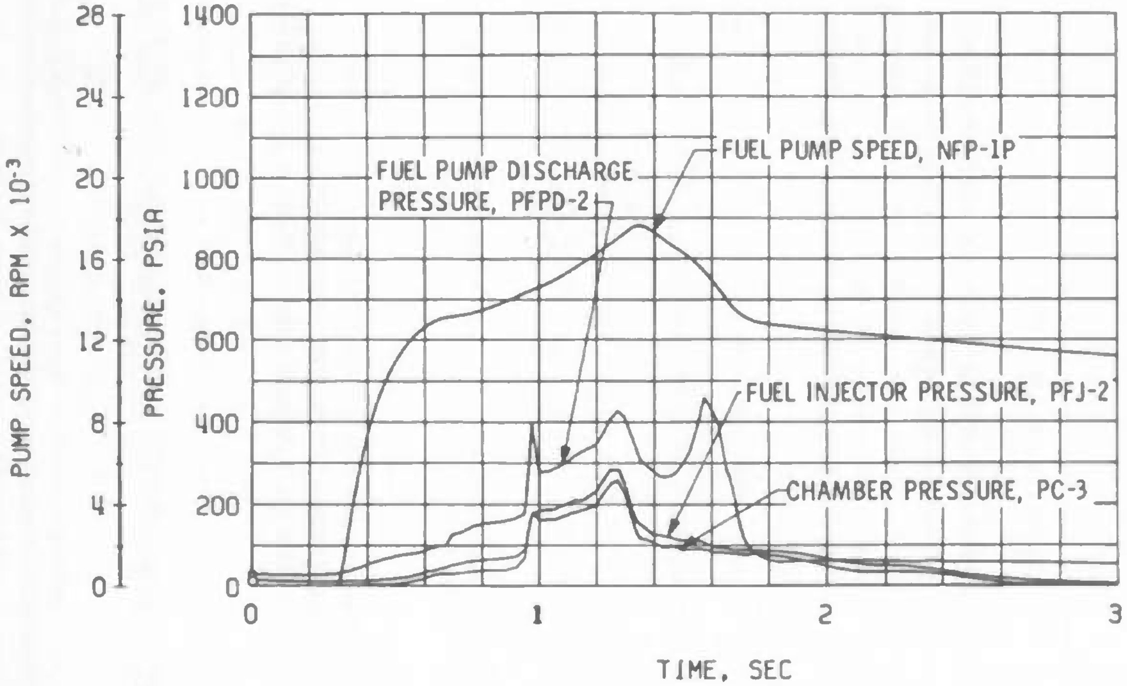
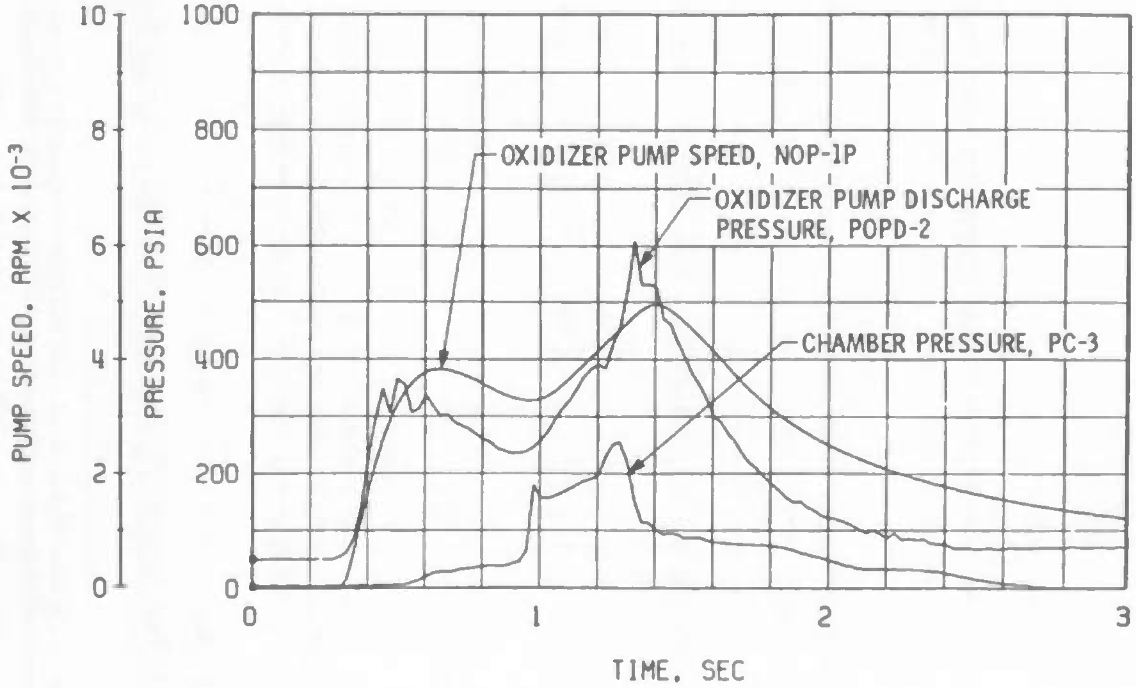


Fig. 17 Fuel Pump Start Transient Performance, Firing 03A



a. Fuel System



b. Oxidizer System

Fig. 18 Engine Start and Shutdown Transients, Firing 03B

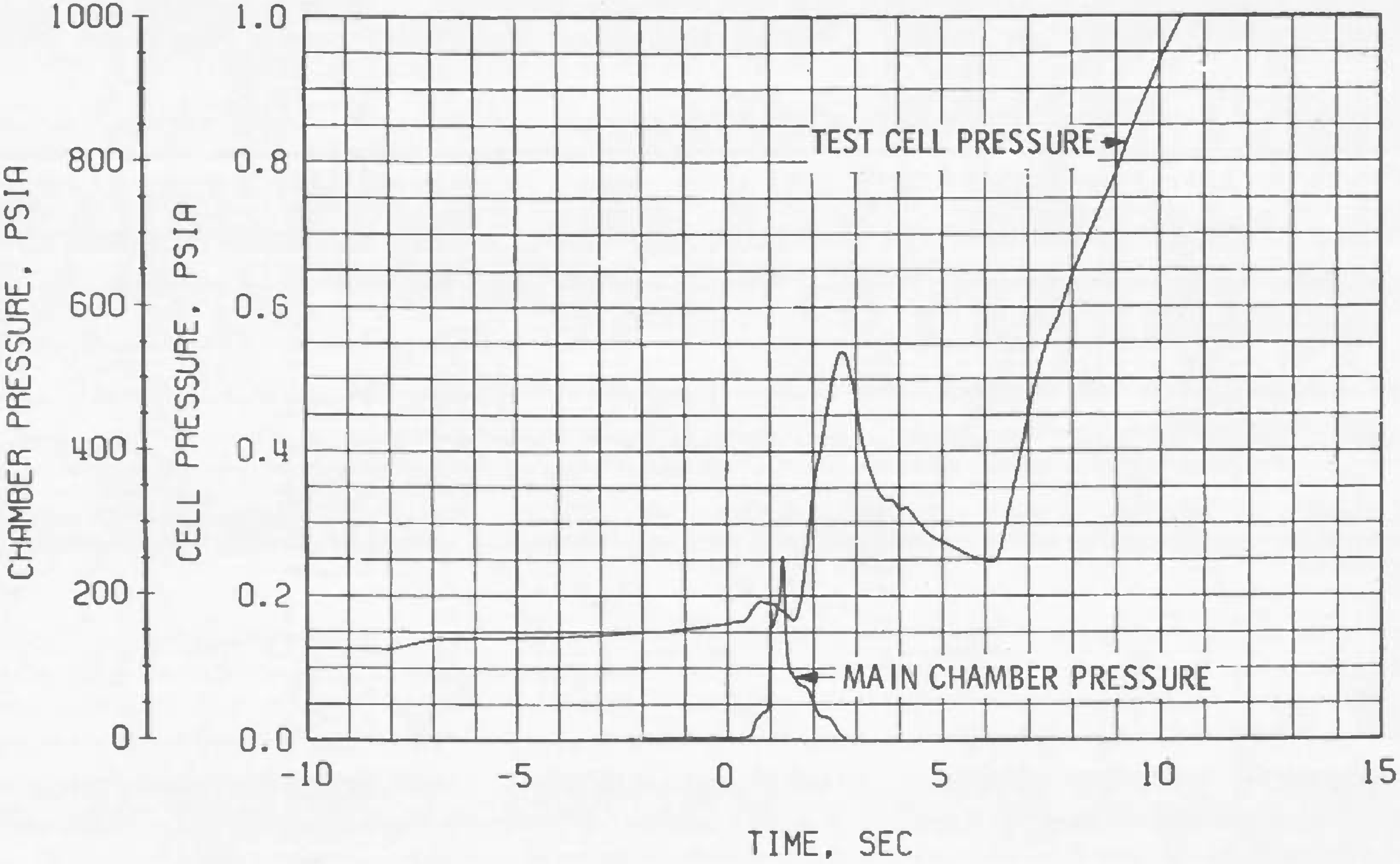
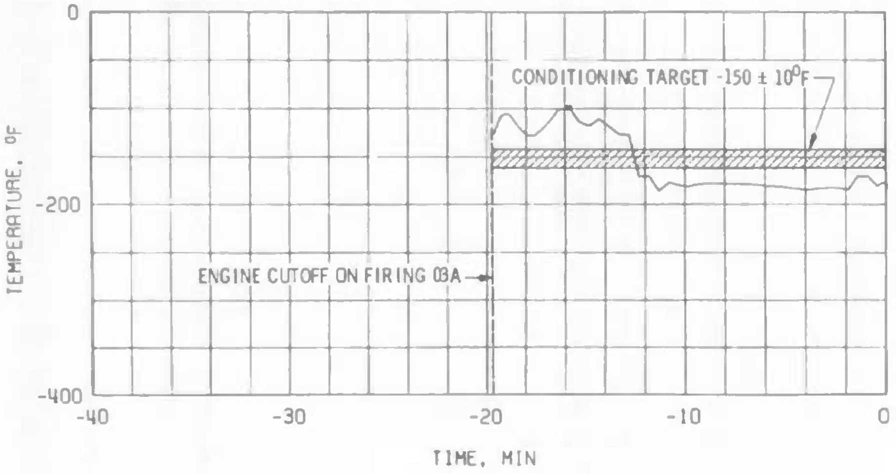
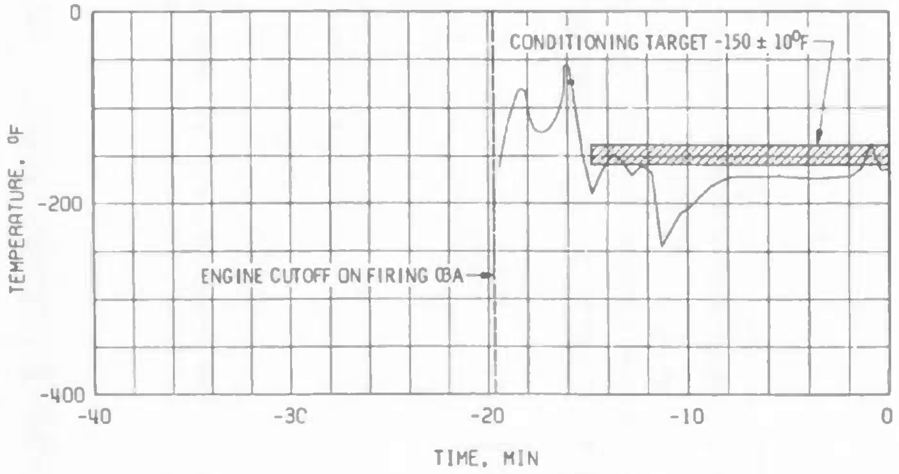


