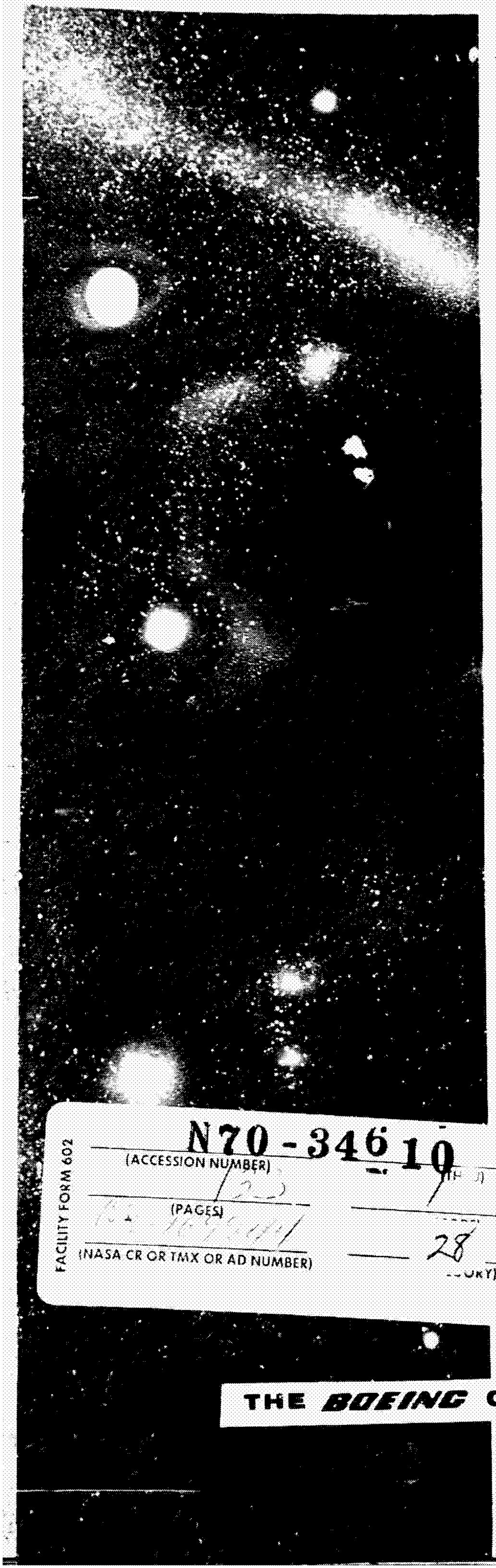

ENGINE RESTART AND THERMODYNAMIC
ANALYSIS OF APOLLO SPACECRAFT
ENGINE TESTS (VOLUME II)

THE BOEING COMPANY · AEROSPACE GROUP
SOUTHEAST DIVISION

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Prepared by:

PROPULSION AND POWER

5-2940

Prepared for:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

Prepared by: B. F. Kerkam
A. A. Wood
M. G. Foley
D. W. Smith

Approved by: 8/26/69
A. M. Momenthy
Propulsion Performance

Approved by: J M Carter
J. M. Carter (5-2940)
Propulsion and Power Systems

THE BOEING COMPANY SPACE DIVISION, HOUSTON, TEXAS

REVISIONS

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ABSTRACT

This report presents analyses of results obtained from cold flow and hot firing tests conducted on the LM Ascent, LM Descent, and Service Propulsion System engines. The purposes of these analyses were to provide a basis for defining the thermodynamic processes and hardware variables that lead to a hypergol engine hard restart after a short duration firing, and to determine if cold flow tests can be formulated to provide data that will allow a reduction in the requirements for costly hypergol engine hot firing restart performance tests. Engine restart characteristics were related to the results from thermodynamic analyses of propellant behavior during the coast phase of cold flow and hot firing tests. Hard restarts were related to the accumulation of frozen propellants in the injector assemblies and the accumulation of nitrates in the thrust chambers. Recommendations were made concerning the conduct of future cold flow test programs and for further experimental and analytical investigations. Details of the test programs are covered in these appendices.

KEY WORDS

Aerojet
Aerozine-50
Arnold Engineering Development Center
Atlantic Research Corporation
Boeing Tulalip Test Site
Chamber Pressure Overshoot
Hydrazine
Injector
Injector Freezing
LM Ascent Engine
LM Descent Engine
Mixed Hydrazines
Nitrogen Tetroxide
Propellant Temperature
Propellant Thermodynamics
Restart
Rocket Engine Restart
Rocketdyne
Service Propulsion System Engine
TRW
UDMH

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DEFINITIONS

AEDC	Arnold Engineering Development Center
ARC	Atlantic Research Corporation
Bell	Bell Aerospace Corporation
TRW	TRW Systems Group
SPS	Service Propulsion System
APS	Ascent Propulsion System
DPS	Descent Propulsion System
LMAE	Lunar Module Ascent Engine
LMDE	Lunar Module Descent Engine
GAEC	Grumman Aircraft Engineering Corporation
GAC	Grumman Aerospace Corporation

Dry Start	An engine start mode with no fuel or oxidizer between the engine ball valves, and (APS and DPS only) no fuel in the valve actuator lines.
Wet Start	An engine start mode with propellants between the engine ball valves and (APS and DPS) fuel in the valve actuator lines.
Initial Start	The first start of a test series.
Hard Start	An engine firing which produced higher chamber pressure and accelerometer readings at ignition than initial starts made at the same test conditions.

NOMENCLATURE

m	mass
C_p	specific heat capacity
T	temperature (absolute)
τ	time
h_{lv}	enthalpy of vaporization
e	base of natural logarithms
S	solubility
p	pressure
scc	standard cubic centimeters
C_D	discharge coefficient
A	area
γ	ratio of specific heats, C_p/C_v
n	molecular weight
R_0	gas constant (universal)

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APPENDIX A - INJECTOR COLD-FLOW TESTING AT ARC

A.0 GENERAL

This appendix describes the Atlantic Research Corporation (ARC) test facility used for injector cold-flow testing. The test series and engine instrumentation are described, and an evaluation of the cold-flow testing is presented. Detailed information on the test facility, test article, instrumentation and test series conduct is presented in References 1 and 2.

A.1 TEST FACILITY AND TEST APPARATUS

A.1.1 Facility

The stainless steel altitude chamber used during the ARC injector tests was 6 feet in diameter and 25 feet long. The chamber was evacuated by a 5 stage steam ejector system designed to produce a no-load altitude of 245,000 feet, and to maintain an altitude of 200,000 feet with a 25 gram/second mass influx rate.

The propellant supply system for the Bell ascent injector tests is shown in Figure A-1. The fuel tank was pressurized with helium gas, and was thermally conditioned by immersion in a temperature-controlled water/alcohol bath. The fuel supply line was one inch in diameter and was thermally conditioned by coils through which the cooled water/alcohol solution was circulated. A short two inch diameter fuel line section was equipped with an orifice flowmeter and a delta-p transducer to measure flow as a function of time. The oxidizer/oxidizer simulant tank was located inside the altitude chamber and was pressurized with nitrogen. Thermal conditioning was provided by water/alcohol solution which was circulated through tubing coils wrapped around the exterior of the tank and the oxidizer feed lines. The quantity of oxidizer or oxidizer simulant used during a test was determined by observing the oxidizer tank sight glass before and after each pulse. The level of propellant saturation with pressurization gases was neither controlled nor measured during the test series. The Bell ascent injector was cooled to 40°F, when required, by using trickle flows of methyl alcohol and Freon MF to cool the fuel and oxidizer passages respectively.

Prior to the TRW descent injector and Rocketdyne ascent injector tests, the oxidizer supply system was modified to make it similar to the fuel supply system. A schematic of the modified system is shown in Figure A-2. The major changes to the oxidizer supply system included relocating the oxidizer tank outside of the altitude chamber, providing a temperature-controlled bath for the oxidizer tank and installing an orifice flow meter in the 2 inch diameter section of the oxidizer supply line. A nitrogen gas valve actuation system produced engine valve opening times of approximately 200 milliseconds and closing times of approximately 170 ms during the TRW descent injector tests.

The TRW descent injector was cooled to 40°F, when required, by spraying isopropyl alcohol from two circular manifolds onto the valve side of the injector. Two heat lamps were aimed at the lower area of the injector manifolds to compensate for the overcooling which occurred when alcohol drained down and collected near the bottom of the injector plate. The Rocketdyne injector was cooled to 40°F, when required, by using trickle flows of methyl alcohol and Freon MF to cool the fuel and oxidizer passages respectively.

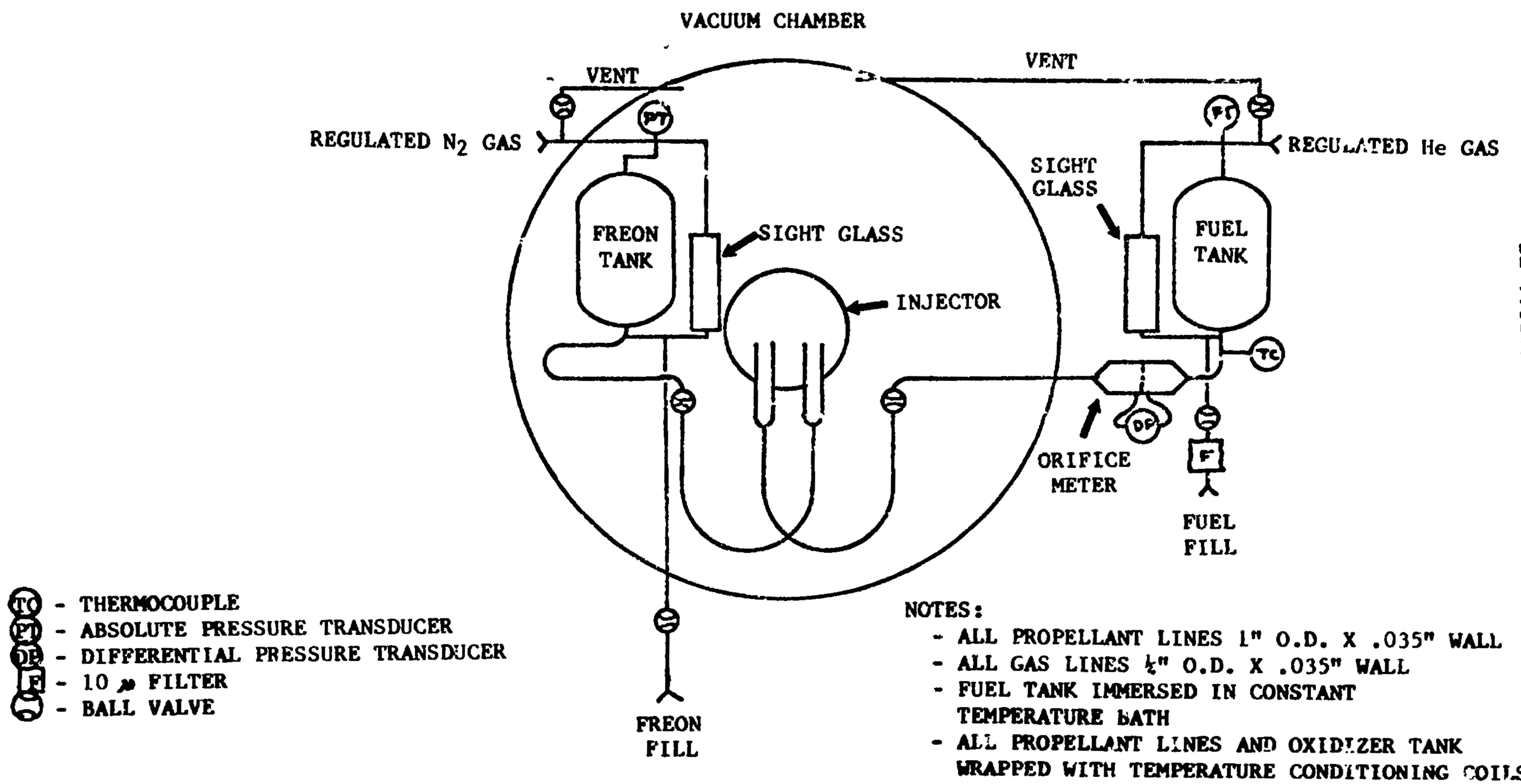
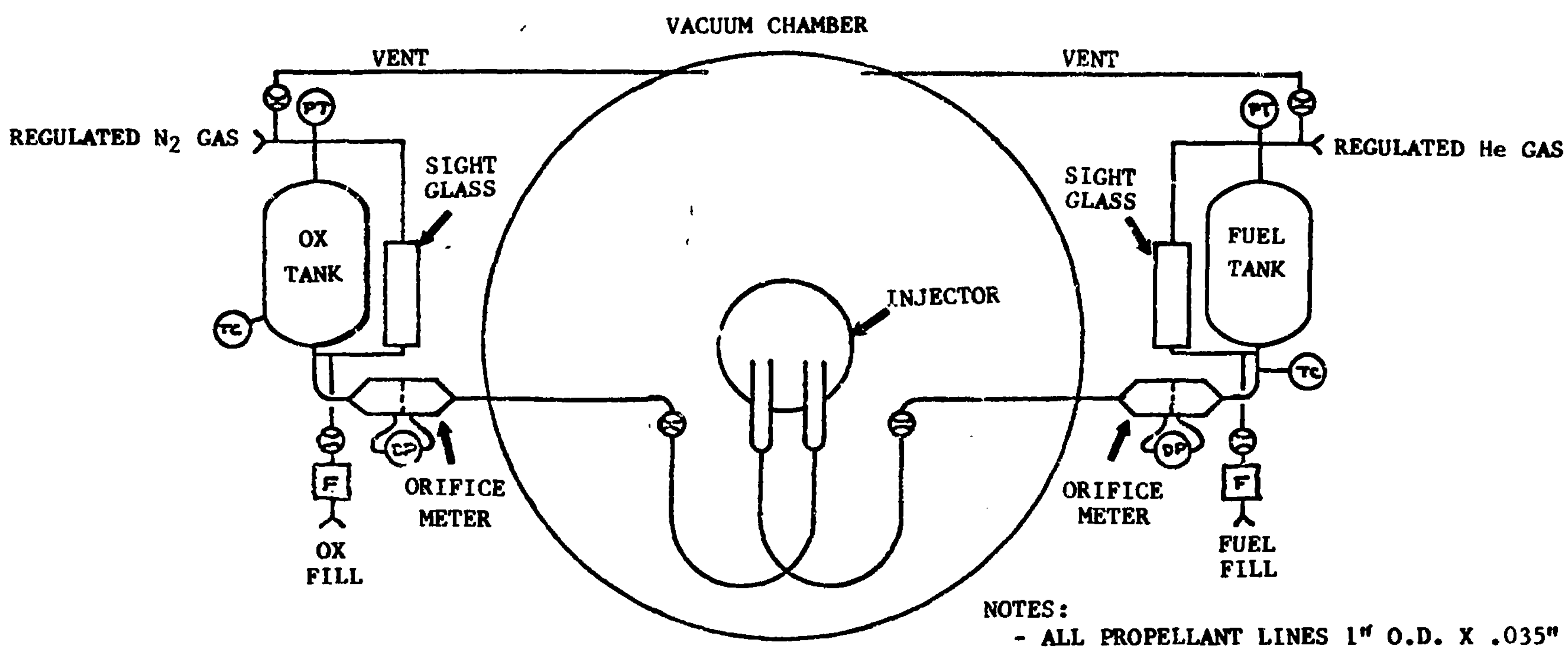


FIGURE A - 1 PROPELLANT SUPPLY SYSTEM, BELL INJECTOR PHASE I TESTS



NOTES:
- ALL PROPELLANT LINES 1" O.D. X .035" WALL
- ALL GAS LINES ¼" O.D. X .035" WALL
- BOTH TANKS IMMERSED IN CONSTANT TEMPERATURE BATHS
- ALL PROPELLANT LINES WRAPPED WITH TEMPERATURE CONDITIONING COILS

TC - THERMOCOUPLE
PT - ABSOLUTE PRESSURE TRANSDUCER
DP - DIFFERENTIAL PRESSURE TRANSDUCER
F - 10 μ FILTER
- BALL VALVE

FIGURE A - 2 MODIFIED PROPELLANT SUPPLY SYSTEM

A.1.2 Distillation Apparatus

A vacuum distillation apparatus was used to determine the volume of propellant mass residuals located in the injectors and injector manifolds. A schematic drawing of the apparatus used with all three of the injectors is shown in Figure A-3. The following procedure was used to determine the propellant residual volumes. At a preselected time interval after a propellant cold flow, the test chamber pressure was brought up to ambient, and a cap was placed over the injector. The distillation apparatus was connected to the fuel injector manifold, and the residual propellants were drawn off into the evacuated cold traps, labeled A, B, and C on Figure A-3. The traps were maintained at approximately 0°C, -60°C, and -200°C by using ice, dry ice, and liquid nitrogen respectively. At preselected time intervals, the traps were removed, the volume of residuals was determined, and the traps were reinstalled for further distillation.

The propellant distillation procedure was evaluated prior to starting Phase II of the Bell ascent injector tests. Since the procedure was not evaluated prior to the Phase I tests, the accuracy of the Phase I tests is not known. Data for the first ten evaluation runs are shown on Table A-1. The duration of the distillation was 30 minutes. Beginning with test #8, the distillation procedure was changed because strong hydrazine odors were observed at the vacuum pump exhaust, and a large fraction of the distillate was recovered in the coldest trap (trap "C"). The procedure change involved keeping the stopcock between traps "B" and "C" closed during most of the distillation period. Although this procedure change reduced the repeatability of the distillation procedure, it was retained for safety reasons.

A second procedure change was incorporated after test #10. The distillation was performed in two periods, lasting 30 minutes and 20 minutes. The amount recovered during the final 20 minute period was always less than 10% of the amount recovered during the first 30 minute period, indicating that the distillation was essentially complete. In addition, the tubing between the injector and the traps was heated to reduce the possibility of condensate collecting in the line. This procedure, which gave recovery rates of 85 $\pm$ 5% of known samples, was used for the Phase II Bell injector tests. The reported mass residual data are not corrected for the accuracy of the recovery process.

Prior to the TRW descent injector tests, the distillation procedure was revised to incorporate at least two 30 minute distillation periods, and the dry ice trap was replaced by a liquid nitrogen trap. The 30 minute distillation periods were repeated until no further material condensed in the traps. Data obtained during the evaluation of this procedure are given in Table A-1. This procedure was used for the Rocketdyne ascent injector tests, but no separate evaluation of the

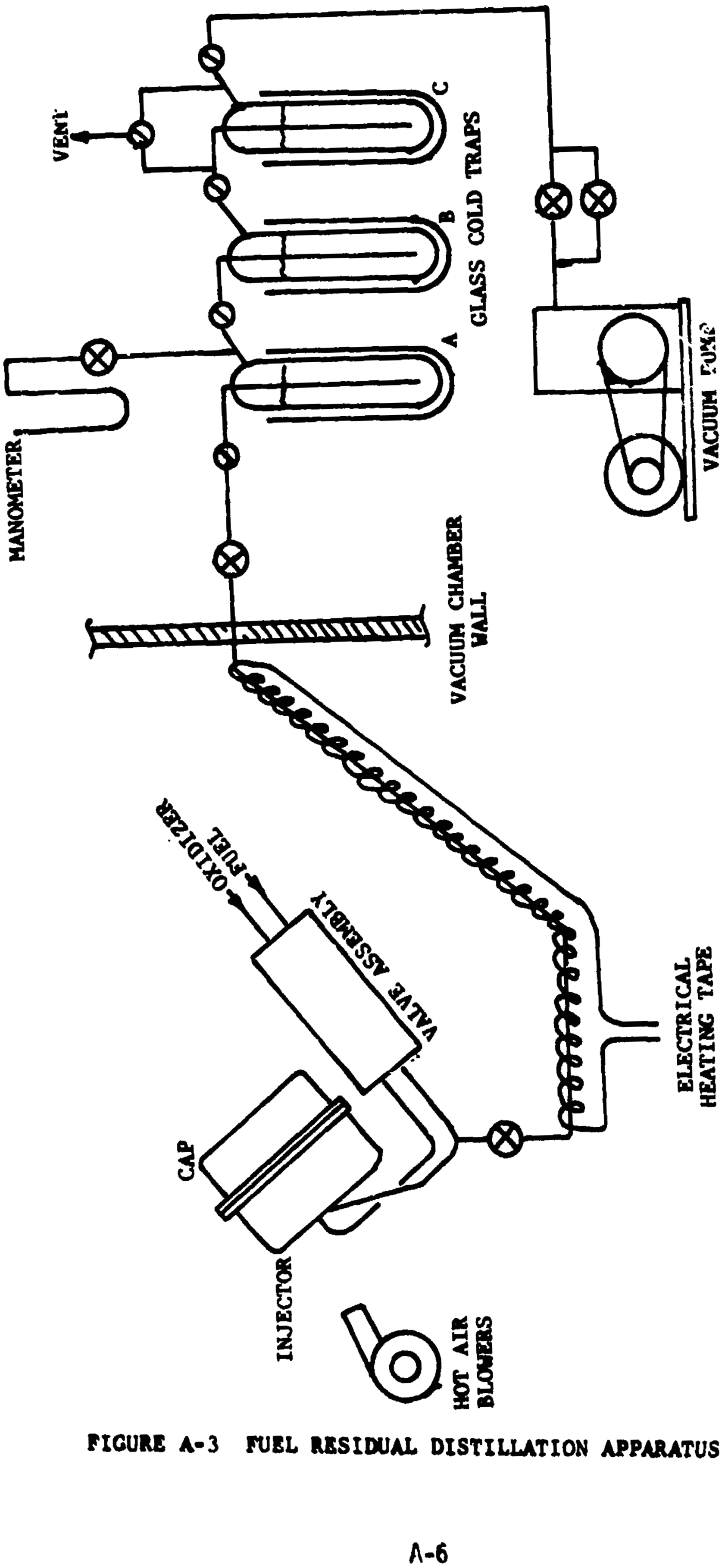


FIGURE A-3 FUEL RESIDUAL DISTILLATION APPARATUS

TABLE A-1

DISTILLATION APPARATUS CALIBRATION DATA

BELL ASCENT INJECTOR TESTS

TEST NUMBER	TIME FOR DISTILLATION (MIN.)	SAMPLE AMOUNT (ML)	AMOUNT RECOVERED (ML)	AMOUNT RECOVERED (PER CENT)
1	5	9.8	7	71
2	5	50	23.0	46
3	10	50	40.0	80
4	30	50	47.4	95
5	30	50	48.8	98
6	30	212	196	92
7	30	100	99	99
8	30	50	42.4	85
9	30	50	40.4	81
10	30	10	9.8	98

TRW DESCENT INJECTOR TESTS

TEST NUMBER	TIME FOR DISTILLATION (MIN.)	SAMPLE AMOUNT (ML)	AMOUNT RECOVERED (ML)	AMOUNT RECOVERED (PER CENT)
1	-	100	95	95
2	-	100	95	95
3	-	10	8	80
4	-	10	8	80

A.1.2 Distillation Apparatus (Continued)

distillation process was made for the Rocketdyne injector. The reported mass residual data are not corrected for the accuracy of the distillation procedure.

A.1.3 Instrumentation

Thermocouples were installed at several positions on the injector and valve assemblies to obtain temperature histories. Detailed thermocouple locations are given in the test article description section for each injector. Some of the thermocouples were installed directly on the exterior of the injectors and others were installed so that they protruded into the propellant passages. Pressure transducers were installed to provide pressure histories of the injector manifolds, the feed lines and altitude chamber. All pressure and temperature transducer data were recorded on a 24 channel Honeywell Model 1508 oscillograph. Movie cameras were mounted in positions to obtain photographic coverage of the injector face for each of the injectors tested. For the beaker tests, the movie cameras were installed to obtain a side view of the beaker.

A.2 BELL ASCENT ENGINE INJECTOR TESTS, PHASE I

A.2.1 Test Objectives

The objectives of the Phase I cold-flow tests conducted with a Bell ascent injector were:

1. Determine extent of injector orifice obstruction from frozen propellant to support LM-1 mission.

2. Determine quantity of residual propellants.

3. Obtain photographic records of the phenomena occurring during propellant flow and coast periods.

A.2.2 Test Article Configuration and Instrumentation

The test article consisted of an instrumented Bell ascent engine injector, propellant ducts and quad-redundant propellant shut-off valve assembly. The injector valves were actuated by a solenoid-controlled nitrogen gas system, and were protected from the injector propellant spray by a splash plate. The injector and valve assembly was fastened to structural supports which were attached to the side walls of the ARC High Altitude Test Facility. The test assembly was oriented so that the injector faced upward at 30° from the horizontal, as shown on Figure A-4. Propellant supply system characteristics are described in Section A.1 and Figure A-1 of this appendix.

The location of temperature and pressure transducers is shown in Figure A-5. Copper-constantan thermocouples were used for all temperature measurements. Fuel manifold pressures were measured with a 0-250 psia transducer. Because the accuracy of the manifold pressure measurements in the low pressure range (less than 10 psia) was questionable, no fuel manifold pressure data were reported for the Phase I tests. High speed (1000 frames per second) and low speed (24 fps) color movies were taken of all Phase I tests.