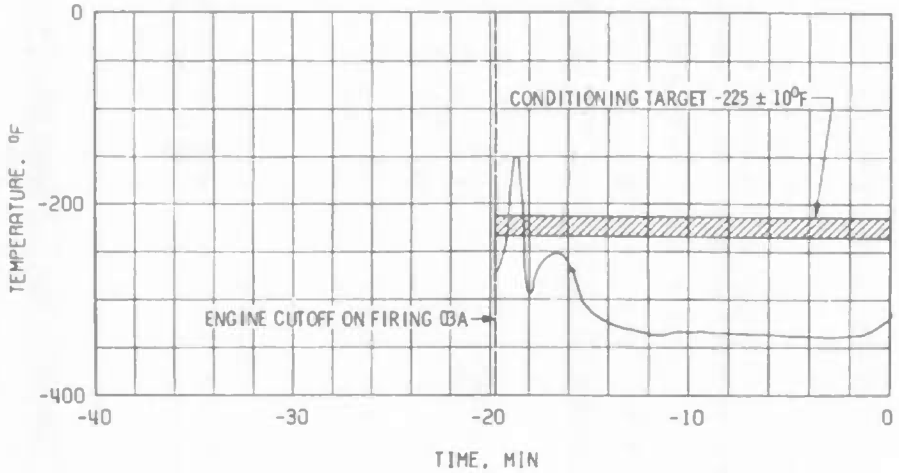
Fig. 19 Engine Ambient and Combustion Chamber Pressure, Firing 03B



a. Pneumatic Regulator, TBHR-2

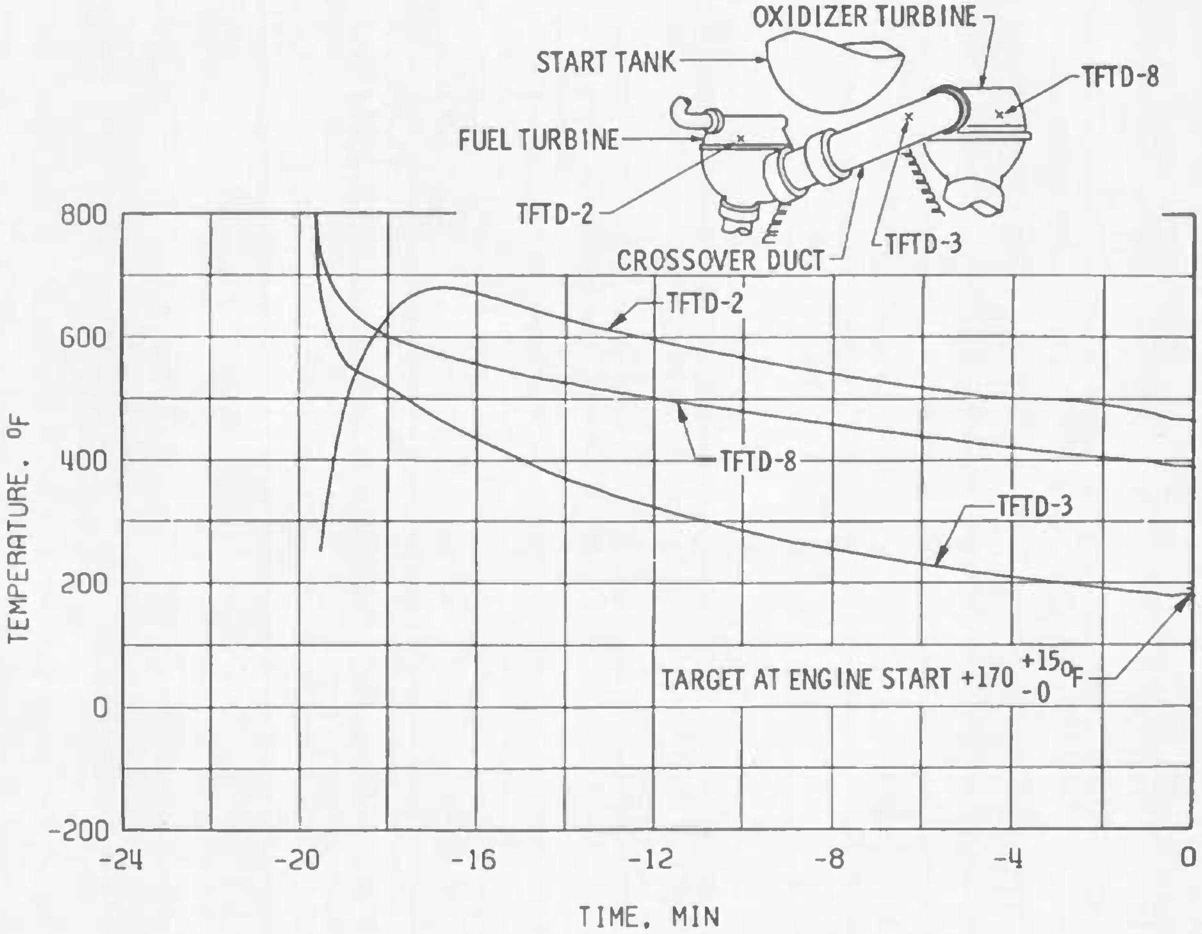


b. Main Oxidizer Closing Control Line, TSOVAL-2



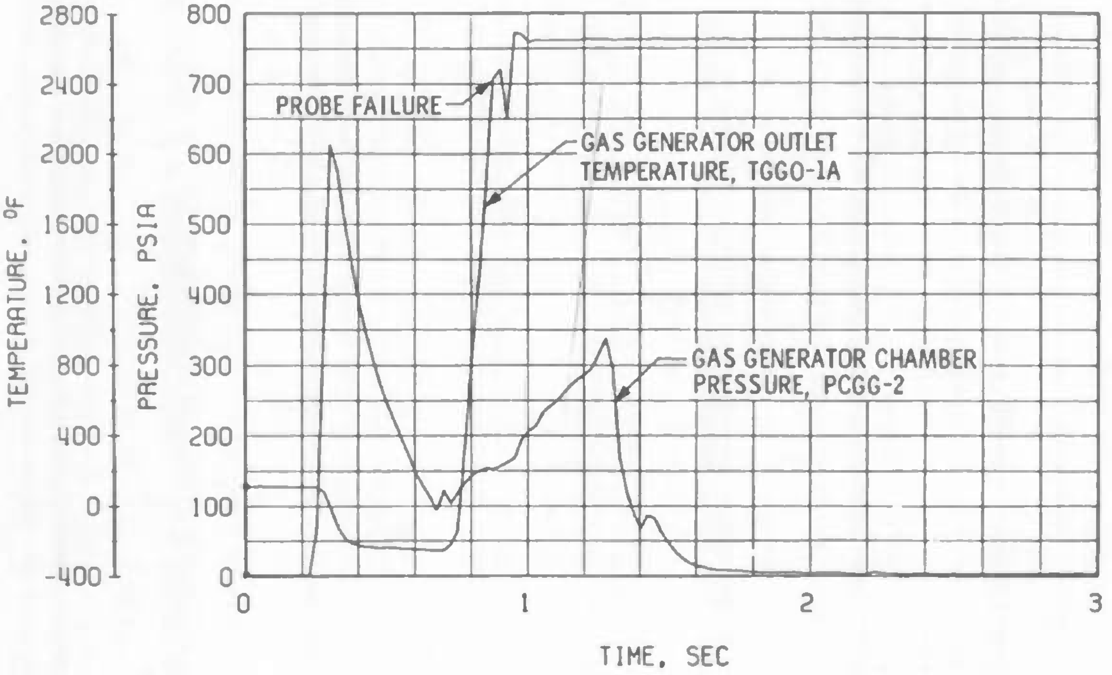
c. Main Oxidizer-Second-Stage Actuator, TSOVC-2