A.2.3 Test Series Description

A total of 27 injector and 4 beaker cold flow tests were conducted during the period from October 17, 1967 through December 8, 1967. The basic characteristics of these tests are summarized in Table A-2. Qualitative data to support the LM-1 mission was obtained from the series of two-pulse tests having an 85 minute coast period between pulses. An additional test series, consisting of five fuel flow pulses, separated by 5 minute coast periods, was run to determine the thermal conditions and duty cycle required to produce injector orifice blockage by frozen fuel.

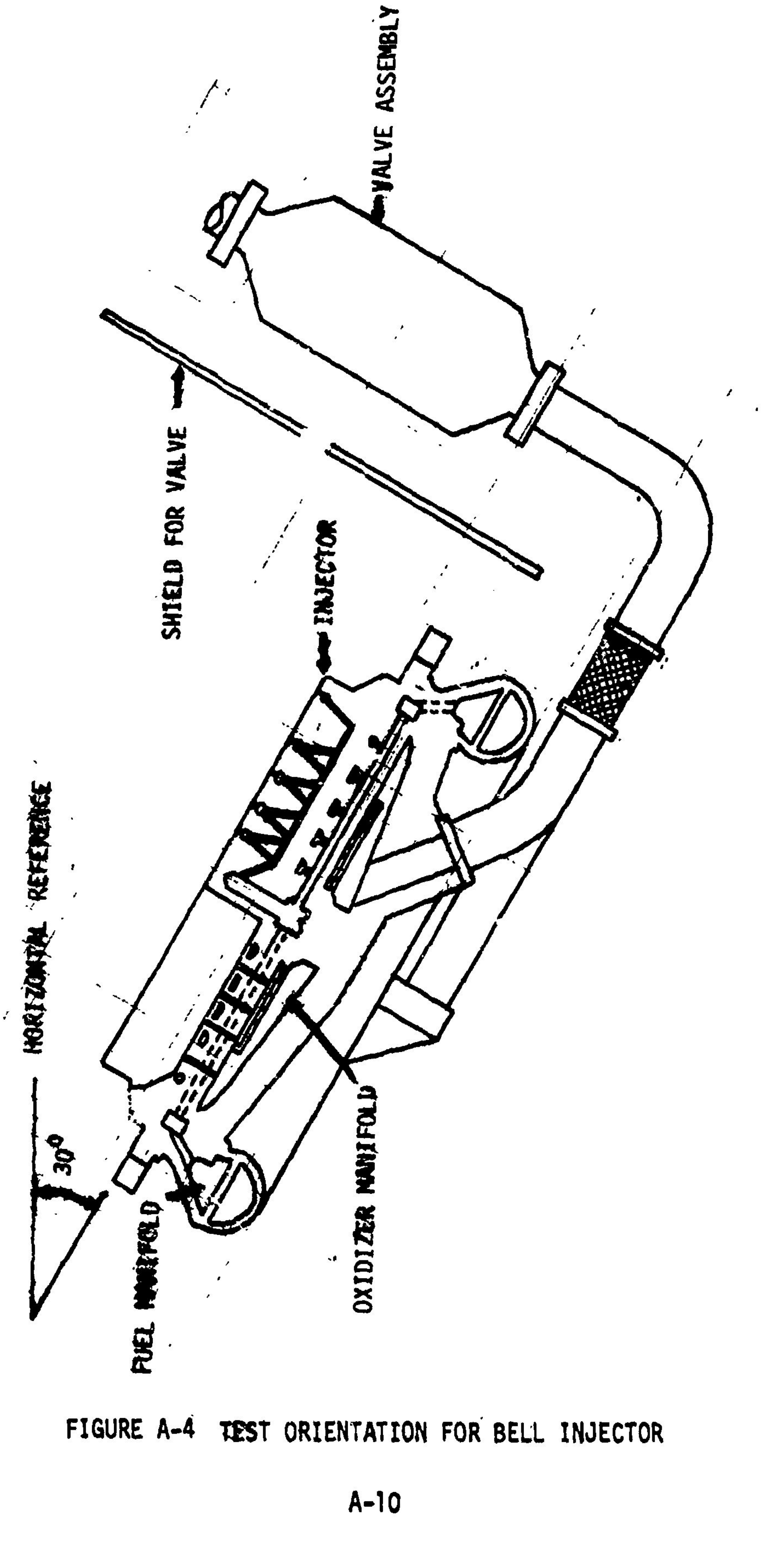


FIGURE A-4 TEST ORIENTATION FOR BELL INJECTOR

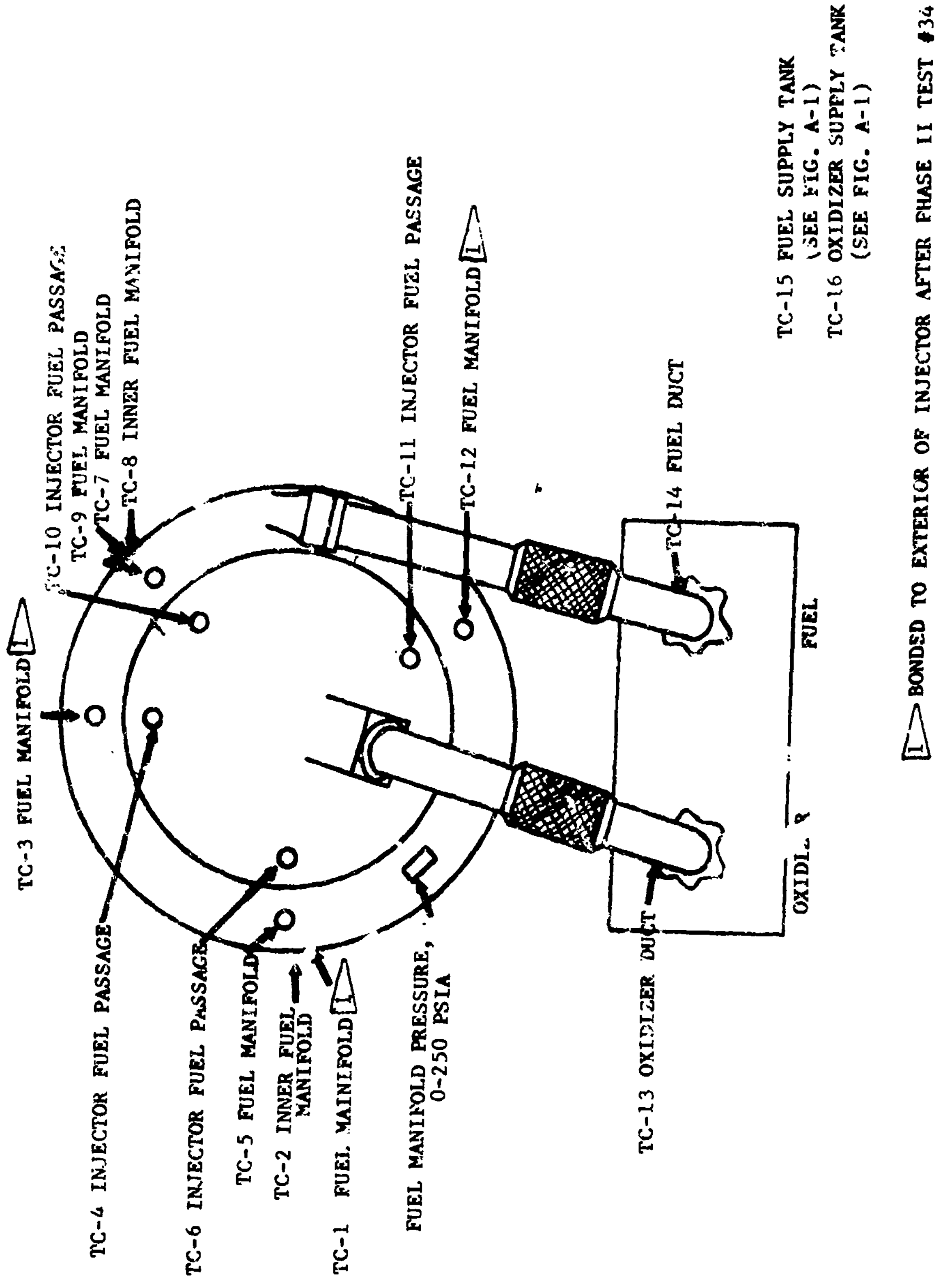


FIGURE A-5 INSTRUMENTATION LOCATIONS ON BELL INJECTOR

TEST NUMBER	FLUIDS	NOMINAL INITIAL TEMPERATURE (°F) ⚠1	PROPELLANT PULSE DURATION (SECONDS)	FIRST COAST PERIOD	SECOND COAST PERIOD	DATA AVAILABLE ⚠2	REMARKS
1	A-50	NA/65	0.6	85 min.	none	M	No attenuation of fuel flow
2	A-50	NA/64	0.6	85 min.	none	M	No attenuation of fuel flow
3	A-50, Freon MF	NA/64	0.6	85 min.	none	M	785 cc Freon trickle flow No attenuation of fuel flow
4	A-50	75/65	0.6	85 min.	none	T, M	No attenuation of fuel flow
5	A-50, Freon MF	70/65	0.6	85 min.	none	M	1130 cc Freon trickle flow No attenuation of fuel flow
6	A-50, Freon MF	55/65	0.6	85 min.	none	T, M	1130 cc Freon trickle flow No attenuation of fuel flow
7	A-50	30/67	0.6	5 min.	5 min. ⚠3	T, M	No attenuation of fuel flow
8	A-50	NA/NA				M	Test aborted after 2 pulses, facility vacuum lost
9	A-50	60/45	0.6	5 min.	5 min. ⚠3	T, M	No attenuation of fuel flow
10	A-50	103/30	0.6	5 min.	5 min. ⚠3	T, M	No attenuation of fuel flow
11	A-50, Freon MF	NA/NA	0.6	5 min.	5 min. ⚠3	none	Ran out of fuel on 4th pulse Data not reduced
12	A-50, Freon MF	35/46	0.6	5 min.	5 min. ⚠3	T, M	1070 ml Freon trickle during each pulse. Reduced fuel flow on 3rd pulse, no flow on 4th and 5th pulses
13	A-50, Freon MF	50/62	0.6	5 min.	5 min. ⚠3	T, M	1070 ml Freon trickle during each pulse. Reduced fuel flow during 4th and 5th pulses
14	A-50	62/65	0.6	5 min.	none	T, R, M	5 cc residual
15	A-50	65/42	0.6	5 min.	none	T, R, M	Trace residual
16	A-50	63/42	0.6	4 min.	none	T, R, M	22 cc residual

TABLE A-2
BELL INJECTOR PHASE I TEST SUMMARY

TEST NUMBER	FLUIDS	NOMINAL INITIAL TEMPERATURE (°F) [1]	PROPELLANT PULSE DURATION (SECONDS)	FIRST COAST PERIOD	SECOND COAST PERIOD	DATA AVAILABLE [2]	REMARKS
17	A-50	20/62	0.6	4 min.	none	T, R, M	7 cc residual
18	A-50, Freon MF	60/46	0.6	5 min.	5 min. [3]	T, R, M	535 ml Freon trickle during first 4 pulses. Reduced fuel flow during 3rd and 4th pulses, essentially no flow on 5th pulse
19	A-50, Freon MF	60/46	0.6	5 min.	5 min. [3]	T, R, M	535 ml Freon trickle during first 4 pulses. Same results as test 18
20	A-50, Freon MF	NA/NA	0.6	5 min.	5 min. [3]	none	Inadvertent Freon flow into fuel manifold - data not reduced
21	A-50, Freon MF	22/46	0.6	5 min.	5 min. [3]	T, R, M	535 ml Freon trickle during first 4 pulses. Reduced fuel flow during 3rd and 4th pulses, essentially no flow on 5th pulse.
22	N_2O_4	NA/NA	0.6	85 min.	none	none	Test terminated after one pulse
23	N_2O_4	50/40	0.6	85 min.	4-1/2 min.	T, R, M	No oxidizer flow attenuation
24	N_2O_4	70/40	0.6	5 min.	5 min. [3]	T, M, R	No attenuation of oxidizer flow
25	N_2O_4	NA/50	-	-	none	M	Beaker test Insufficient lighting - movies do not show details
26	A-50	-	none	-	none	M	Beaker test
27	A-50	-	none	-	none	M	Beaker test
28	N_2O_4	75/45	0.48	5 min.	5 min. [3]	T, M	No attenuation of oxidizer flow
29	N_2O_4	57/42	0.48	5 min.	5 min. [3]	T, M	No attenuation of oxidizer flow
30	A-50	-	none	-	none	M	Beaker test

TABLE A-2
BELL INJECTOR PHASE I TEST SUMMARY (CONTINUED)

TEST NUMBER	FLUIDS	NOMINAL INITIAL TEMPERATURE (°F) [1]	PROPELLANT PULSE DURATION (SECONDS)	FIRST COAST PERIOD	SECOND COAST PERIOD	DATA AVAILABLE [2]	REMARKS
31	N_2O_4	70/42	0.48	5 min.	5 min. [4]	T, M, R	No attenuation of oxidizer flow

[1] Measured average values of injector/fuel initial temperatures. "N.A." indicates that data has not been reported by ARC.