Fig. 20 Thermal Conditioning History of Engine Components, Pre-Fire 03B

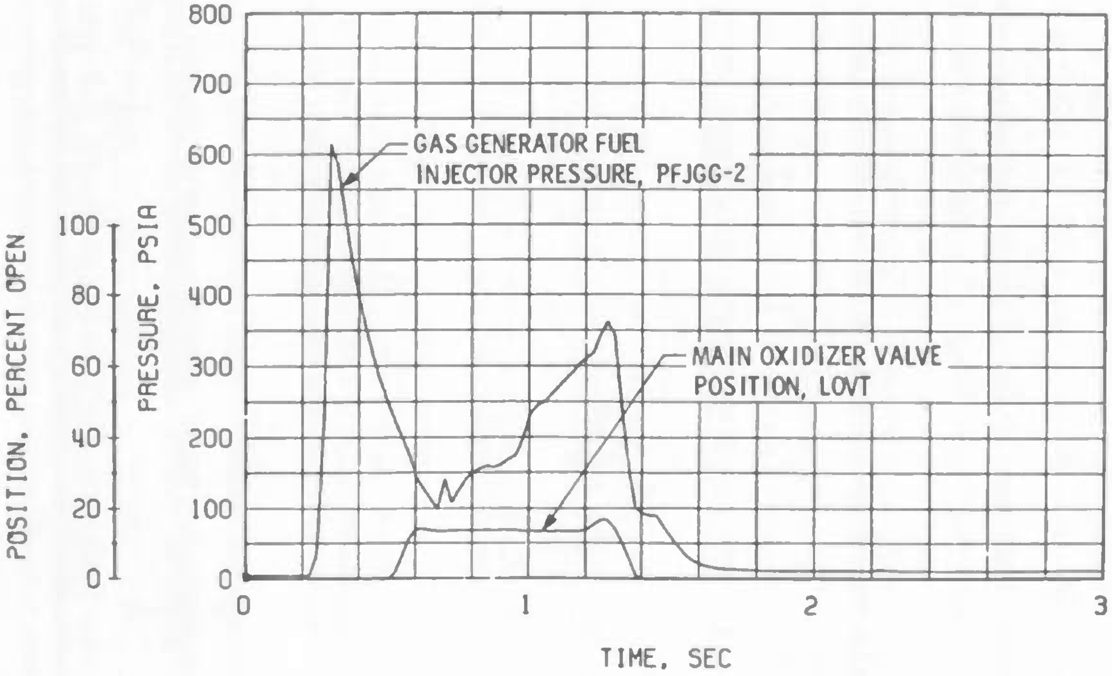


d. Crossover Duct, TFTD

Fig. 20 Concluded



a. Gas Generator Outlet Temperature and Chamber Pressure



b. Gas Generator Fuel Injector Pressure and Main Oxidizer Valve Position

Fig. 21 Gas Generator Ignition and Shutdown Transients, Firing 03B

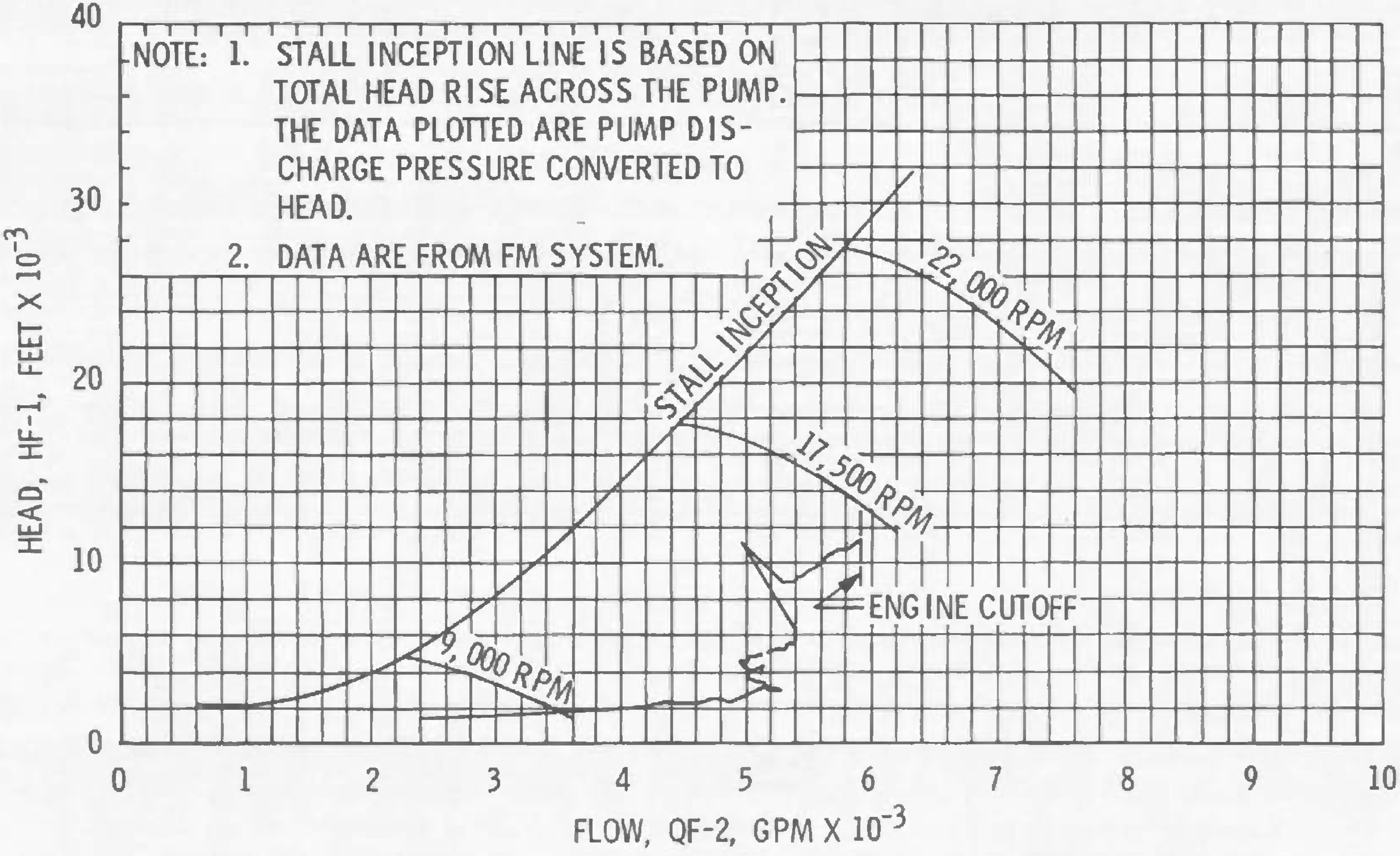


Fig. 22 Fuel Pump Start Transient Performance, Firing 03B

TABLE I
MAJOR ENGINE COMPONENTS

Part Name	P/N	S/N
Thrust Chamber Body	206600-31	4076553
Thrust Chamber Injector Assembly	208021-11	4084917
Fuel Turbopump Assembly	459000-161	4062324
Oxidizer Turbopump Assembly	458175-71	6623549
Start Tank	303439	0064
Augmented Spark Igniter	206280-21	3661349
Gas Generator Fuel Injector and Combustor	308360-11	2008734
Pneumatic Control Assembly	556947	4079720
Electrical Control Package	502670-11	4081748
Primary Flight Instrumentation Package	703685	4078716
Auxiliary Flight Instrumentation Package	703680	4078718
Main Fuel Valve	409120	4056924
Main Oxidizer Valve	411031	4089563
Gas Generator Control Valve	309040	4078714
Start Tank Discharge Valve	306875	4079062
Oxidizer Turbine Bypass Valve	409940	4048489
Propellant Utilization Valve	251351-11	4068944
Main-Stage Control Valve	558069	8313568
Ignition Phase Control Valve	558069	8275775
Helium Control Valve	106012000	342270
Start Tank Vent and Relief Valve	557828-X2	4046446
Helium Tank Vent Valve	106012000	342277
Fuel Bleed Valve	309034	4077749
Oxidizer Bleed Valve	309029	4077746
Augmented Spark Igniter Oxidizer Valve	308880	4077205
P/A Purge Control Valve	557823	4073021
Start Tank Fill/Refill Valve	558000	4079001
Fuel Flowmeter	251225	4077752
Oxidizer Flowmeter	251216	4074114
Fuel Injector Temperature Transducer	NA5-27441	12401
Restartable Ignition Detect Probe	XEOR915389	211

TABLE II
SUMMARY OF ENGINE ORIFICE

Orifice Name	Part Number	1801-02	1801-03	Comments
		Diameter		
Gas Generator Oxidizer Supply Line	RD251-4106	0.276 in.	0.284 in.	
Augmented Spark Igniter Oxidizer	406361	0.110 in.	0.110 in.	Thermostatic Orifice
Oxidizer Turbine Bypass Nozzle	RD273-8002	1.300 in.	1.430 in.	
Gas Generator Fuel Supply Line	RD273-4107	0.472 in.	0.489 in.	
Main Oxidizer Valve Closing Control	410437	8.34 scfm.	8.34 scfm.	
Oxidizer Turbine Exhaust	RD251-9004	9.99 in.	9.99 in.	Installed on the Engine before Shipment to AEDC

TABLE III
ENGINE MODIFICATIONS AT AEDC BETWEEN TESTS J4-1801-02 AND J4-1801-03

Modification	Completion* Date	Description of Modification
RFD† -44-1-67		Moved TOBS-5 to TOBS-2A
RFD-56-67		Installed One Heater and One Thermocouple on Both the Electrical Control and the Primary Instrument Packages
RFD-55-67	7-24-67	Installed Gas Generator Fuel Supply Orifice Installed Gas Generator Oxidizer Supply Orifice Installed Oxidizer Turbine Bypass Nozzle
RFD-54-67		Installed Turbine Component Isolation Insulation
RFD-57-67		Modified Oxidizer Pump Discharge Pressure Sensing Line (Static Stage)

†RFD - Rocketdyne Field Directive

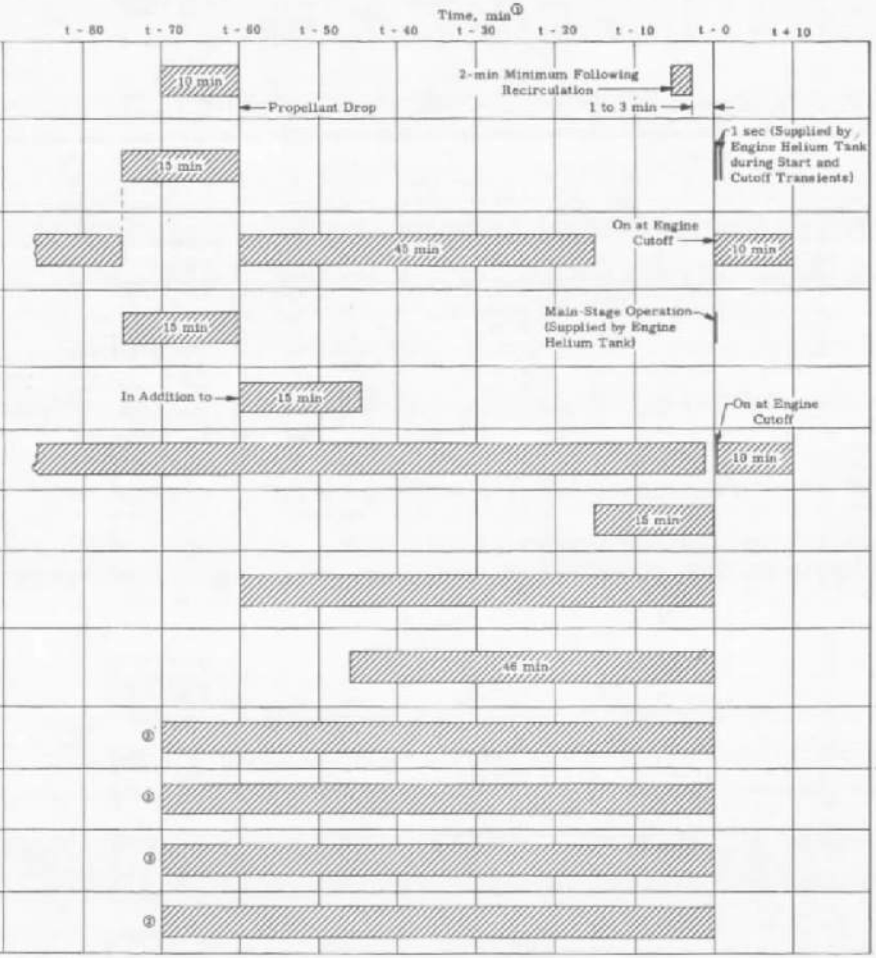
*All Dates Not Available from Rocketdyne

TABLE IV
ENGINE COMPONENT REPLACEMENTS (BETWEEN TESTS J4-1801-02 AND J4-1801-03)

Replacement	Completion Date	Component Replaced
UCR† No. 007971	7-24-67	Gas Generator Chamber and Oxidizer Pump Seal Cavity Pressure Transducer
Replaced for Reliability	7-17-67	Gas Generator Outlet Temperature Probe

† UCR - Unsatisfactory Condition Report

TABLE V
ENGINE PURGE AND COMPONENT CONDITIONING SEQUENCE



		Time, min①
Turbopump and Gas Generator Purge (Purge Manifold System)	Helium; 82 - 125 psia 50 - 200°F (Nominal) 8 scfm at Customer Connect	10 min Propellant Drop 2-min Minimum Following Recirculation 1 to 3 min
Oxidizer Dome and Gas Generator Liquid Oxygen Injector (Engine Pneumatic System)	Nitrogen; 400 ± 25 psig 50 - 200°F (Minimum) 230 scfm	15 min 1 sec (Supplied by Engine Helium Tank during Start and Cutoff Transients)
Oxidizer Dome (Facility Line to Port CO3A)	Nitrogen; 400 - 450 psig 100 - 200°F (Nominal) 200 scfm	45 min On at Engine Cutoff 10 min
Oxidizer Turbopump Intermediate Seal Cavity (Engine Pneumatic System)	Helium; 400 ± 25 psig Ambient Temperature 3800 - 7000 scfm	15 min Main-Stage Operation (Supplied by Engine Helium Tank)
Thrust Chamber Jacket (Customer Connect) Panel	Helium; 40 - 60 psig 50 - 200°F (Nominal) 60 scfm	In Addition to 15 min
	Helium; 12 - 14 psig 50 - 200°F (Nominal) 10 scfm	On at Engine Cutoff 10 min
Thrust Chamber Temperature Conditioning	Helium; 1000 psig -300°F to Ambient 10 - 20 lb_m/min	15 min
Pump Inlet Pressure and Temperature Conditioning	Oxidizer; 35 to 48 psia -298 to -280°F Fuel; 28 to 46 psia -424 to -410°F	
Hydrogen Start Tank and Helium Tank Pressure and Temperature Conditioning	Hydrogen; 1200 to 1400 psia -300 to -140°F Helium; 1700 to 3250 psia -300 to -140°F	46 min
Crossover Duct Temperature Conditioning	Helium; -300°F to Ambient	②
Main Oxidizer Valve Actuator Temperature Conditioning	Helium; -300°F to Ambient	②
Main Oxidizer Valve Closing Control Line (Conditioning)	Helium; -300°F to Ambient	③
Pneumatic Regulator Temperature Conditioning	Helium; -300°F to Ambient	②

① Times are adjusted for one and two orbit restart simulation firings.

② Component conditions to be maintained within limits for last 30 min before engine start or coast duration, whichever is longer.

③ Component conditions to be maintained within limits for last 15 min before engine start.

TABLE VI
SUMMARY OF TEST REQUIREMENTS AND RESULTS

Firing Number, J4-1801-03		03A		03B	
		Target	Actual	Target	Actual
Time of Day, hr/Firing Date		1240	July 26, 1967	1259	July 26, 1967
Pressure Altitude at Engine Start, ft (Ref. 4)		100,000	101,000	100,000	105,000
Firing Duration, sec①		30	30.072	5	1.245
Fuel Pump Inlet Conditions at Engine Start	Pressure, psia	28.0 ± 1.0	29.6	28.0 ± 1.0	28.3
	Temperature, °F	-420.4 ± 0.4	-420.6	-421.5 ± 0.4	-420.6
Oxidizer Pump Inlet Conditions at Engine Start	Pressure, psia	48.0 ± 1.0	48.6	48.0 ± 1.0	47.8
	Temperature, °F	-295.3 ± 0.5	-294.6	-295.3 ± 0.4	-294.3
Start Tank Conditions at Engine Start	Pressure, psia	1400 ± 10	1394	1400 ± 10	1401
	Temperature, °F	-200 ± 10	-202.7	-240 ± 0	-242.8
Helium Tank Conditions at Engine Start	Pressure, psia	---	2143	---	2119
	Temperature, °F	---	-207.7	---	-242.5
Thrust Chamber Temperature Conditions at Engine Start, °F	Throat	-200 ± 15	-186	---	110
	Average	---	-187	---	23
Crossover Duct Temperature at Engine Start, °F②	TFTD-2	-20 ± 15	-73	---	464
	TFTD-3		-38	170^{+15}_{-0}	181
	TFTD-8		-34	---	391
Main Oxidizer Valve Closing Control Line Temperature at Engine Start, °F③		-100 ± 10	-132	-150 ± 10	-163
Main Oxidizer Valve Second-Stage Actuator Temperature at Engine Start, °F③		-175 ± 10	-299	-225 ± 10	-321
Pneumatic Control Package Temperature at Engine Start, °F②		-100 ± 10	-95	-150 ± 10	-178
Fuel Lead Time, sec①		3	3.002	8	7.992
Propellant in Engine Time, min		44	143	---	10
Propellant Recirculation Time, min		t_0 -11	-11	t_0 -11	---
Prevalve Sequencing Logic		Auxiliary	Auxiliary	Normal	Normal
Bootstrap Line Temperature at t = 0, °F	TOBS-1	---	-134	---	-126
	TOBS-2	---	-159	---	-182
	TOBS-2A	---	-249	---	-265
Start Tank Discharge Valve Body Temperature at Engine Start, °F		---	-95	---	-123
Vibration Safety Counts Duration (msec) and Occurrence Time (sec) from t_0		--- ---	25 0.997	--- ---	9 0.968
Gas Generator Outlet Temperature, °F	Initial Peak	---	2491	---	2651
	Overshoot	---	---	---	---
Main Chamber Ignition (P_c = 100 psia) Time, sec (Ref. t_0)①		---	1.002	---	0.962
Main Oxidizer Valve Second-Stage Initial Movement, sec (Ref. t_0)①		---	1.002	---	1.185
Main-Stage Pressure No. 2, sec (Ref. t_0)①		---	1.638	---	---
550-psia Chamber Pressure Attained, sec (Ref. t_0)		---	1.925	---	---
Propellant Utilization Valve Position at Engine Start, deg	Engine Start/t_0 + 10 sec	Null Closed	Null Closed	Open No Excursion	Open ---

①Data reduced from oscillograph.