[2] Data available: T = temperature, M = movies, R = mass residuals.

[3] A total of 5 flow pulses were made, each separated by a 5 minute coast period

[4] A total of 11 flow pulses were made, each separated by a 5 minute coast period

TABLE A-2
BELL INJECTOR PHASE I TEST SUMMARY (CONTINUED)

A.2.3 Test Series Description (Continued)

Thermal conditioning of the injector was accomplished by spraying the injector with CO_2 after the altitude chamber had been evacuated. Additional thermal conditioning was accomplished, if required, by using trickle flows of Freon MF and methyl alcohol in the oxidizer and fuel ducts respectively.

Simulation of oxidizer cooling effects was attempted during 10 of the Phase I injector tests. The amount of Freon MF required to simulate N_2O_4 cooling effects was initially determined by comparing the heats of vaporization on a volumetric basis. The Freon MF heat of vaporization (63.5 cal/ml) is 45% of the nitrogen tetroxide heat of vaporization (142 cal/ml). Since the injector oxidizer manifold capacity is 475 ml (29 in^3), the volume of Freon required to simulate 475 ml of oxidizer is 1070 ml. The Freon was flowed into the injector over a period of 30 seconds to 2 minutes. This trickle flow of simulant resulted in significantly higher refrigeration effects on the fuel side of the injector than did actual pulse flows of nitrogen tetroxide. The last 3 of the tests using an oxidizer simulant were run with a reduced amount of simulant (535 ml) in order to reduce the refrigeration effect. A complete evaluation of the oxidizer simulants is presented in Section A.6 of this appendix.

A.3 BELL ASCENT ENGINE INJECTOR TESTS, PHASE II

A.3.1 Test Objectives

The objectives of the Phase II cold flow tests conducted with a Bell ascent injector were:

1. Determine pressure and temperature histories of the injector assembly and the retained propellants.
2. Determine fuel residuals as a function of coast time for single and dual pulse tests.
3. Evaluate the propellant residual distillation procedure for the Bell injector.

A.3.2 Test Article Configuration and Instrumentation

The test article consisted of an instrumented Bell ascent engine injector, propellant ducts and a quad-redundant propellant shut-off valve assembly. The injector valves were actuated by a solenoid-controlled nitrogen gas system, and were protected from the injector propellant spray by a splash plate. The injector and valve assembly was fastened to structural supports which were attached to the side walls of the ARC High Altitude Test Facility as in Phase I.

The orientation and instrumentation for the Phase II tests was identical, with one exception, to the Phase I orientation and instrumentation shown on Figures A-4 and A-5. The single exception was the substitution of a 0-5 psia transducer and gage protector for the 0-250 psia transducer used during the Phase I tests. Photographic coverage consisted of 1000 fps (frames per second) and 24 fps color films.

A.3.3 Test Series Description

A total of 64 injector cold flow tests were conducted during the period from April 15, 1968 through July 12, 1968. The basic characteristics of these tests are summarized in Table A-3. All tests used simultaneous pulsed flows of Aerozine-50 and an oxidizer simulant, Freon MF. Propellant supply system characteristics are described in Section A.2 and Figure A-1 of this appendix.

The principle difference between the Phase II and Phase I tests was the method of flowing the oxidizer simulant. During all Phase II tests, the oxidizer simulant (Freon MF) flowed through the injector valves, producing the same flow duration for both simulant and fuel. The typical flow pulse duration was 0.6 seconds. In contrast, the Phase I tests employed a trickle flow of simulant lasting for 30 to 120 seconds.

A.3.3 Test Series Description (Continued)

The effectiveness of using pulsed simulant flows was not evaluated by testing. However, an evaluation of simulant properties (Section A.6) indicates that pulsed flows of Freon MF provides a valid oxidizer simulation.

A significant number of the Phase II tests were unsuccessful at a nominal initial temperature of 40°F. The principle reason was failure to attain a uniform injector temperature of 40 ± 5°F prior to starting a test. These test failures resulted from difficulty in using trickle flows of Freon MF and isopropyl alcohol to attain uniform injector temperatures.

TEST NUMBER	FLUIDS	NOMINAL INITIAL TEMPERATURE (°F)	PROPELLANT PULSE DURATION (SECONDS)	FIRST COAST PERIOD (SEC)	SECOND COAST PERIOD (SEC)	DATA AVAILABLE 1	REMARKS
1	-	-	-	-	-	-	
2	-	-	-	-	-	-	
3	-	-	-	-	-	-	
4	-	-	-	-	-	-	
5	-	-	-	-	-	-	
6	-	-	-	-	-	-	
7	-	-	-	-	-	-	
8	A-50, Freon MF	40	0.55	46	none	P,T,R,M	89 cc residual
9	A-50, Freon MF	40	0.55	125	none	P,T,R,M	72 cc residual
10	A-50, Freon MF	40	0.55	624	none	P,T,R,M	36 cc residual
11	A-50, Freon MF	40	0.55	49	none	P,T,R,M	87 cc residual
12	-	-	-	-	-	-	
13	A-50, Freon MF	40	0.55	635	none	P,T,R,M	32 cc residual
14	-	-	-	-	-	-	
15	-	-	-	-	-	-	
16	-	-	-	-	-	-	
17	A-50, Freon MF	40	0.55	38.5	none	P,T,R,M	59 cc residual
18	-	-	-	-	-	-	
19	-	-	-	-	-	-	
20	A-50, Freon MF	40	0.55	616	none	P,T,R,M	43 cc residual
21	-	-	-	-	-	-	
22	-	-	-	-	-	-	
23	-	-	-	-	-	-	
24	-	-	-	-	-	-	

TABLE A-3
BELL INJECTOR PHASE II TEST SUMMARY

TEST NUMBER	FLUIDS	NOMINAL INITIAL TEMPERATURE (°F)	PROPELLANT PULSE DURATION (SECONDS)	FIRST COAST PERIOD (SEC)	SECOND COAST PERIOD (SEC)	DATA AVAILABLE 1	REMARKS
25	A-50, Freon MF	40	0.55	15.4	none	P,T,R,M	106 cc residual
26	-	-	-	-	-	-	
27	A-50, Freon MF	70	0.55	120	none	P,T,R,M	18 cc residual
28	A-50, Freon MF	70	0.55	120	none	P,T,R,M	14 cc residual
29	A-50, Freon MF	70	0.55	28.7	none	P,T,R,M	27 cc residual
30	A-50, Freon MF	70	0.55	33.9	none	P,T,R,M	33 cc residual
31	A-50, Freon MF	70	0.55	14.5	none	P,T,R,M	42 cc residual
32	A-50, Freon MF	70	0.55	16.7	none	P,T,R,M	40 cc residual
33	A-50, Freon MF	70	0.55	600	none	P,T,R,M	8 cc residual
34	A-50, Freon MF	70	0.55	600	none	P,T,R,M	8 cc residual
35	A-50, Freon MF	40	0.55	15	none	P,T,R,M	74 cc residual
36	A-50, Freon MF	40	0.55	1860	none	P,T,R,M	30 cc residual
37	-	-	-	-	-	-	
38	-	-	-	-	-	-	
39	A-50, Freon MF	40	0.5	299.4	294	P,T,R,M	
40	A-50, Freon MF	40	0.5	300	241	P,T,R,M	71 cc residual

TABLE A-3
BELL INJECTOR PHASE II TEST SUMMARY (CONTINUED)

TEST NUMBER	FLUIDS	NOMINAL INITIAL TEMPERATURE (°F)	PROPELLANT PULSE DURATION (SECONDS)	FIRST COAST PERIOD (SEC)	SECOND COAST PERIOD (SEC)	DATA AVAILABLE 1	REMARKS
41	-	-	-	-	-	-	
42	A-50, Freon MF	40	0.5	300	1790	P,T,R,M	16 cc residual
43	A-50, Freon MF	40	0.5	340.7	1830	P,T,R,M	47 cc residual
44	A-50, Freon MF	40	0.5	300	300	P,T,R,M	66 cc residual
45	A-50, Freon MF	40	0.5	7200	none	P,T,R,M	7 cc residual
46	A-50, Freon MF	40	0.5	309.4	7146	P,T,R,M	19 cc residual
47	-	-	-	-	-	-	
48	-	-	-	-	-	-	
49	-	-	-	-	-	-	
50	-	-	-	-	-	-	
51	A-50, Freon MF	40	0.5	301	130	P,T,R,M	80.1 cc residual
52	A-50, Freon MF	40	0.5	305	7200	P,T,R,M	14.9 cc residual
53	-	-	-	-	-	-	
54	-	-	-	-	-	-	
55	A-50, Freon MF	40	0.5	307	303	P,T,R,M	73.75 cc residual
56	-	-	-	-	-	-	
57	A-50, Freon MF	70	0.5	304	300	P,T,R,M	24.9 cc residual
58	A-50, Freon MF	70	0.5	303	305	P,T,R,M	2.4 cc residual

TABLE A-3
BELL INJECTOR PHASE II TEST SUMMARY (CONTINUED)

TEST NUMBER	FLUIDS	NOMINAL INITIAL TEMPERATURE (°F)	PROPELLANT PULSE DURATION (SECONDS)	FIRST COAST PERIOD (SEC)	SECOND COAST PERIOD (SEC)	DATA AVAILABLE ◁	REMARKS
59	A-50, Freon MF	70	0.5	302	2394	P,T,R,M	17.9 cc residual
60	A-50, Freon MF	70	0.5	306	529	P,T,R,M	29.5 cc residual
61	A-50, Freon MF	70	0.5	302	4048	P,T,R,M	14.6 cc residual
62	A-50, Freon MF	70	0.5	306	7080	P,T,R,M	4.2 cc residual
63	A-50, Freon MF	70	0.5	302	41	P,T,R,M	42.5 cc residual
64	A-50, Freon MF	70	0.5	303	36	P,T,R,M	41.6 cc residual

◁ Data Available: P = Pressure, T = Temperature, R = Mass residuals, M = Movies

TABLE A-3
BELL INJECTOR PHASE II TEST SUMMARY (CONTINUED)

A.4 TRW DESCENT ENGINE INJECTOR TESTS

A.4.1 Test Objectives

The objectives of the cold flow tests conducted with a TRW injector were:

1. Determine pressure and temperature histories of the injector assembly and the retained propellants.

2. Determine fuel residuals as a function of coast time for single and dual pulse tests.

3. Evaluate the distillation procedure for the TRW injector.

4. Evaluate the oxidizer simulant (Freon TF) used during fuel side tests.

5. Obtain photographic records of the phenomena occurring during propellant flow and coast periods.

A.4.2 Test Article Configuration and Instrumentation

The test article consisted of an instrumented descent engine injector, injector manifolds and quad-redundant propellant shut-off valve assembly. The variable-area cavitating venturis were replaced with straight segments of tubing. Significant modifications incorporated into the ARC test article are listed below:

1. A fixed thrust assembly to provide accurate adjustment of the injector gap for 10% thrust setting.

2. Fixed-area cavitating venturis (changeable inserts) to provide proper propellant flow rates.

3. An injector mounting ring to serve as a base for the fixed thrust assembly, a support for the injector dome cover, and as a mount for the assembly in Atlantic Research Corporation's test setup.

4. An injector dome cover to be used in vacuum distilling residual propellant from the manifolds.

5. Provision for extensive instrumentation to acquire the necessary test data.

The test injector was mounted in the ARC High Altitude Test Facility by fastening the injector mounting ring to a specially designed support structure. The injector face was tipped down so that the thrust vector axis (+X) was oriented 8° above the horizontal. The thrust

A.4.2 Test Article Configuration and Instrumentation (Continued)

assembly and fixed-area cavitating venturis were adjusted to provide a 10% thrust setting. Propellant supply system characteristics are described in Section A.2 and are shown in Figure A-2 of this appendix.

The location of pressure and temperature transducers is shown on Figure A-6. Copper-constantan thermocouples were used for all temperature measurements. Temperature measurements identified as TC-113, -114, -115, and -116 were immersion thermocouples supplied by ARC. Thermocouples TC-1 through TC-12 were supplied and mounted by TRW. TC-1, -2, -6, -8 were cemented; all others were spot welded. Available information on pressure transducers is given in the table below.