②Component conditioning to be maintained within limits for last 15 min before engine start.

③Component conditioning to be maintained within limits for last 30 min before engine start or coast duration, whichever is longer.

TABLE VII

ENGINE VALVE TIMINGS

Firing Number J4-1801-	Start																					
	Start Tank Discharge Valve						Main Fuel Valve			Main Oxidizer Valve First Stage			Main Oxidizer Valve Second Stage			Gas Generator Fuel Poppet			Gas Generator Oxidizer Poppet			Oxidizer Turbine Bypass Valve		
	Time of Opening Signal	Valve Delay Time, sec	Valve Opening Time, sec	Time of Closing Signal	Valve Delay Time, sec	Valve Closing Time, sec	Time of Opening Signal	Valve Delay Time, sec	Valve Opening Time, sec	Time of Opening Signal	Valve Delay Time, sec	Valve Opening Time, sec	Time of Opening Signal	Valve Delay Time, sec	Valve Opening Time, sec	Time of Opening Signal	Valve Delay Time, sec	Valve Opening Time, sec	Time of Opening Signal	Valve Delay Time, sec	Valve Opening Time, sec	Time of Closing Signal	Valve Delay Time, sec	Valve Closing Time, sec
03A	0	0.156	0.150	0.441	0.100	0.265	-3.001	0.059	0.060	0.441	0.055	0.060	0.441	0.560	1.786	0.441	0.102	0.062	0.441	0.212	0.070	0.441	0.275	0.312
03B	0	0.166	0.154	0.441	0.098	0.277	-7.991	0.058	0.048	0.441	0.067	0.068	0.441	0.754	---	0.441	0.113	0.065	0.441	0.233	0.095	0.441	0.255	0.307
Pre-Fire Final Sequence	0	0.098	0.107	0.447	0.095	0.245	-1.010	0.047	0.063	0.447	0.049	0.042	0.447	0.580	1.730	0.447	0.077	0.041	0.447	0.139	0.053	0.447	0.215	0.283
Full Sequence	0	0.098	0.107	0.448	0.091	0.246	-0.987	0.041	0.065	0.448	0.046	0.042	0.446	0.586	1.733	0.448	0.072	0.045	0.448	0.135	0.053	0.448	0.204	0.287

Firing Number J4-1801-	Shutdown														
	Main Fuel Valve			Main Oxidizer Valve			Gas Generator Fuel Poppet			Gas Generator Oxidizer Poppet			Oxidizer Turbine Bypass Valve		
	Time of Closing Signal	Valve Delay Time, sec	Valve Closing Time, sec	Time of Closing Signal	Valve Delay Time, sec	Valve Closing Time, sec	Time of Closing Signal	Valve Delay Time, sec	Valve Closing Time, sec	Time of Closing Signal	Valve Delay Time, sec	Valve Closing Time, sec	Time, of Opening Signal	Valve Delay Time, sec	Valve Opening Time, sec
03A	30.071	0.120	0.330	30.071	0.100	0.245	30.071	0.078	0.112	30.071	0.030	0.018	30.071	0.313	0.586
03B	1.345	0.112	0.310	1.245	---	0.110	1.245	0.091	0.081	1.245	0.023	0.043	1.245	0.173	0.545
Pre-Fire Final Sequence	6.002	0.087	0.246	6.002	0.062	0.127	6.002	0.086	0.045	6.002	0.062	0.020	6.002	0.225	0.580
Full Sequence	29.628	0.087	0.246	29.626	0.063	0.130	29.626	0.083	0.040	29.626	0.056	0.023	20.626	0.223	0.575

Notes:
1. All valve signal times are referenced to t_0.
2. Valve delay time is the time required for initial valve movement after the valve open or closed solenoid has been energized.
3. Final sequence check is conducted without propellants and within 12 hr before testing.
4. Data reduced from oscillogram.

TABLE VIII
COMPARISON OF TEST CONDITIONS FOR FIRINGS 02A AND 03A

Firing Number: J4-1801-		02A	03A
Fuel Pump Inlet Conditions at Engine Start	Pressure, psia	28.3	29.6
	Temperature, °F	-420.4	-420.6
Oxidizer Pump Inlet Conditions at Engine Start	Pressure, psia	47.7	48.6
	Temperature, °F	-295.8	-294.6
Start Tank Conditions at Engine Start	Pressure, psia	1395	1394
	Temperature, °F	-203	-203
Thrust Chamber Temperature Conditions at Engine Start, °F	Throat	-115	-186
	Average	-91	-187
Crossover Duct Temperature Conditions at Engine Start, °F	TFTD-3	-17	-38
Main Oxidizer Valve Closing Control Line Temperature at Engine Start, °F		-96	-132
Main Oxidizer Valve Second-Stage Actuator Temperature at Engine Start, °F		-166	-299
Pneumatic Control Package Temperature at Engine Start, °F		-116	-95
Fuel Lead Time, sec		3.000	3.002
Bootstrap Line Temperature at t = 0, °F	TOBS-1	-76	-134
	TOBS-2	-50	-159
	TOBS-3	-252	-249

TABLE IX
ENGINE PERFORMANCE SUMMARY

Firing Number J4-1801-03A		Site*	Normalized†
Time, sec		29.5	29.5
Overall Engine Performance	Thrust, lb$_f$	229,100	226,900
	Chamber Pressure, psia	773	763
	Mixture Ratio	5.69	5.66
	Fuel Weight Flow, lb$_m$/sec	81.2	80.5
	Oxidizer Weight Flow, lb$_m$/sec	461.8	455.1
	Total Weight Flow, lb/sec	543.0	535.6
Thrust Chamber Performance	Mixture Ratio	5.911	5.878
	Total Weight Flow, lb/sec	535.8	528.4
	Characteristic Velocity, ft/sec	7904	7912
Fuel Turbopump Performance	Pump Efficiency	73.8	73.8
	Pump Speed, rpm	26,598	26,328
	Turbine Inlet Temperature, °F	1263	1235
	Turbine Weight Flow, lb/sec	7.18	7.14
Oxidizer Turbopump Performance	Pump Efficiency	80.4	80.4
	Pump Speed, rpm	8613	8546
	Turbine Inlet Temperature, °F	821	801
Gas Generator Performance	Mixture Ratio	0.978	0.961
	Chamber Pressure, psia	675.8	669.0

*Site - Test Data

†Normalized - Test data corrected to standard pump inlet and engine ambient vacuum conditions.

APPENDIX III
INSTRUMENTATION AT AEDC FOR J-2 ROCKET ENGINE S/N J2052 ON TEST J4-1801-03

The instrumentation for AEDC Test J4-1801-03 is tabulated in Table III-1. The location of selected major engine instrumentation is shown in Fig. III-1.

TABLE III-1
LIST OF ENGINE INSTRUMENTATION

AEDC Code	Parameter	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillograph	Strip Chart	X-Y Plotter
	Current		amp					
ICC	Control		0 to 30	x		x		
IIC	Ignition		0 to 30	x		x		
	Event							
EECL	Engine Cutoff Lockin		On/Off	x		x		
EECO	Engine Cutoff Signal		On/Off	x	x	x		
EES	Engine Start Command		On/Off	x		x		
EFBVC	Fuel Bleed Valve Closed Limit		Open/Closed	x				
EFJT	Fuel Injector Temperature		On/Off	x		x		
EFPVC/O	Fuel Prevalve Closed/Open Limit		Closed/Open	x		x		
EHCS	Helium Control Solenoid		On/Off	x		x		
EID	Ignition Detected		On/Off	x		x		
EIPCS	Ignition Phase Control Solenoid		On/Off	x		x		
EMCS	Main-Stage Control Solenoid		On/Off	x		x		
EMP-1	Main-Stage Pressure No. 1		On/Off	x		x		
EMP-2	Main-Stage Pressure No. 2		On/Off	x		x		
EOBVC	Oxidizer Bleed Valve Closed Limit		Open/Closed	x				
EOPVC	Oxidizer Prevalve Closed Limit		Closed	x		x		
EOPVO	Oxidizer Prevalve Open Limit		Open	x		x		
ESTDCS	Start Tank Discharge Control Solenoid		On/Off	x	x	x		
	Sparks							
RASIS-1	Augmented Spark Igniter Spark No. 1		On/Off			x		
RASIS-2	Augmented Spark Igniter Spark No. 2					x		
RGGS-1	Gas Generator Spark No. 1		On/Off			x		
RGGS-2	Gas Generator Spark No. 2		On/Off			x		
	Flows		gpm					
QF-1A	Fuel	PFF	0 to 9000	x		x		
QF-2	Fuel	PFFA	0 to 9000	x	x	x		
QFRP	Fuel Recirculation		0 to 160	x				
QO-1A	Oxidizer	POF	0 to 3000	x		x		
QO-2	Oxidizer	POFA	0 to 3000	x	x	x		
QORP	Oxidizer Recirculation		0 to 50	x			x	
	Forces		lbf					
FSP-1	Side Load (Pitch)		±20,000	x		x		
FSY-1	Side Load (Yaw)		±20,000	x		x		
	Position		Percent Open					
LFVT	Main Fuel Valve		0 to 100	x		x		
LGGVT	Gas Generator Valve		0 to 100	x		x		
LOTBVT	Oxidizer Turbine Bypass Valve		0 to 100	x		x		
LOVT	Main Oxidizer Valve		0 to 100	x	x	x		
LPUTOP	Propellant Utilization Valve		0 to 100	x		x	x	
LSTDVT	Start Tank Discharge Valve		0 to 100	x		x		

TABLE III-1 (Continued)

AEDC Code	Parameter	Tap No	Range	Micro-SADIC	Magnetic Tape	Oscillograph	Strip Chart	X-Y Plotter
	Pressure		psia					
PA1	Test Cell		0 to 0.5	x		x		
PA2	Test Cell		0 to 1.0	x	x			
PA3	Test Cell		0 to 5.0	x			x	
PC-1P	Thrust Chamber	CG1	0 to 1000	x			x	
PC-3	Thrust Chamber	CG1A	0 to 1000	x	x	x		
PCASI-2	Augmented Spark Igniter Chamber	IG1	0 to 1000	x				
PCGG-1P	Gas Generator Chamber	GG1	1000	x	x	x		
PCGG-2	Gas Generator Chamber	GG1A	0 to 1000	x				
PFASIJ	Augmented Spark Igniter Fuel Injection		0 to 1000	x				
PFJ-1A	Main Fuel Injection	CF2	0 to 1000	x		x		
PFJ-2	Main Fuel Injection	CF2A	0 to 1000	x	x			
PFJGG-1A	Gas Generator Fuel Injection	GF4	0 to 1000	x				
PFJGG-2	Gas Generator Fuel Injection	GF4	0 to 1000	x		x		
PFMI	Fuel Jacket Inlet Manifold	CF1	0 to 2000	x				
PFOI-1A	Fuel Tapoff Orifice Outlet	HF2	0 to 1000	x				
PFPC-1A	Fuel Pump Balance Piston Cavity	PF5	0 to 1000	x				
PFPD-1P	Fuel Pump Discharge	PF3	0 to 1500	x				
PFPD-2	Fuel Pump Discharge	PF2	0 to 1500	x	x	x		
PFPI-1	Fuel Pump Inlet		0 to 100	x				x
PFPI-2	Fuel Pump Inlet		0 to 200	x				x
PFPI-3	Fuel Pump Inlet		0 to 200		x	x		
PFPS-1P	Fuel Pump Interstage	PF6	0 to 200	x				
PFRPO	Fuel Recirculation Pump Outlet		0 to 60	x				
PFRPR	Fuel Recirculation Pump Return		0 to 50	x				
PFST-1P	Fuel Start Tank	TF1	0 to 1500	x		x		
PFST-2	Fuel Start Tank	TF1	0 to 1500	x				x
PFUT	Fuel Tank Ullage		0 to 100	x				
PFVI	Fuel Tank Repressurization Line Nozzle Inlet		0 to 1000	x				
PFVL	Fuel Tank Repressurization Line Nozzle Throat		0 to 1000	x				
PGBXI	Bypass Nozzle Inlet	TG8	0 to 200	x				
PHECMO	Pneumatic Control Module Outlet		0 to 750	x				
PHEOP	Oxidizer Recirculation Pump Purge		0 to 150	x				
PHET-1P	Helium Tank	NN1	0 to 3500	x		x		
PHET-2	Helium Tank	NN1	0 to 3500	x				x
PHRO-1A	Helium Regulator Outlet	NN2	0 to 750	x	x			
POBSC	Oxidizer Bootstrap Conditioning		0 to 50	x				
POBV	Gas Generator Oxidizer Bleed Valve	GO2	0 to 2000	x				
POJ-1A	Main Oxidizer Injection	CO3	0 to 1000	x				
POJ-2	Main Oxidizer Injection	CO3A	0 to 1000	x		x		
POJGG-1A	Gas Generator Oxidizer Injection	GO5	0 to 1000	x		x		

TABLE III-I (Continued)

AEDC Code	Parameter	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillograph	Strip Chart	X-Y Plotter
	Pressure							
POJGG-2	Gas Generator Oxidizer Injection	GO5	0 to 1000	x				
POPBC-1A	Oxidizer Pump Bearing Coolant	PO7	0 to 500	x				
POPD-1P	Oxidizer Pump Discharge	PO3	0 to 1500	x				
POPD-2	Oxidizer Pump Discharge	PO2	0 to 1500	x	x	x		
POPI-1	Oxidizer Pump Inlet		0 to 100	x				x
POPI-2	Oxidizer Pump Inlet		0 to 200	x				x
POPI-3	Oxidizer Pump Inlet		0 to 100			x		
POPSC-1A	Oxidizer Pump Primary Seal Cavity	PO6	0 to 50	x				
PORPO	Oxidizer Recirculation Pump Outlet		0 to 115	x				
PORPR	Oxidizer Recirculation Pump Return		0 to 100	x				
POTI-1A	Oxidizer Turbine Inlet	TG3	0 to 200	x				
POTO-1A	Oxidizer Turbine Outlet	TG4	0 to 100	x				
POUT	Oxidizer Tank Ullage		0 to 100	x				
POVCC	Main Oxidizer Valve Closing Control		0 to 500	x	x			
POVI	Oxidizer Tank Repressurization Line Nozzle Inlet		0 to 1000	x				
POVL	Oxidizer Tank Repressurization Line Nozzle Throat		0 to 1000	x				
PPUVI-1A	Propellant Utilization Valve Inlet	PO8	0 to 1000	x				
PPUVO-1A	Propellant Utilization Valve Outlet	PO9	0 to 500	x				
PTCFJP	Thrust Chamber Fuel Jacket Purge		0 to 100	x				
PTPP	Turbopump and Gas Generator Purge		0 to 250	x				
	Speeds		rpm					
NFP-1P	Fuel Pump	PFV	0 to 30,000	x	x	x		
NFRP	Fuel Recirculation Pump		0 to 15,000	x				
NOP-1P	Oxidizer Pump	POV	0 to 12,000	x	x	x		
NORP	Oxidizer Recirculation Pump		0 to 15,000	x				
	Temperatures		°F					
TA1	Test Cell (North)		-50 to +800	x				
TA2	Test Cell (East)		-50 to +800	x				
TA3	Test Cell (South)		-50 to +800	x				
TA4	Test Cell (West)		-50 to +800	x				
TAIP-1A	Auxiliary Instrument Package		-300 to +200	x				
TBHR-1	Helium Regulator Body (North Side)		-100 to +50	x				
TBHR-2	Helium Regulator Body (South Side)		-100 to +50	x			x	
TBSC	Oxidizer Bootstrap Conditioning		-350 to +150	x				
TCLC	Main Oxidizer Valve Closing Control Line Conditioning		-325 to +200	x				

TABLE III-1 (Continued)

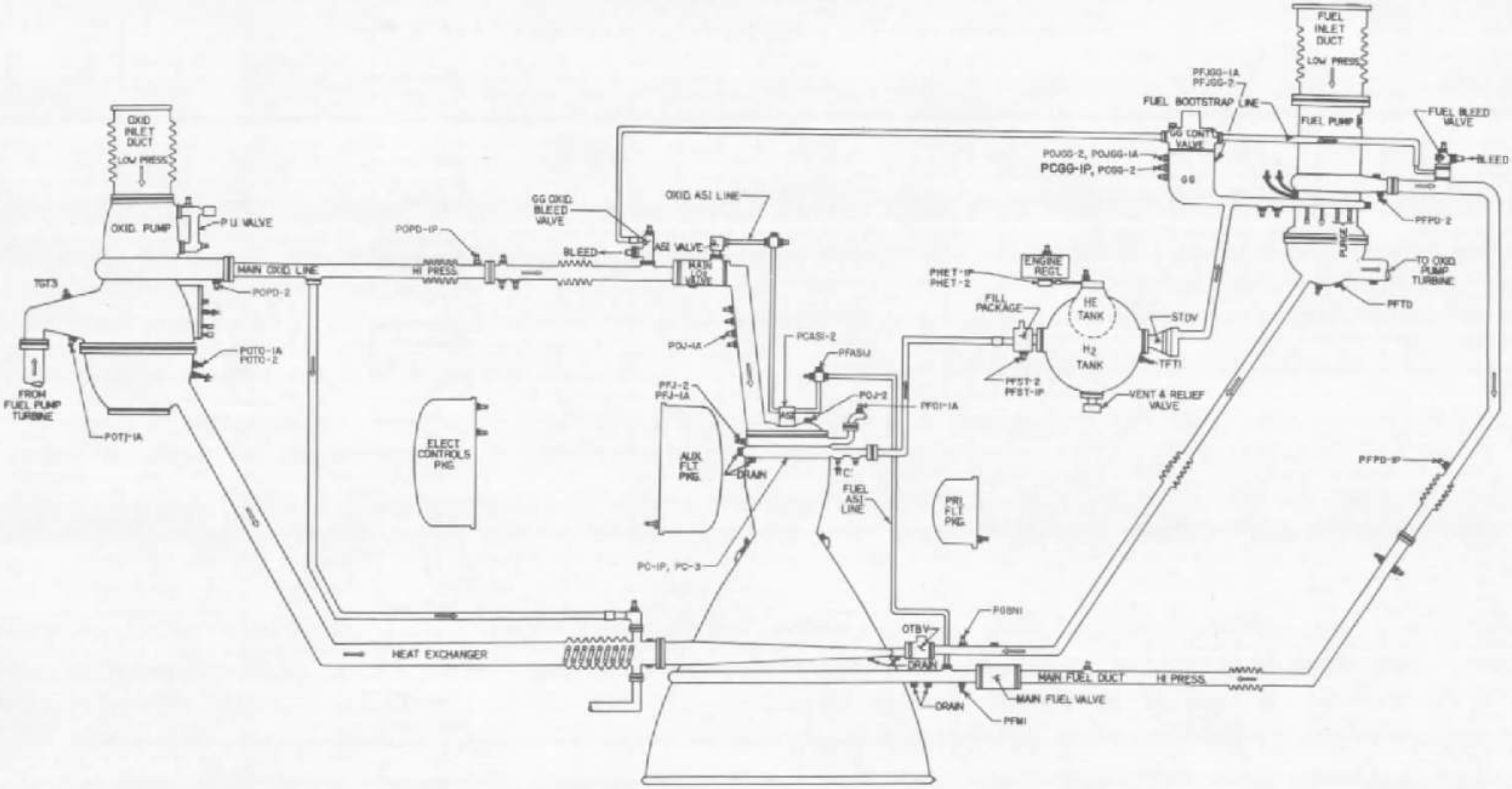
AEDC Code	Parameter	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillograph	Strip Chart	X-Y Plotter
	Temperatures		°F					
TECP-1P	Electrical Controls Package	NST1A	-300 to +200	x			x	
TFASIJ	Augmented Spark Igniter Fuel Injection	IFT1	-425 to -100	x		x		
TFASIL-1	Augmented Spark Igniter Line Temperature		-300 to +200	x	x			
TFASIL-2	Augmented Spark Igniter Line Temperature		-300 to +300	x	x			
TFBV-1A	Fuel Bleed Valve	GFT1	-425 to -475	x				
TFJ-1P	Main Fuel Injection	CFT2	-425 to +250	x	x	x		
TFPD-1P	Fuel Pump Discharge	PFT1	-425 to -400	x	x	x		
TFPD-2	Fuel Pump Discharge	PFT1	-425 to -400	x				
TFPDD	Fuel Pump Discharge Duct		-320 to +300	x				
TFPI-1	Fuel Pump Inlet		-425 to -400	x				x
TFPI-2	Fuel Pump Inlet		-425 to -400	x				x
TFRPO	Fuel Recirculation Pump Outlet		-425 to -410	x				
TFRPR	Fuel Recirculation Pump Return Line		-425 to -250	x				
TFRT-1	Fuel Tank		-425 to -410	x				
TFRT-2	Fuel Tank		-425 to -410	x				
TFST-1P	Fuel Start Tank	TFT1	-350 to +100	x				
TFST-2	Fuel Start Tank	TFT1	-350 to +100	x				x
TFTD-1	Fuel Turbine Discharge Duct		-200 to +800	x				
TFTD-1R	Fuel Turbine Discharge Collector		-200 to +900	x				
TFTD-2	Fuel Turbine Discharge Duct		-200 to +1000	x			x	
TFTD-3	Fuel Turbine Discharge Duct		-200 to +1000	x			x	
TFTD-3R	Fuel Turbine Discharge Line		-200 to +900	x				
TFTD-4	Fuel Turbine Discharge Duct		-200 to +1000	x				
TFTD-4R	Fuel Turbine Discharge Line		-200 to +900	x				
TFTD-5	Fuel Turbine Discharge Duct		-200 to +1400	x				
TFTD-6	Fuel Turbine Discharge Duct		-200 to +1400	x				
TFTD-7	Fuel Turbine Discharge Duct		-200 to +1400	x				
TFTD-8	Fuel Turbine Discharge Duct		-200 to +1400	x			x	
TFTI-1P	Fuel Turbine Inlet	TFT1	0 to 1800	x			x	
TFTO	Fuel Turbine Outlet	TFT2	0 to 1800	x				
TGGO-1A	Gas Generator Outlet	GGT1	0 to 1800	x	x	x		
THET-1P	Helium Tank	NST1	-350 to -100	x				x
TMOVC	Main Oxidizer Valve Actuator Conditioning		-325 to +200	x				
TOBS-1	Oxidizer Bootstrap Line		-300 to +250	x				
TOBS-2	Oxidizer Bootstrap Line		-300 to +250	x				
TOBS-2A	Oxidizer Bootstrap Line		-300 to +250	x				
TOBS-3	Oxidizer Bootstrap Line		-300 to +250	x				
TOBS-4	Oxidizer Bootstrap Line		-300 to +250	x				
TOBSCI	Oxidizer Bootstrap Conditioning Inlet		0 to 100	x				
TOBSCO	Oxidizer Bootstrap Conditioning Outlet		0 to 100	x				
TOBV-1A	Oxidizer Bleed Valve	GOT2	-300 to -250	x				

TABLE III-1 (Continued)