TRANSDUCER CHARACTERISTICS

TEST NUMBERS	P_1 FUEL DUCT	P_2 FUEL MANIFOLD	P_3 FUEL MANIFOLD	P_4 OXID DUCT	P_5 OXID INJECTOR
1-3	0-350 psia	none	0-5 psia	none	none
3-22	0-15 psia	0-5 psia	none	none	none
23-26	none	none	none	0-15 psia	0-5 psia

The low range pressure transducers (0-5 psia and 0-15 psia) were equipped with a protective device which prevented them from being damaged by exposure to the normal manifold pressures experienced during propellant flow pulses.

A.4.3 Test Series Description

A total of 26 injector cold flow tests were conducted during the period from August 26 through September 27, 1968 and December 8, 1968 through March 8, 1969. The descent injector tests were temporarily suspended in October in order to accomplish higher priority tests on the Rocketdyne ascent injector. The basic characteristics of these tests are summarized in Table A-4. No beaker tests were conducted during the TRW injector test series.

The TRW descent injector was cooled to 40°F, when required, by spraying isopropyl alcohol from two circular manifolds onto the valve side of the injector. Two heat lamps were aimed at the lower area of the injector manifolds to compensate for the overcooling which occurred when alcohol drained down and collected near the bottom of the

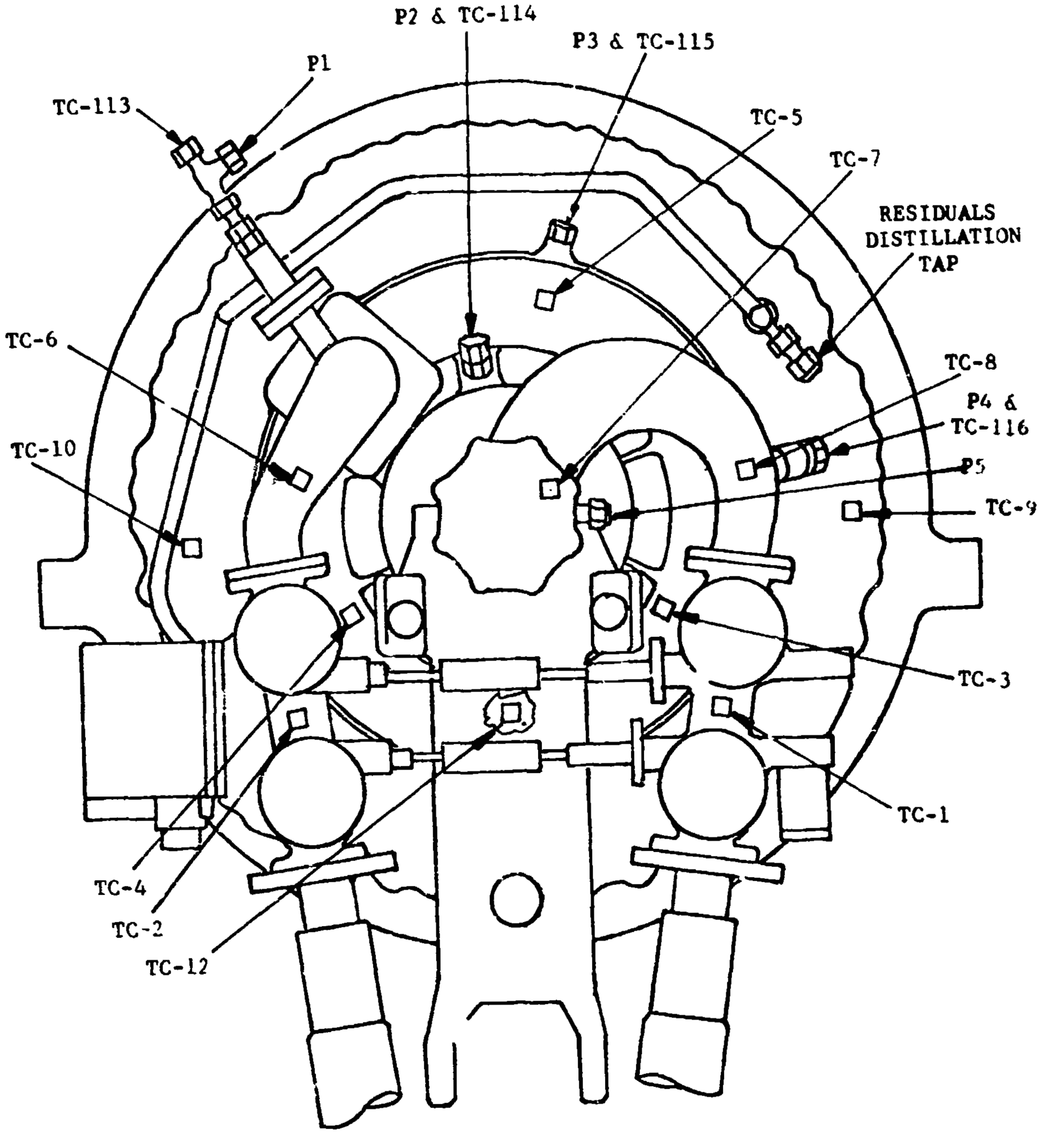


FIGURE A-6 INSTRUMENTATION LOCATIONS ON TRW INJECTOR

TEST NUMBER	FLUIDS	NOMINAL INITIAL TEMPERATURE (°F)	PROPELLANT PULSE DURATION (SECONDS)	FIRST COAST PERIOD (SEC)	SECOND COAST PERIOD (SEC)	DATA AVAILABLE (1)	REMARKS
1	A-50, Freon TF					None	Test unsuccessful
2	A-50, Freon TF	70	3.0	1800	none	P,T,R,M	7 cc residual
3	A-50, Freon TF	70	3.0	1799	none	P,T,R,M	9 cc residual
4	A-50, Freon TF	70	3.5	1800	none	P,T,R,M	10 cc residual
5	A-50, Freon TF	70	3.5	297	none	P,T,R,M	21 cc residual
6	A-50, Freon TF	70	3.5	297	none	P,T,R,M	21 cc residual
7	A-50, Freon TF	70	3.5	137	none	P,T,R,M	34 cc residual
8	A-50, Freon TF	70	3.5	135	none	P,T,R,M	37 cc residual
9	A-50, Freon TF	70	3.5	137	none	P,T,R,M	37 cc residual
10	A-50, Freon TF	40	3.5	1800	none	P,T,R,M	45 cc residual
11	A-50, Freon TF	70	3.5	297	288	P,T,M	-
12	A-50, Freon TF	70	3.5	288	1800	P,T,R,M	36 cc residual
13	A-50, Freon TF	70	3.5	292	125	P,T,R,M	85 cc residual
14	A-50, Freon TF	40	3.7	300.3	none	P,T,R,M	114.35 cc residual
15	A-50, Freon TF	40	3.6	604.6	none	P,T,R,M	89-1/4 cc residual

TABLE A-4
TRW INJECTOR TEST SUMMARY

TEST NUMBER	FLUIDS	NOMINAL INITIAL TEMPERATURE (°F)	PROPELLANT PULSE DURATION (SECONDS)	FIRST COAST PERIOD (SEC)	SECOND COAST PERIOD (SEC)	DATA AVAILABLE [1]	REMARKS
16	A-50, Freon TF	40	3.6	1220.8	none	P,T,R,M	98.3 cc residual
17	A-50, Freon TF	40	3.6	1824.4	none	P,T,R,M	66.4 cc residual
18	A-50, Freon TF	40	3.6	3596.4	none	P,T,R,M	54.1 cc residual
19	A-50, Freon TF	40	3.6	7196.4	none	P,T,R,M	34.9 cc residual
20	A-50, Freon TF	40	3.6	1218.2	none	P,T,R,M	86.7 cc residual
21	A-50, Freon TF	40	3.6	212.4	none	P,T,R,M	143.5 cc residual
22	-	-	-	-	-	none	No data on this test
23	Freon TF	40	3.6	3605	none	P,T	Simulant evaluation
24	Freon TF	40	3.6	3600	none	P,T	Simulant evaluation
25	N_2O_4	40	3.6	3602	none	P,T	Simulant evaluation
26	N_2O_4	40	3.6	3602	none	P,T	Simulant evaluation

[1] Data Available: P = Pressure, T = Temperature, R = Mass Residuals, M = Movies

TABLE A-4
TRW INJECTOR TEST SUMMARY (CONTINUED)

A.4.3 Test Series Description (Continued)

injector plate.

An oxidizer simulant, Freon TF, was used during all fuel flow tests. The simulant was flowed through the injector oxidizer valves, producing the same flow duration for both the simulant and the A-50 fuel. Two simulant flows and two oxidizer flows were conducted to determine the injector thermal response to oxidizer flows and to evaluate the effectiveness of the oxidizer simulant. These tests are evaluated in Section A.6 of this appendix.

A.5 ROCKETDYNE ASCENT ENGINE INJECTOR TESTS

A.5.1 Test Objectives

The objectives of the cold flow tests conducted with a Rocketdyne ascent injector were:

1. Determine pressure and temperature histories of the injector assembly and the retained propellants.
2. Determine fuel residuals as a function of coast time for single pulse tests.
3. Obtain photographic records of the phenomena occurring during propellant flow and coast periods.

The objectives of the cold flow tests conducted with a beaker simulating the Rocketdyne ascent injector were:

1. Evaluate the effects of orientation on propellant expulsion and freezing phenomena.
2. Evaluate the effects of injector filters on propellant expulsion and freezing phenomena.
3. Obtain photographic records of the propellant expulsion and freezing phenomena.

A.5.2 Test Article Configuration and Instrumentation

The test article consisted of an instrumented Rocketdyne ascent engine injector, injector manifolds, and quad-redundant propellant ball valve assembly. The injector contained LM-3 configuration ("old") injector filters. The test article was modified to provide the required instrumentation ports and tap for the propellant distillation apparatus. The injector ball valves were actuated by gaseous nitrogen, and were protected from the injector propellant spray by a splash plate.

The test injector was mounted in the ARC High Altitude Test Facility by fastening the injector flange to a specially designed support structure. The injector face was tipped down so that the thrust vector axis (+X) was oriented 8° above the horizontal.

The location of pressure and temperature transducers is shown on Figure A-7. Copper-constantan thermocouples were used for all temperature measurements. A low range (0-5 psia) transducer was erroneously mounted on a chamber pressure tap instead of an injector fuel manifold tap. A high range (0-250 psia) transducer was mounted on the injector fuel manifold tap labelled P-2 on Figure A-7. Since the high range transducer did not produce accurate pressure data in the range of

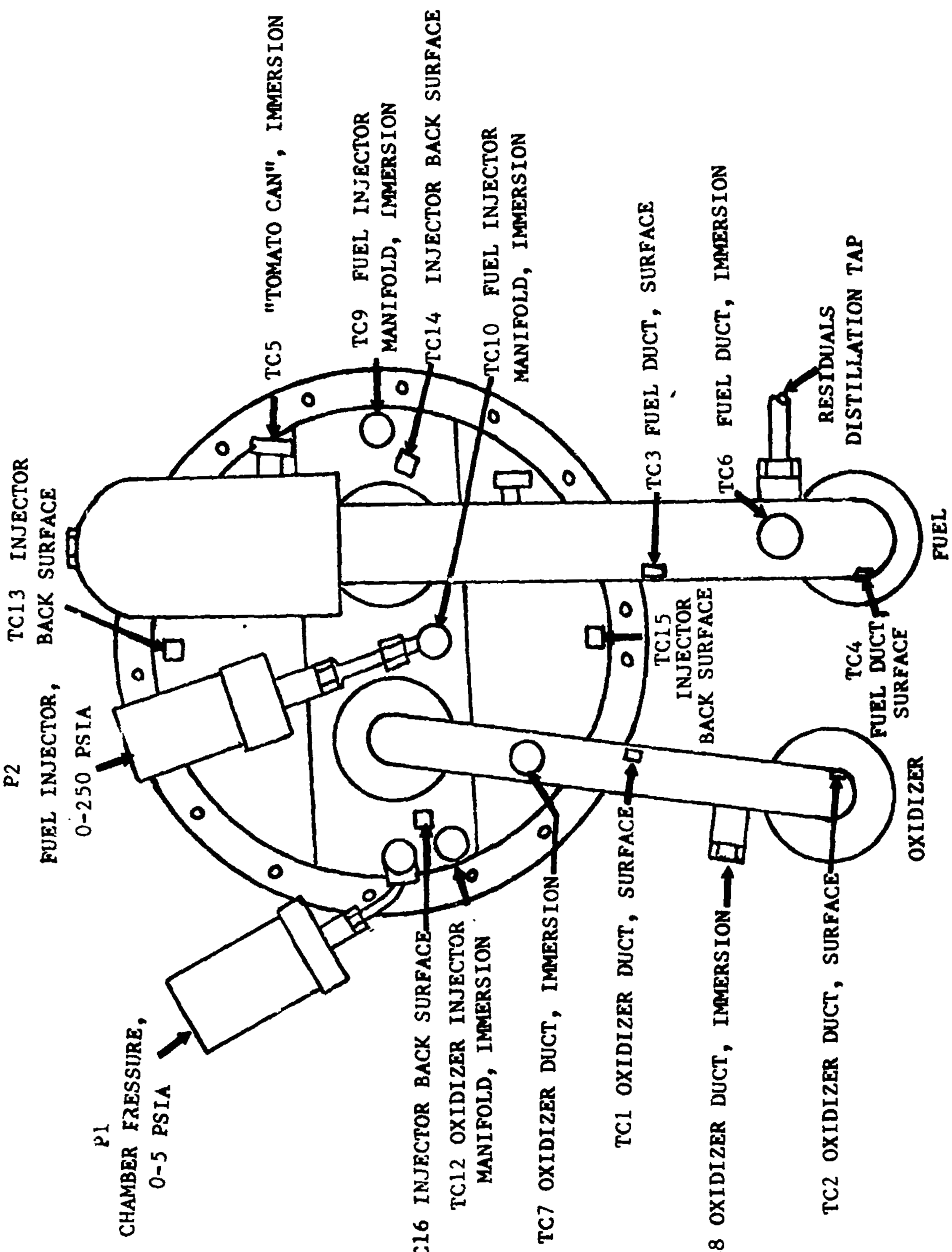


FIGURE A-7 INSTRUMENTATION LOCATIONS ON ROCKETDYNE INJECTOR

A.5.2 Test Article Configuration and Instrumentation (Continued)

interest (less than 10 psia), the high range transducer data were not reported. Therefore, there was no usable fuel manifold pressure data from the ARC Rocketdyne injector tests.