AEDC Code	Parameter	Tap No	Range	Micro-SADIC	Magnetic Tape	Oscillograph	Strip Chart	X-Y Plotter
	Temperatures		°F					
TOPB-1A	Oxidizer Pump Bearing Coolant	POT4	-300 to -250	x				
TOPD-1P	Oxidizer Pump Discharge	POT3	-300 to -250	x	x	x	x	
TOPD-2	Oxidizer Pump Discharge	POT3	-300 to -250	x				
TOPI-1	Oxidizer Pump Inlet		-310 to -270	x				x
TOPI-2	Oxidizer Pump Inlet		-310 to -270	x				x
TORPO	Oxidizer Recirculation Pump Outlet		-300 to -250	x				
TORPR	Oxidizer Recirculation Pump Return		-300 to -140	x				
TORT-1	Oxidizer Tank		-300 to -287	x				
TORT-3	Oxidizer Tank		-300 to -287	x				
TOTI-1P	Oxidizer Turbine Inlet	TGT3	0 to 1200	x			x	
TOTO-1P	Oxidizer Turbine Outlet	TGT4	0 to 1000	x				
TOVL	Oxidizer Tank Repressurization Line Nozzle Throat		-300 to +100	x				
TPIP-1P	Primary Instrument Package		-300 to +200	x				
TPPC	Pneumatic Package Conditioning		-325 to +200	x				
TSC2-1	Thrust Chamber Skin		-300 to +500	x				
TSC2-2	Thrust Chamber Skin		-300 to +500	x				
TSC2-3	Thrust Chamber Skin		-300 to +500	x				
TSC2-4	Thrust Chamber Skin		-300 to +500	x				
TSC2-5	Thrust Chamber Skin		-300 to +500	x				
TSC2-6	Thrust Chamber Skin		-300 to +500	x				
TSC2-7	Thrust Chamber Skin		-300 to +500	x				
TSC2-8	Thrust Chamber Skin		-300 to +500	x				
TSC2-9	Thrust Chamber Skin		-300 to +500	x				
TSC2-10	Thrust Chamber Skin		-300 to +500	x				
TSC2-11	Thrust Chamber Skin		-300 to +500	x				
TSC2-12	Thrust Chamber Skin		-300 to +500	x				
TSC2-13	Thrust Chamber Skin		-300 to +500	x			x	
TSC2-14	Thrust Chamber Skin		-300 to +500	x				
TSC2-15	Thrust Chamber Skin		-300 to +500	x				
TSC2-16	Thrust Chamber Skin		-300 to +500	x				
TSC2-17	Thrust Chamber Skin		-300 to +500	x				
TSC2-18	Thrust Chamber Skin		-300 to +500	x				
TSC2-19	Thrust Chamber Skin		-300 to +500	x				
TSC2-20	Thrust Chamber Skin		-300 to +500	x				
TSC2-21	Thrust Chamber Skin		-300 to +500	x				
TSC2-22	Thrust Chamber Skin		-300 to +500	x				
TSC2-23	Thrust Chamber Skin		-300 to +500	x				
TSC2-24	Thrust Chamber Skin		-300 to +500	x				
TSOVAL-1	Oxidizer Valve Closing Control Line		-200 to +100	x				

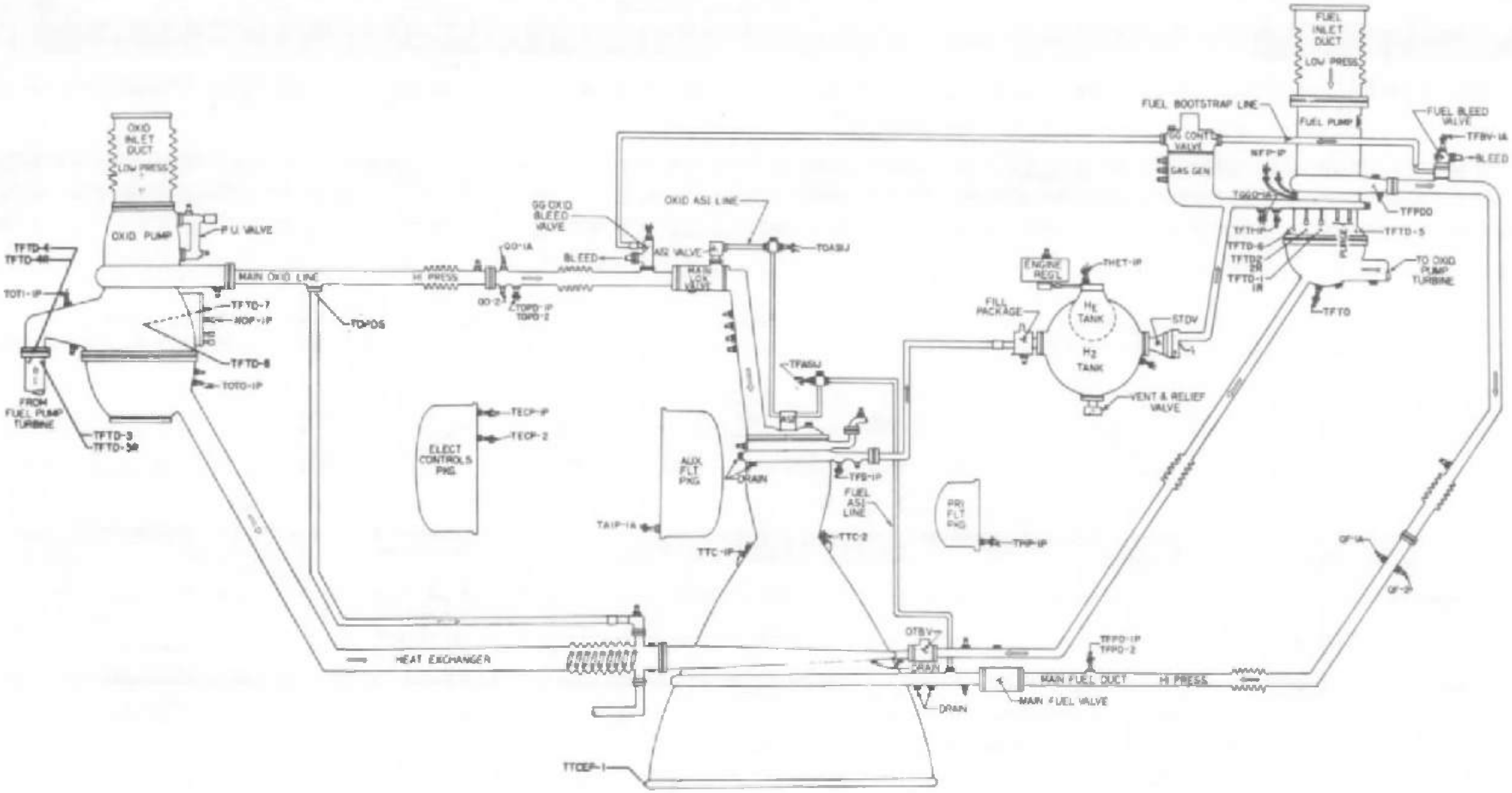
TABLE III-1 (Concluded)

AEDC Code	Parameter	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillograph	Strip Chart	X-Y Plotter
	Temperatures		°F					
TSOVAL-2	Oxidizer Valve Closing Control Line		-200 to +100	x			x	
TSOVC-1	Oxidizer Valve Actuator Cap		-325 to +150	x				
TSOVC-2	Oxidizer Valve Actuator Filter Flange		-325 to +150	x				
TSTC	Start Tank Conditioning		-350 to +150	x				
TSTDVCC	Start Tank Discharge Valve Closing Control Port		-350 to +100	x				
TSTDVOC	Start Tank Discharge Valve Opening Control Port		-350 to +100	x				
TTC-1P	Thrust Chamber Jacket (Control)	CS1	-425 to +500	x			x	
TTCEP-1	Thrust Chamber Exit		-425 to +500	x				
TXOC	Crossover Duct Conditioning		-325 to +200	x				
	Vibrations		g					
UFPR	Fuel Pump Radial 90 deg		±200		x			
UOPR	Oxidizer Pump Radial 90 deg		±200		x			
UTCD-1	Thrust Chamber Dome		±500		x	x		
UTCD-2	Thrust Chamber Dome		±500		x	x		
UTCD-3	Thrust Chamber Dome		±500		x	x		
U1VSC	No. 1 Vibration Safety Counts		On/Off			x		
U2VSC	No. 2 Vibration Safety Counts		On/Off			x		
	Voltage		Volts					
VCB	Control Bus		0 to 36	x		x		
VIB	Ignition Bus		0 to 36	x		x		
VIDA	Ignition Detect Amplifier		9 to 16	x		x		
VPUTEP	Propellant Utilization Valve Excitation		0 to 5	x				



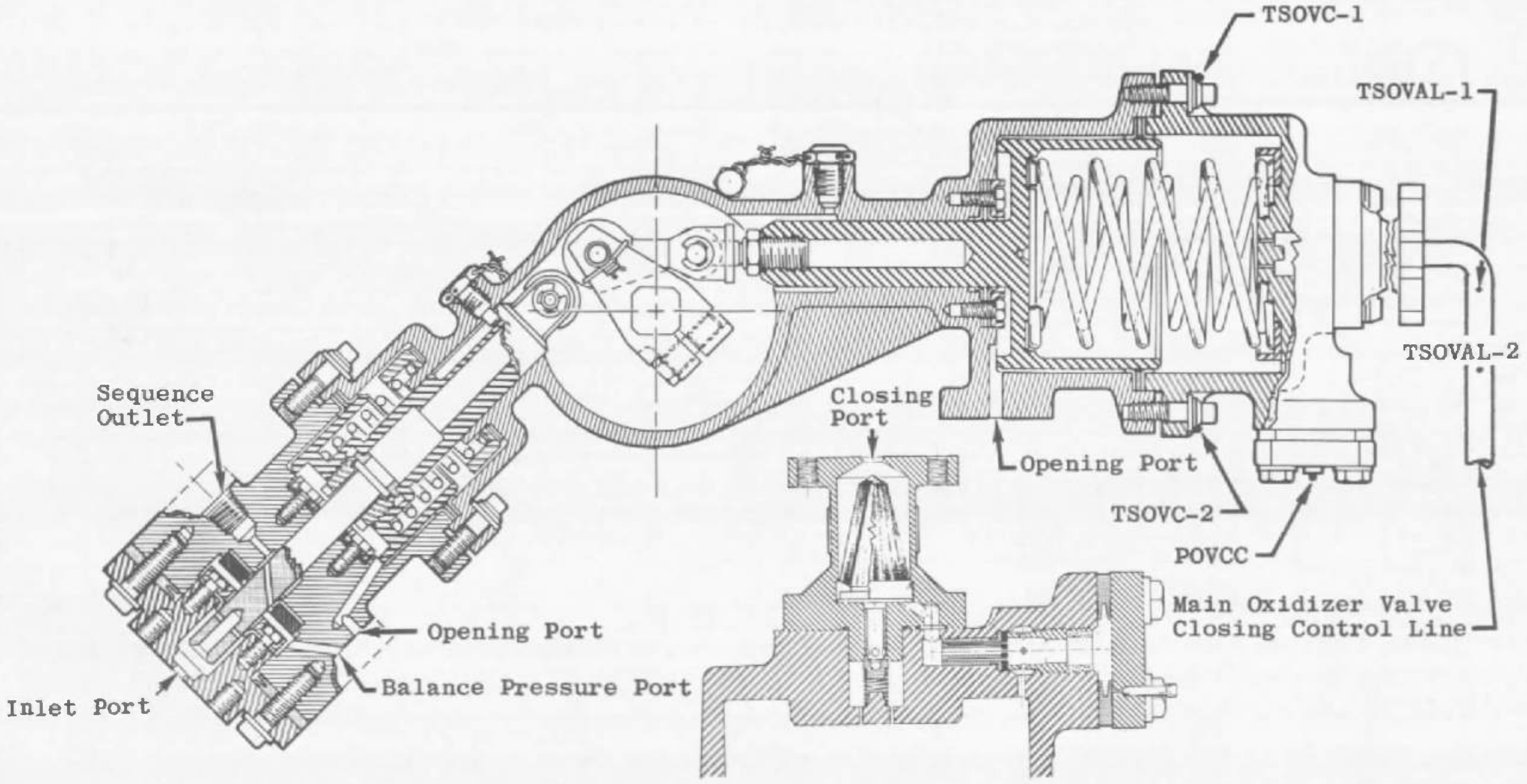
a. Engine Pressure Tap Locations

Fig. III-1 Instrumentation Locations



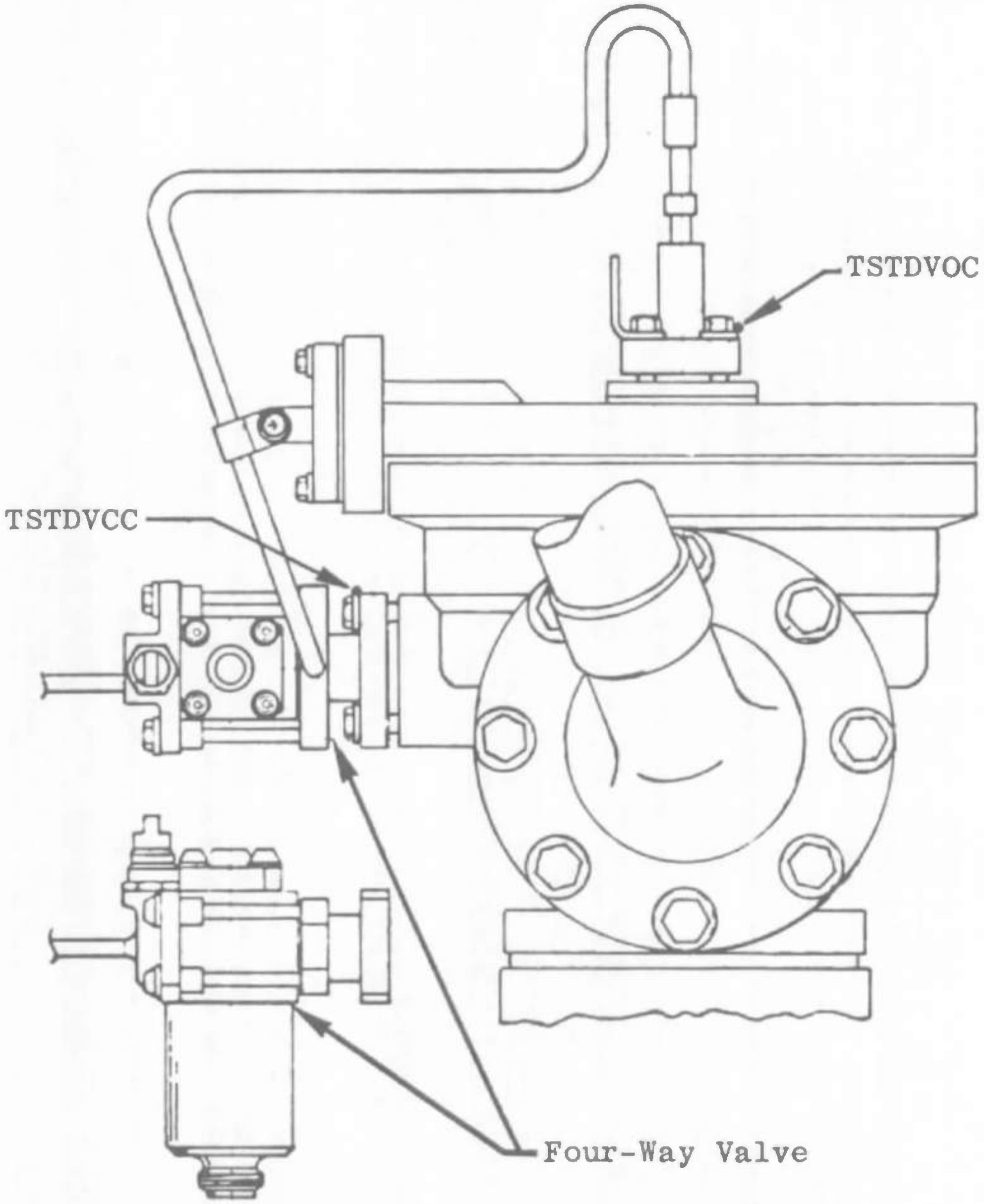
b. Engine Temperature, Flow, and Speed Instrumentation Locations

Fig. III-1 Continued

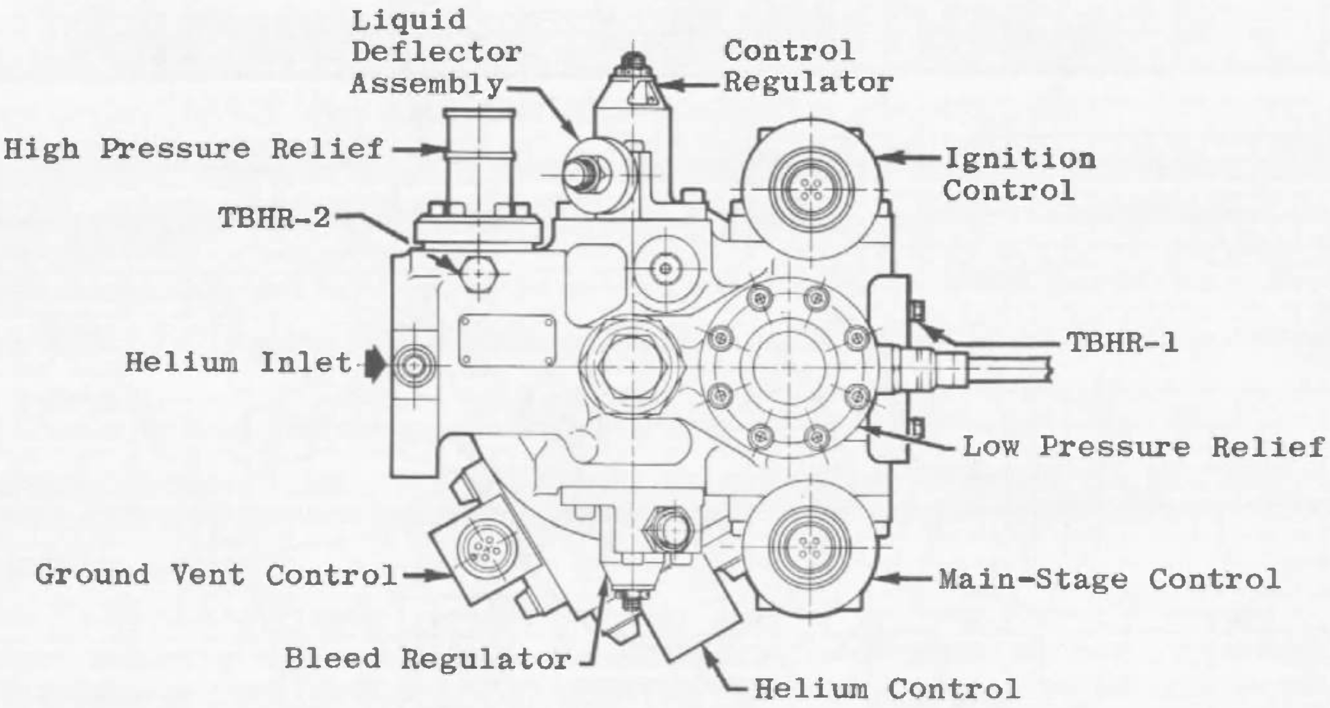


c. Main Oxidizer Valve

Fig. III-1 Continued



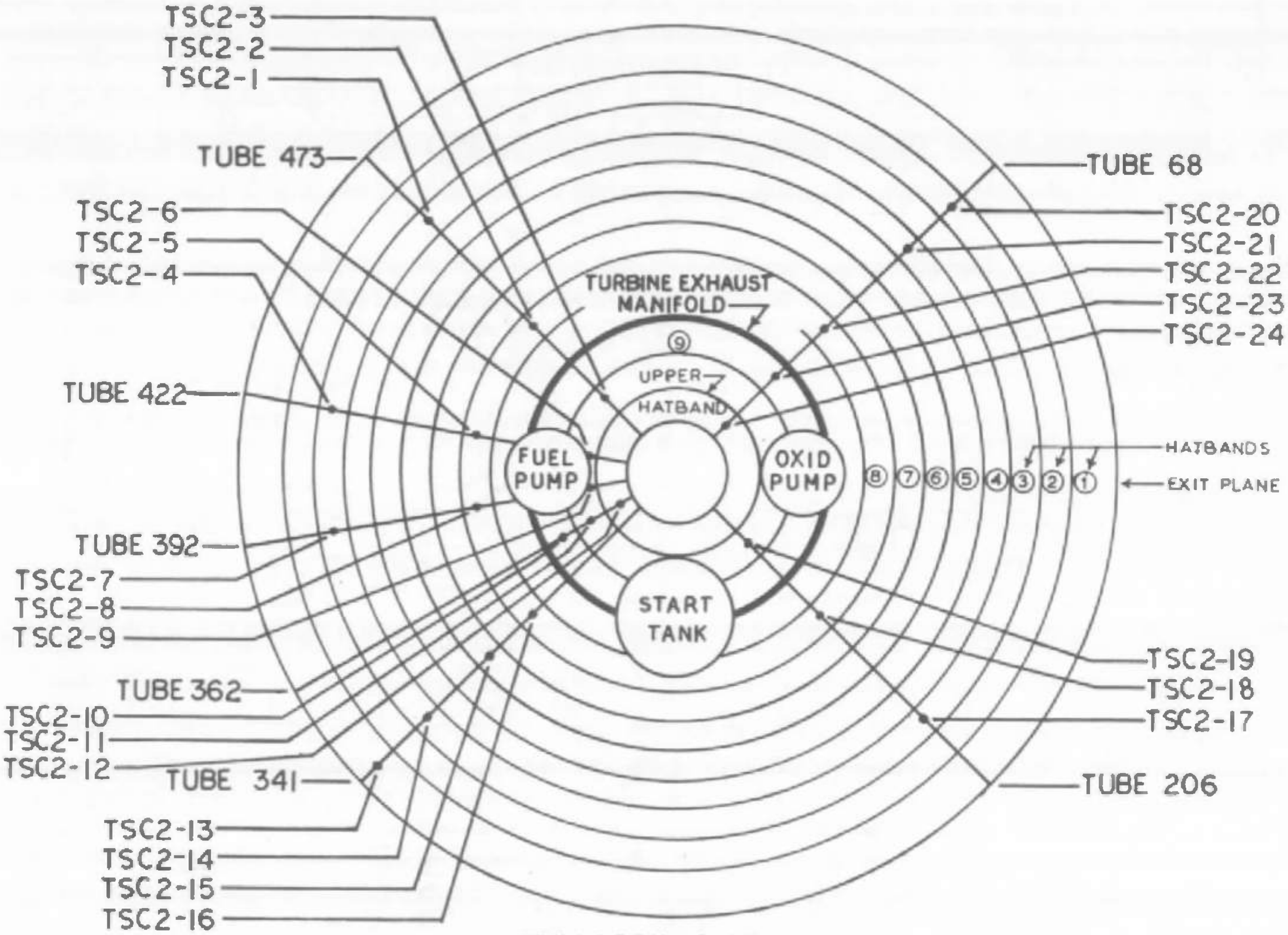
d. Start Tank Discharge Valve
Fig. III-1 Continued



Top View

e. Helium Regulator

Fig. III-1 Continued



VIEW LOOKING AFT

f. Thrust Chamber

Fig. III-1 Concluded

APPENDIX IV
METHODS OF CALCULATIONS (PERFORMANCE PROGRAM)

TABLE IV-1
PERFORMANCE PROGRAM DATA INPUTS

Item No.	Parameter
1	Thrust Chamber (Injector Face) Pressure, psia
2	Thrust Chamber Fuel and Oxidizer Injection Pressures, psia
3	Thrust Chamber Fuel Injection Temperature, °F
4	Fuel and Oxidizer Flowmeter Speeds, Hz
5	Fuel and Oxidizer Engine Inlet Pressures, psia
6	Fuel and Oxidizer Pump Discharge Pressures, psia
7	Fuel and Oxidizer Engine Inlet Temperatures, °F
8	Fuel and Oxidizer (Main Valves) Temperatures, °F
9	Propellant Utilization Valve Center Tap Voltage, volts
10	Propellant Utilization Valve Position, volts
11	Fuel and Oxidizer Pump Speeds, rpm
12	Gas Generator Chamber Pressure, psia
13	Gas Generator (Bootstrap Line at Bleed Valve) Temperature, °F
14	Fuel* and Oxidizer Turbine Inlet Pressure, psia
15	Oxidizer Turbine Discharge Pressure, psia
16	Fuel and Oxidizer Turbine Inlet Temperature, °F
17	Oxidizer Turbine Discharge Temperature, °F

*At AEDC, fuel turbine inlet pressure is estimated from gas generator chamber pressure.