The beaker used during the Rocketdyne injector test series was a modification of the beaker used during the Phase I Bell injector test series. The modification consisted of machining a hole in the middle aluminum spacer so that a Rocketdyne injector filter could be installed. The beaker had an internal volume of approximately 35 in^3 and an orifice area of approximately 0.24 in^2. In comparison, the Rocketdyne injector had a fuel side volume of 62.7 in^3 and a fuel orifice area of 0.203 in^2.

The beaker configuration is shown in Figure A-8. The beaker was constructed from 4 inch diameter pyrex glass sections with aluminum end caps and filter holder. Holes were drilled in the top end cap to simulate the area of the ascent injector fuel orifices. Beaker instrumentation was limited to four copper-constantan thermocouples. Locations for these thermocouples are shown on the schematic drawing of the beaker, Figure A-8.

A.5.3 Test Series Description

A total of 17 ascent injector cold flow tests were conducted during the period from October 14, 1968 through October 22, 1968. The basic characteristics of these tests are summarized in Table A-5.

A total of 20 beaker tests were conducted during the period from September 30 through October 12, 1968 and November 10-23, 1968. The characteristics of these tests are summarized in Table A-6. High-speed (1000 fps) and normal speed (24 fps) motion pictures were made for each test. The high speed films covered the first 4 seconds of each test, while the normal speed films covered the entire test period. The beaker tests were run according to the following general outline:

1. Condition the beaker and test fluid to the desired temperature.
2. Fill the evacuated beaker with test fluid.
3. Evacuate the test chamber, and remove the beaker cap by actuating a pneumatic cylinder.
4. Allow the beaker to remain uncapped until all residual fuel is frozen.
5. Recap the beaker, allow the residuals to melt, and measure the height of liquid in the beaker.

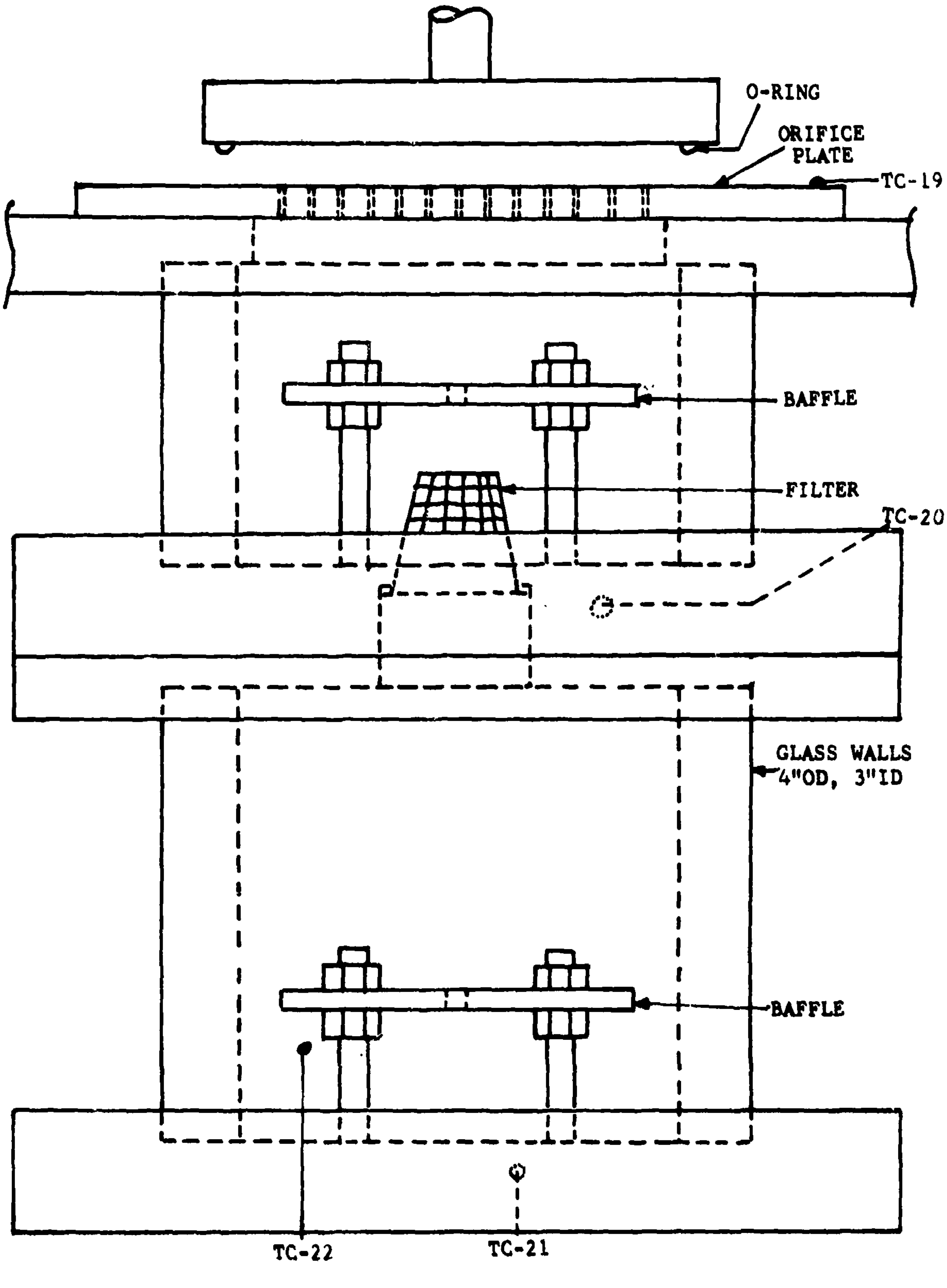


FIGURE A-8 BEAKER CONFIGURATION AND INSTRUMENTATION

TEST NUMBER	FLUIDS	NOMINAL INITIAL TEMPERATURE (°F)	PROPELLANT PULSE DURATION (SECONDS)	FIRST COAST PERIOD (SEC)	SECOND COAST PERIOD (SEC)	DATA AVAILABLE [1]	REMARKS
1	A-50, Freon MF	40	0.7	7197	none	T,R,M	0 cc residual
2	A-50, Freon MF	40	0.7	3600	none	T,R,M	13 cc residual
3	A-50, Freon MF	40	0.7	1797	none	T,R,M	15 cc residual
4	-	-	-	-	-	-	No data on this test
5	A-50, Freon MF	40	0.56	1200	none	T,R,M	20 cc residual
6	A-50, Freon MF	40	0.56	601	none	T,R,M	31 cc residual
7	A-50, Freon MF	40	0.56	120	none	T,R,M	33 cc residual
8	A-50, Freon MF	40	0.55	61	none	T,R,M	35 cc residual
9	A-50, Freon MF	40	0.55	60	none	T,R,M	29 cc residual
10	A-50, Freon MF	40	0.56	1800	none	T,R,M	16 cc residual
11	-	-	-	-	-	-	No data on this test
12	A-50, Freon MF	70	0.55	60	none	T,R,M	20 cc residual
13	A-50, Freon MF	70	0.55	123	none	T,R,M	12 cc residual
14	-	-	-	-	-	-	No data on this test
15	A-50, Freon MF	70	0.56	603	none	T,R,M	3 cc residual
16	A-50, Freon MF	70	0.56	64	none	T,R,M	16 cc residual
17	A-50, Freon MF	70	0.56	120	none	T,R,M	11 cc residual

[1] Data Available: P = Pressure, R = Mass Residuals, M = Movies

TABLE A-5
ROCKETDYNE INJECTOR TEST SUMMARY

TEST NUMBER	FLUID	NOMINAL INITIAL TEMPERATURE (°F)	FILTER [1]	BEAKER ORIENTATION	DATA AVAILABLE [2]	REMARKS
1	A-50	70	-	-	T,M	
2	A-50	70	old	up	T,R,M	40 cc residual @ 10 min.
3	A-50	70	-	-	T,M	
4	A-50	70	new	up	T,R,M	20 cc residual @ 10 min.
5	A-50	70	new	horizontal	T,R,M	12 cc residual @ 10 min.
6	A-50	70	old	horizontal	T,R,M	12 cc residual
7	A-50	70	none	horizontal	T,R,M	2 cc residual @ 840 sec.
8	A-50	40	none	up	T,R,M	172 cc residual @ 900 sec.
9	A-50	40	old	up	T,R,M	186 cc residual @ 660 sec.
10	A-50	40	new	up	T,R,M	424 cc residual @ 660 sec.
11	A-50	30	new	up	T,M	Beaker inadvertently recapped
12	A-50	30	new	up	T,R,M	424 cc residual @ 2400 sec.
13	A-50	30	none	up	T,R,M	411 cc residual @ 1400 sec.
14	A-50	30	none	down	T,R,M	No residual @ 300 sec.
15	A-50	30	new	down	T,R,M	No residual @ 400 sec.
16	A-50	30	new	horizontal	T,R,M	79 cc residual @ 800 sec.
17	A-50	30	none	horizontal	T,R,M	12 cc residual @ 370 sec.
18	N_2O_4	30	new	horizontal	T,R,M	48 cc residual @ 220 sec.
19	N_2O_4	30	new	down	T,R,M	No residual @ 83 sec.
20	N_2O_4	30	new	up	T,R,M	188 cc residual @ 980 sec.

[1] "Old" Filter - LM-3 configuration; "New" Filter - LM-5 configuration
[2] Data Available: T = Temperature, R = Mass residuals, M = Movies

TABLE A-6
ROCKETDYNE INJECTOR, BEAKER SIMULATION TEST SUMMARY

A.6 EVALUATION OF INJECTOR COLD FLOW TESTING

A.6.1 Limitations of Cold Flow Testing

This section evaluates specific cold flow test factors which can limit the application of cold flow test results to the prediction and evaluation of hot firing test results. The specific test factors are listed below. One significant factor, the absence of combustion heat input, is omitted from this section, but is discussed in Appendix E.

1. Cold flow tests were run without a thrust chamber.

2. Cold flow tests used oxidizer simulants (Freon MF or TF) in place of the oxidizer during all fuel flows.

3. Injector valves were actuated by a GN_2 system during cold flow tests. Hot-firing tests used a fuel actuation system. Total actuation times approximated those of the fuel actuated system.

The cold flow test data can be significantly affected by the level of the "chamber pressure" existing at the injector face during, and after, each propellant flow pulse. The "chamber pressure" measured during cold flow tests was actually the altitude chamber pressure, which was significantly lower than the chamber pressure that would result from propellant combustion in a thrust chamber. During the propellant flow pulse, and immediately after termination of the flow pulse, the liquid flow rate out of the injector will be increased by any reduction in the "chamber pressure" existing at the injector face. When the flow through the injector orifices becomes two phase (liquid plus gas) or single phase (gas only), the chamber pressure level may affect the flow rate out of the injector. At the present time, the effect of chamber pressure on the gas and two phase flow rates through the injector "orifices" is not known. Analysis of the flow phenomena is complicated by the fact that the injector "orifices" are actually cylindrical holes having an L/D of approximately 2.7 to 3.8. As a result, the injector orifice flow may not "choke" in the same way that flow through the thrust chamber throat becomes "choked". If the injector orifices do not choke, the injector flow rate will always be a function of the "chamber pressure", and the comparison of cold flow tests to hot firing tests will become difficult and complex. Because of the importance of injector flow characteristics in the cold flow to hot firing correlation, the injector flow characteristics require further investigation.