NOMENCLATURE

A	Area, in.^{2}
B	Horsepower, hp
C^*	Characteristic velocity, ft/sec
C_p	Specific heat at constant pressure, Btu/lb/°F
D	Diameter, in.
H	Head, ft
h	Enthalpy, Btu/lb$_m$
M	Molecular weight
N	Speed, rpm
P	Pressure, psia
Q	Flow rate, gpm
R	Resistance, sec^2/ft^3-in.^{2}
r	Mixture ratio
T	Temperature, °F
TC^*	Theoretical characteristic velocity, ft/sec
W	Weight flow, lb/sec
Z	Pressure drop, psi
β	Ratio
γ	Ratio of specific heats
η	Efficiency
θ	Degrees
ρ	Density, lb/ft^3

SUBSCRIPTS

A	Ambient
AA	Ambient at thrust chamber exit
B	Bypass nozzle

BIR	Bypass nozzle inlet (Rankine)
BNI	Bypass nozzle inlet (total)
C	Thrust chamber
CF	Thrust chamber, fuel
CO	Thrust chamber, oxidizer
CV	Thrust chamber, vacuum
E	Engine
EF	Engine fuel
EM	Engine measured
EO	Engine oxidizer
EV	Engine, vacuum
e	Exit
em	Exit measured
F	Thrust
FIT	Fuel turbine inlet
FM	Fuel measured
FY	Thrust, vacuum
f	Fuel
G	Gas generator
GF	Gas generator fuel
GO	Gas generator oxidizer
H1	Hot gas duct No. 1
H1R	Hot gas duct No. 1 (Rankine)
H2R	Hot gas duct No. 2 (Rankine)
IF	Inlet fuel
IO	Inlet oxidizer
ITF	Isentropic turbine fuel
ITO	Isentropic turbine oxidizer
N	Nozzle
NB	Bypass nozzle (throat)

NV	Nozzle, vacuum
O	Oxidizer
OC	Oxidizer pump calculated
OF	Outlet fuel pump
OFIS	Outlet fuel pump isentropic
OM	Oxidizer measured
OO	Oxidizer outlet
PF	Pump fuel
PO	Pump oxidizer
PUVO	Propellant utilization valve oxidizer
RNC	Ratio bypass nozzle, critical
SC	Specific, thrust chamber
SCV	Specific thrust chamber, vacuum
SE	Specific, engine
SEV	Specific, engine vacuum
T	Total
T_o	Turbine oxidizer
TEF	Turbine exit fuel
TEFS	Turbine exit fuel (static)
TF	Fuel turbine
TIF	Turbine inlet fuel (total)
TIFM	Turbine inlet, fuel, measured
TIFS	Turbine inlet fuel isentropic
TIO	Turbine inlet oxidizer
t	Throat
V	Vacuum
v	Valve
XF	Fuel tank repressurant
XO	Oxidizer tank repressurant

PERFORMANCE PROGRAM EQUATIONS

MIXTURE RATIO

Engine

$$r_E = \frac{\dot{W}_{EO}}{\dot{W}_{EF}}$$

$$\dot{W}_{EO} = \dot{W}_{OM} - \dot{W}_{XO}$$

$$\dot{W}_{EF} = \dot{W}_{FM} - \dot{W}_{XF}$$

$$\dot{W}_E = \dot{W}_{EO} + \dot{W}_{EF}$$

Thrust Chamber

$$r_C = \frac{\dot{W}_{CO}}{\dot{W}_{CF}}$$

$$\dot{W}_{CO} = \dot{W}_{OM} - \dot{W}_{XO} - \dot{W}_{GO}$$

$$\dot{W}_{CF} = \dot{W}_{FM} - \dot{W}_{XF} - \dot{W}_{GF}$$

$$\dot{W}_{XO} = 0.8 \text{ lb/sec}$$

$$\dot{W}_{XF} = 1.8 \text{ lb/sec}$$

$$\dot{W}_{GO} = \dot{W}_T - \dot{W}_{GF}$$

$$\dot{W}_{GF} = \frac{\dot{W}_T}{1 + r_G}$$

$$\dot{W}_T = \frac{P_{TIF} A_{TIF} K_7}{TC^*_{TIF}}$$

$$K_7 = 32.174$$

$$\dot{W}_C = \dot{W}_{CO} + \dot{W}_{CF}$$

CHARACTERISTIC VELOCITY

Thrust Chamber

$$C^* = \frac{K_7 P_c A_t}{\dot{W}_C}$$

$$K_7 = 32.174$$

DEVELOPED PUMP HEAD

Flows are normalized by using the following inlet pressures, temperatures, and densities.

$P_{IO} = 39$ psia

$P_{IF} = 30$ psia

$\rho_{IO} = 70.79\ \mathrm{lb/ft^3}$

$\rho_{IF} = 4.40\ \mathrm{lb/ft^3}$

$T_{IO} = -295.212\,^\circ\mathrm{F}$

$T_{IF} = -422.547\,^\circ\mathrm{F}$

Oxidizer

$$H_O = K_4 \left(\frac{P_{OO}}{\rho_{OO}} - \frac{P_{IO}}{\rho_{IO}}\right)$$

$K_4 = 144$

ρ = National Bureau of Standards Values f (P,T)

Fuel

$$H_f = 778.16\ \Delta h_{OFIS}$$

$$\Delta h_{OFIS} = h_{OFIS} - h_{IF}$$

$$h_{OFIS} = f(P,T)$$

$$h_{IF} = f(P,T)$$

PUMP EFFICIENCIES

Fuel, Isentropic

$$\eta_f = \frac{h_{OFIS} - h_{IF}}{h_{OF} - h_{IF}}$$

$$h_{OF} = f(P_{OF}, T_{OF})$$

Oxidizer, Isentropic

$$\eta_O = \eta_{OC}\, Y_O$$

$$\eta_{OC} = K_{40}\left(\frac{Q_{PO}}{N_O}\right)^2 + K_{50}\left(\frac{Q_{PO}}{N_O}\right) + K_{60}$$

$K_{40} = 5.0526$

$K_{50} = 3.8611$

$K_{60} = 0.0733$

$Y_O = 1.000$

TURBINES

Oxidizer, Efficiency

$$\eta_{TO} = \frac{B_{TO}}{B_{ITO}}$$

$$B_{TO} = K_5 \frac{W_{PO} H_O}{\eta_O}$$

$$K_5 = 0.001818$$

$$W_{PO} = W_{OM} + W_{PUVO}$$

$$W_{PUVO} = \sqrt{\frac{Z_{PUVO} \rho_{OO}}{R_v}}$$

$$Z_{PUVO} = A + B (P_{OO})$$

$$A = -1597$$

$$B = 2.3828$$

$$\text{IF } P_{OO} \geq 1010 \text{ Set } P_{OO} = 1010$$

$$\ln R = A_3 + B_3 (\theta_{PUVO}) + C (\theta_{PUVO})^3 + D_3 (e)^{\frac{\theta_{PUVO}}{7}} + E_3 (\theta_{PUVO}) (e)^{\frac{\theta_{PUVO}}{7}} + F_3 \left[(e)^{\frac{\theta_{PUVO}}{7}} \right]^2$$

$$A_3 = 5.5659 \times 10^{-1}$$

$$B_3 = 1.4997 \times 10^{-2}$$

$$C_3 = 7.9413 \times 10^{-6}$$

$$D_3 = 1.2343$$

$$E_3 = -7.2554 \times 10^{-2}$$

$$F_3 = 5.0691 \times 10^{-2}$$

$$\theta_{PUVO} = 16.5239$$

Fuel, Efficiency

$$\eta_{TF} = \frac{B_{TF}}{B_{ITF}}$$

$$B_{ITF} = K_{10} \Delta h_f W_T$$

$$\Delta h_f = h_{TIF} - h_{TEF}$$

$$B_{TF} = B_{PF} = K_5 \left(\frac{W_{PF} H_f}{\eta_f} \right)$$

$$W_{PF} = W_{FM}$$

$$K_{10} = 1.4148$$

$$K_5 = 0.001818$$

Oxidizer, Developed Horsepower

$$B_{TO} = B_{PO} + K_{56}$$

$$B_{PO} = K_5 \; \frac{W_{PO} \; H_O}{\eta_O}$$

$$K_{56} = -15$$

Fuel, Developed Horsepower

$$B_{TF} = B_{PF}$$

$$B_{PF} = K_5 \; \frac{W_{PF} \;\; H_f}{\eta_f}$$

$$W_{PF} = W_{FM}$$

Fuel, Weight Flow

$$W_{TF} = W_T$$

Oxidizer Weight Flow

$$W_{TO} = W_T - W_B$$

$$W_B = \left[\frac{2K_7 \quad H_2}{\gamma_{H2-1}} \left(P_{RNC}\right)^{\frac{2}{\gamma_{H2}}} \right]^{½} \left[1 - \left(P_{RNC}\right)^{\frac{\gamma_{H2-1}}{\gamma_{H2}}} \right] \frac{A_{NB} \;\; P_{BNI}}{\left(R_{H2} T_{BIR}\right)^{½}}$$

$$P_{RNC} = f \; (\beta_{NB}, \gamma_{H2})$$

$$\beta_{NB} = \frac{D_{NB}}{D_B}$$

$$\gamma_{H2}, M_{H2} = f(T_{H2R}, R_G)$$

$$A_{NB} = K_{13} \; D_{NB}$$

$$K_{13} = 0.7854$$

$$T_{BIR} = T_{TIO} + 460$$

$$P_{BNI} = P_{TEFS}$$

$$P_{TEFS} = \text{Iteration of } P_{TEF}$$

$$P_{TEF} = P_{TEFS} \left[1 + K_8 \left(\frac{W_T}{P_{TEFS}} \right)^2 \frac{T_{H2R}}{D^4_{TEF} \; M_{H2}} \left(\frac{\gamma_{H2-1}}{\gamma_{H2}} \right) \right]^{\frac{\gamma_{H2}}{\gamma_{H2-1}}}$$

$$K_8 = 38.8983$$

GAS GENERATOR

Mixture Ratio

$$r_G = D_1 (T_{H1})^3 + C_1 (T_{H1})^2 + B_1 (T_{H1}) + A_1$$

$A_1 = 0.2575$

$B_1 = 5.586 \times 10^{-4}$

$C_1 = -5.332 \times 10^{-9}$

$D_1 = 1.1312 \times 10^{-11}$

$T_{H1} = T_{TIFM}$

Flows

$$TC^*_{TIF} = D_2 (T_{H1})^3 + C_2 (T_{H1})^2 + B_2 (T_{H1}) + A_2$$

$A_2 = 4.4226 \times 10^3$

$B_2 = 3.2267$

$C_2 = -1.3790 \times 10^{-3}$

$D_2 = 2.6212 \times 10^{-7}$

$$P_{TIF} = P_{TIFS} \left[1 + K_8 \left(\frac{W_T}{P_{TIFS}} \right)^2 \frac{T_{H1R}}{D^4_{TIF} M_{H1}} \frac{\gamma_{H1} - 1}{\gamma_{H1}} \right]^{\frac{\gamma_{H1}}{\gamma_{H1} - 1}}$$

$K_8 = 38.8983$

Note: P_{TIF} is determined by iteration.

$T_{HIR} = T_{TIF}$

$M_{H1}, Y_{H1}, C_p, r_{H1} = f\ (T_{HIR}, r_G)$

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13. ABSTRACT

Two firings of the Rocketdyne J-2 rocket engine were conducted in Test Cell J-4 of the Large Rocket Facility. The firings were accomplished during test period J4-1801-03 at pressure altitudes of approximately 100,000 ft to evaluate S-IVB/S-V start condition effects on (1) engine start transients, (2) gas generator outlet temperature, (3) augmented spark igniter operation, and (4) fuel pump low level stall margin for J-2 engine J-2052. Engine components were thermally conditioned to temperatures predicted for S-IVB/S-V first burn and one orbit restart on firings 03A and 03B, respectively. Excessive gas generator outlet temperature peaks were experienced on these firings, resulting in a gas generator outlet temperature probe failure and premature engine shutdown on firing 03B. The two remaining scheduled firings for the test period were cancelled. Post-test inspection revealed erosion of the fuel turbine first stage, and small cracks were found in the curvic coupling of fuel turbine first-stage wheel. The accumulated engine firing duration was 31.3 sec.

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14 KEY WORDS	LINK A		LINK B		LINK C	
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ignition effects						
temperature						
stall						
gas generator						
fuel pump						
Saturn						
Apollo						
simulated altitude tests						

1. Rocket motors -- J-2

2. " " -- Performance

16. 3.