A.6.1 Limitations of Cold Flow Testing (Continued)

The engine ball valves were actuated by a GN_2 system during the ARC cold flow tests, instead of the normal fuel actuation system used during hot firing tests. This difference had no significant effect on the cold flow tests because a cold flow test is valid if the injector and manifolds are filled with propellants during the flow pulse. Although valve actuation times were not reported by ARC, it is concluded that the actuation times were adequate to fill the injector and manifolds during the commanded pulse durations. This conclusion is based on approximate actuation times reported during facility checkout prior to the TRW tests, and examination of test data which show that the temperatures of both the top and bottom of the injectors responded to propellant flow pulses.

An oxidizer simulant, Freon TF or Freon MF, was used during all fuel flow tests on the Rocketdyne injector, the TRW injector and the Phase II Bell injector test series, and on some of the Phase I Bell injector test series. The purpose of using the oxidizer simulants was to produce an accurate thermal simulation of the oxidizer cooling effects without producing combustion. The effectiveness of the oxidizer simulants was evaluated by comparing the injector thermal response to simulant flows and to oxidizer (nitrogen tetroxide) flows.

Simulant evaluation tests were conducted at the end of the Phase I Bell injector tests and at the end of the TRW injector tests. The effect of the simulant pulse flows used during the Rocketdyne injector tests and Phase II Bell injector tests was not evaluated by testing. As a result, the evaluation of simulant effectiveness for these tests must be based on a comparison between simulant and oxidizer properties.

The simulant used during Bell and Rocketdyne ascent injector tests was Freon MF. Because of material compatibility problems in the TRW descent injector, the oxidizer simulant was changed to Freon TF for the descent injector tests. The tables below list the vapor pressure, density, boiling and freezing points and heat of vaporization for N_2O_4, Freon TF (CCl_2F - CCl_4F_2) and Freon MF (CCl_3F).

Property	N_2O_4	Freon TF	Freon MF
Boiling Point	70°F @ 1 atm.	118°F @ 1 atm.	75°F @ 1 atm.
Freezing Point	12°F @ 1 atm.	-31°F @ 1 atm.	-168°F @ 1 atm.
Heat of Vap. at Boiling Point	178 BTU/lb	63.1 BTU/lb	78.3 BTU/lb

A.6.1 Limitations of Cold Flow Testing (Continued)

Temp. °F	N_2O_4 Density lb/ft^3	N_2O_4 Vapor Pressure psia	Freon TF Density lb/ft^3	Freon TF Vapor Pressure psia	Freon MF Density lb/ft^3	Freon MF Vapor Pressure psia
10	94.5	2.5	[illegible]	1.2	97	3.2
20	93.9	3.5	102	1.6	96.5	4.2
30	93.2	4.8	101	2.0	96	5.5
40	92.5	6.5	100	2.7	95	7.2
50	91.7	8.6	99.5	3.3	94	8.9
60	90.9	12.2	99	4.3	93.5	11
70	90.2	14.[illegible]	98.5	5.4	92 5	13

The tables above show that, except for the lower freezing point and lower heat of vaporization, Freon MF is a good N_2O_4 simulant. Freon TF, used during the descent injection tests, is a poorer simulant than Freon MF because the vapor pressure and heat of vaporization are significantly lower than either Freon MF or N_2O_4.

The evaluation of oxidizer simulation during Phase I Bell injector tests was conducted by comparing test results from trickle flows of simulant (Freon MF) to test results from pulsed flows of oxidizer (N_2O_4). The volumes of Freon used (535 ml and 1070 ml) during the evaluation tests representative of the amounts used during the Phase I fuel flow tests. The time required to flow these amounts of Freon, at a supply pressure of 15 psia, was approximately 60 seconds and 120 seconds respectively. The duration of the N_2O_4 flow pulse was 0.6 seconds at a supply pressure of 160 psia.

The thermal response of the Bell injector to a trickle flow of 535 ml of Freon MF and a 0.6 second flow pulse of N_2O_4 is shown on Figure A-9. Following the N_2O_4 flow, the immersion thermocouple in the injector oxidizer flow passage (TC 11) responded rapidly, dropping to 7°F in 7.5 seconds. The injector (TC 8) responded slowly, with the injector temperature still decreasing slowly at the termination of the test (300 seconds). Although the minimum oxidizer flow passage temperature was 7°F, the injector was cooled to a minimum of 32°F, a 10°F decrease in temperature. In contrast, the trickle flow of Freon MF caused a slow temperature decrease to 20°F in the oxidizer injector flow passage. The injector was cooled significantly during the Freon flow, reaching 28°F at 60 seconds. The minimum oxidizer flow passage temperature (19°F) and the minimum injector temperature (24°F) were reached at approximately 120 seconds. Comparing the temperature data show that the simulant produced a significantly greater temperature decrease (19-22°F) than did the oxidizer (9-10°F).

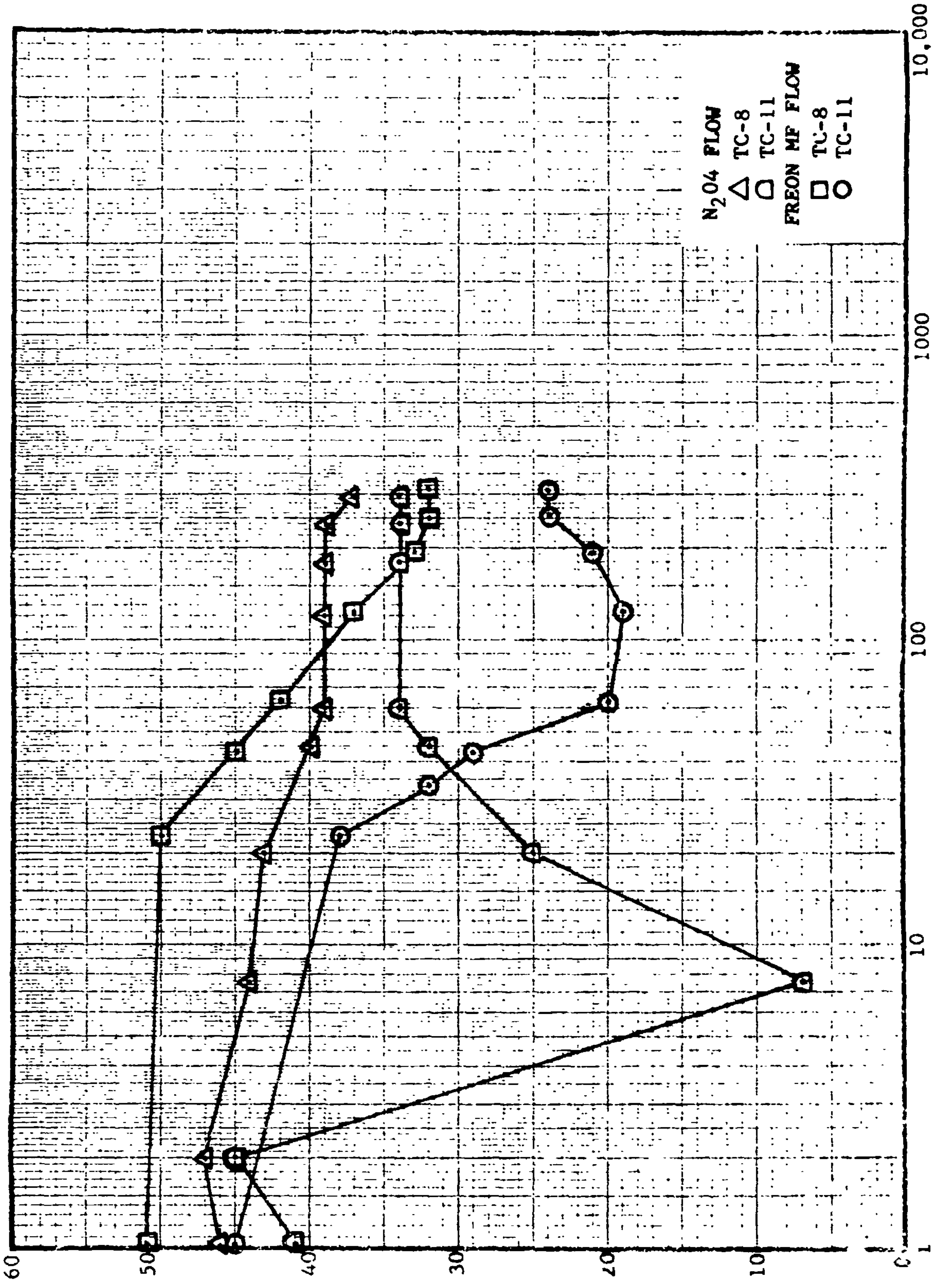


FIGURE A-9 TEMPERATURE HISTORIES FOR BELL INJECTOR OXIDIZER MANIFOLD

A.6.1 Limitations of Cold Flow Testing (Continued)

Additional data on the effect of the Phase I oxidizer simulation is shown on Figure A-10. This figure presents temperature data from Test #9, run without simulant flow, and Test #18, run with a trickle flow of 535 ml of Freon MF during each of the first four fuel pulses. The thermocouple, TC 12, was located on the external surface of the injector fuel manifold, at the bottom of the injector. The data show that the trickle flow of simulant had a significant effect on injector temperature. The injector temperature for Test #18, run with simulant, was from 5^{o} to $20^{o}F$ colder than the injector temperature measured during Test #9. The refrigerating effect of the simulant trickle flow could have significantly increased the amount of frozen fuel residuals, and caused premature injector blockage.

An evaluation of the oxidizer simulant (Freon TF) used during the descent injector tests was conducted during the last 4 tests of the test series. Descent injector tests 23 and 24 were run with 3.6 second pulses of Freon TF, and tests 25 and 26 were run with 3.6 second pulses of N_2O_4.

The thermal reponse of the TRW descent injector to pulsed simulant and oxidizer flows is shown on Figures A-11 and A-12. Figure A-11 shows the response of a thermocouple (TC 8) located on the oxidizer duct in a region which retains fluid after the flow pulse is completed. These data show that the Freon TF did not cool the oxidizer manifold as rapidly nor to as low a temperature as the N_2O_4. From 6 to about 60 seconds, the Freon was approximately $10^{o}F$ warmer than the N_2O_4 and from 60 to 1000 seconds, the temperature difference increased to about $20°F$. From 1000 to 3600 seconds, the Freon and N_2O_4 temperatures converge, until at 3600 seconds, they are again nearly the same. Figure A-12 shows the response of a thermocouple (TC 12) located on the bottom of the injector fuel manifold. The temperature response to simulant and oxidizer flows differed by a maximum of $7^{o}F$.

The similarity of the LMDE fuel manifold thermal responses is a result of the LMDE oxidizer flow path. Oxidizer and simulant flow in the LMDE injector is confined to the center of the injector. This limits the heat transfer between the oxidizer flow path and the large injector fuel manifold located on the periphery of the injector. As a result, the effectiveness of an oxidizer simulant has a relatively small effect on the fuel-side characteristics of the LMDE injector.

The effect of the simulant pulse flows used during the Phase II Bell injector tests and the Rocketdyne injector tests was not evaluated by testing. A comparison between the properties of Freon MF and N_2O_4 shows that the densities and vapor pressures are comparable, but the Freon MF heat of vaporization is only 44% of the N_2O_4 heat of vaporization. The low Freon heat of vaporization did not significantly

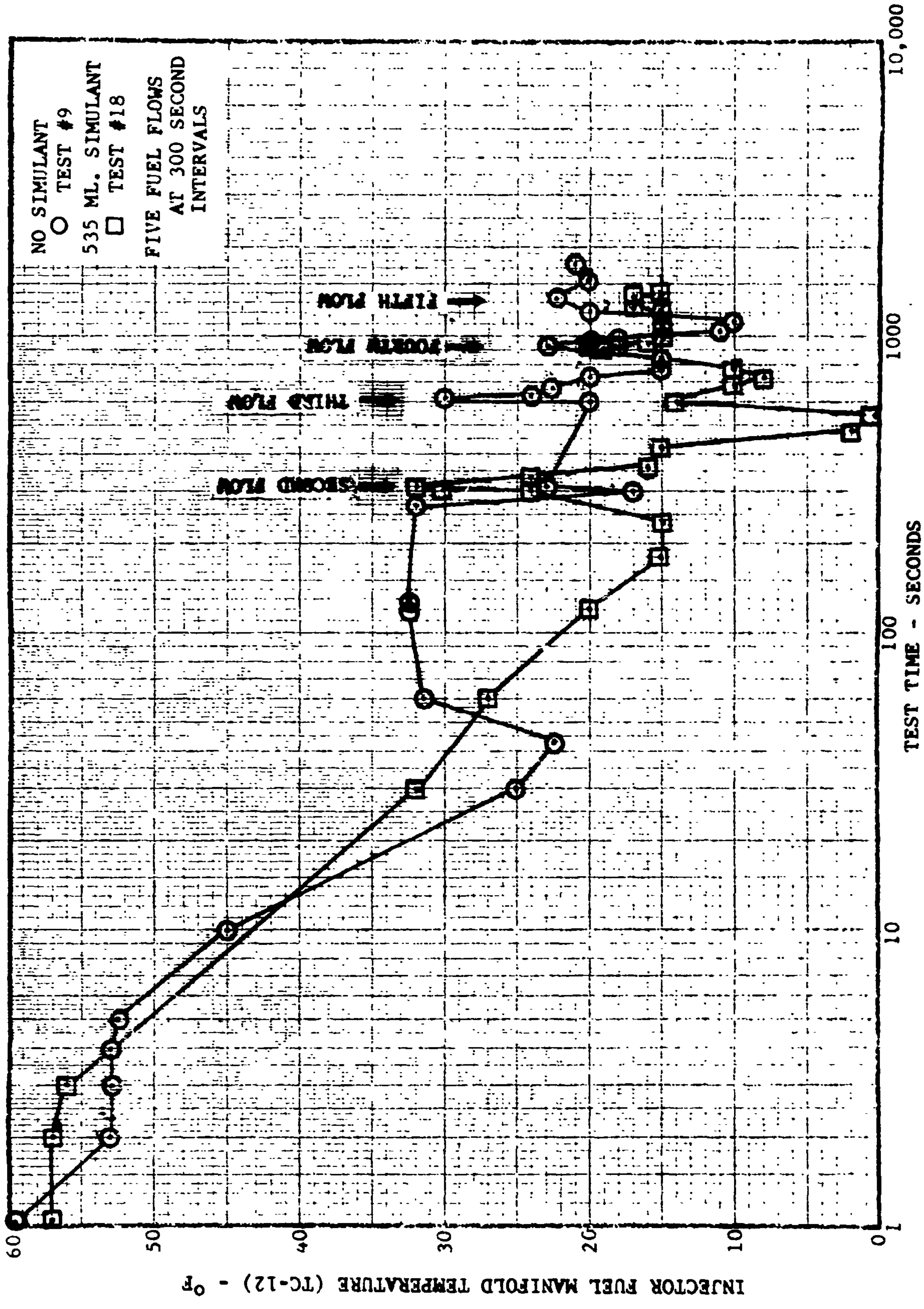


FIGURE A-10 TEMPERATURE HISTORY OF BELL INJECTOR FUEL MANIFOLD

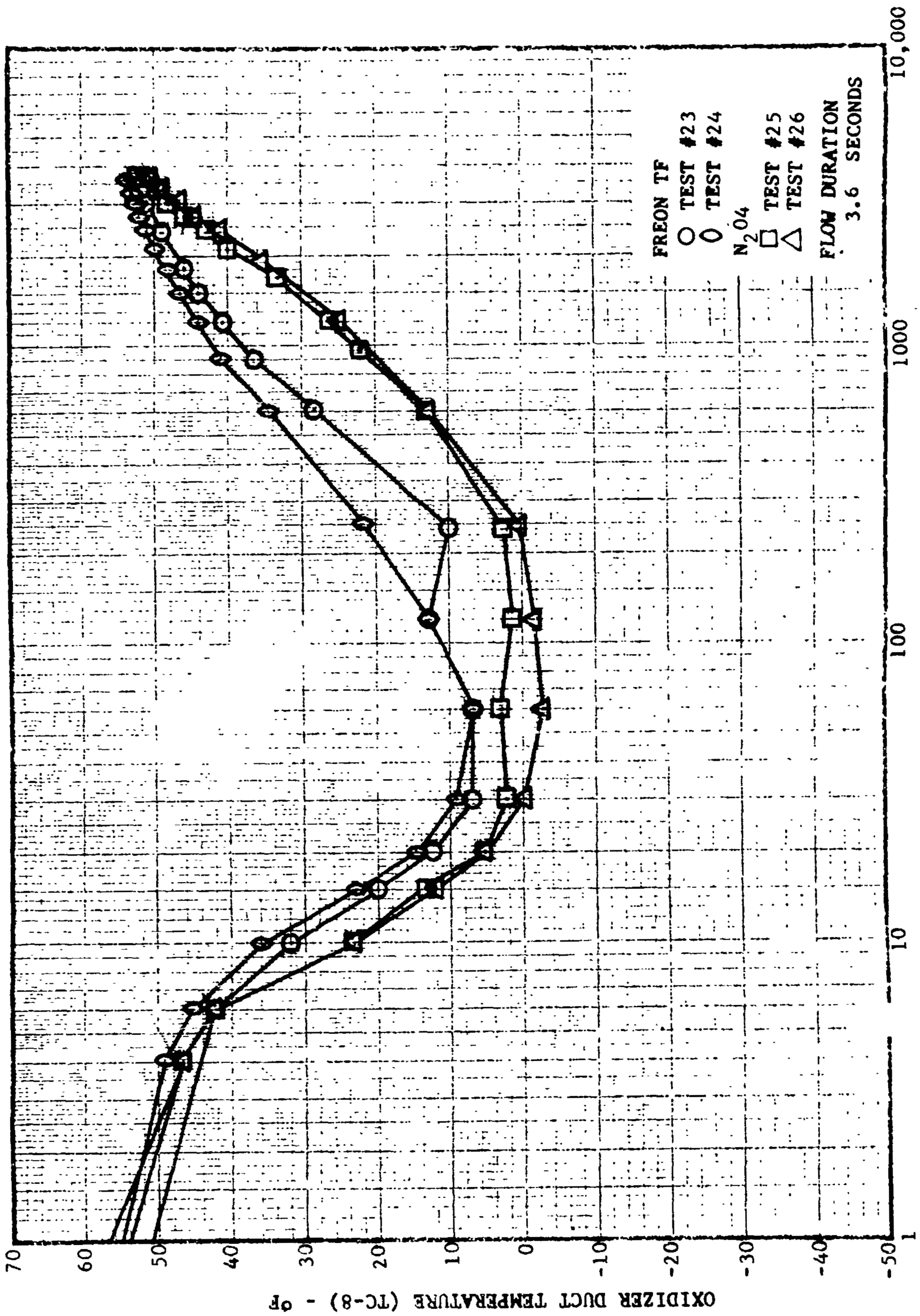
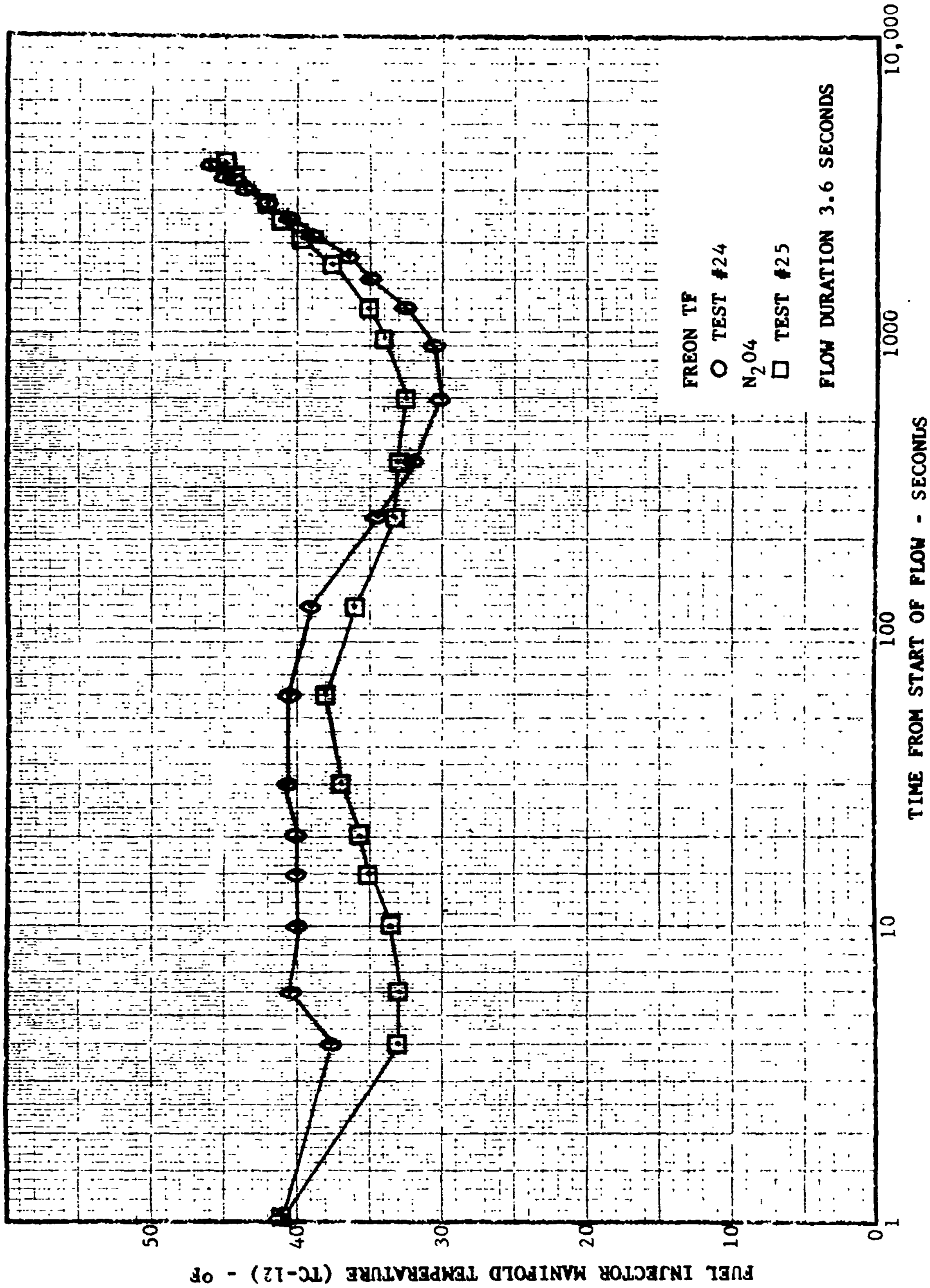


FIGURE A-11 TEMPERATURE HISTORY OF TRW INJECTOR OXIDIZER DUCT



F[illegible] [illegible]PERATURE HISTORY OF TRW INJECTOR FUEL MANIFOLD

A.6.1 Limitations of Cold Flow Testing (Continued)

affect the temperature of the injector because a major part of the oxidizer residuals were trapped in the duct between the injector valves and the injector. Within a few seconds after flow termination, the injector was essentially free from oxidizer or simulant, and the injector passages were primarily cooled by vapor produced in the propellant ducts. As a result, the reduced cooling capacity of the simulant primarily affected the temperature of the oxidizer duct. The relatively large mass of the injector valve assembly (27 lbs) decreased the effect of oxidizer/simulant temperature differences on fuel residuals adjacent to the valve assembly, and the relatively thin duct walls (0.035 inches) thermally isolated the oxidizer/simulant liquid residuals from the injector. It should be noted that the descent injector oxidizer ducts were 10-20^{o}F warmer when simulant (Freon TF) was used, but the injector fuel manifolds were a maximum of only 7^{o}F warmer. As a result of these considerations, it is concluded that Freon MF produced a valid simulation of the oxidizer cooling effects on the injector and fuel ducts.

A.6.2 Adequacy of Cold Flow Testing

The ARC testing was evaluated to determine if the tests were adequate to identify potential problem areas and to permit a thorough engineering evaluation. This evaluation was based on the test matrices, test instrumentation and test facilities. The test matrices and test instrumentation were specified by GAC and the test facility was operated by ARC to meet the required test conditions.

The summary test matrix below shows the basic types of tests conducted on each injector.

Test	Injector Test Series			
	Bell Phase I	Bell Phase II	TRW	Rocketdyne
Fuel Flow	Yes	Yes	Yes	Yes
Oxid. Flow	Yes	No	Yes	No
Distillation Evaluation	No	Yes	Yes	No
Simulant Evaluation	Yes	No	Yes	No

The Phase I tests conducted on the Bell ascent injector were adequate to determine the basic phenomena which occur after engine shutdown in a vacuum. The omission of oxidizer flow tests and simulant evaluation tests from the Bell Phase II test series reduced the effectiveness of the engineering analysis. The effectiveness of the Rocketdyne tests was significantly reduced by the omission of oxidizer, distillation

A.6.2 Adequacy of Cold Flow Testing (Continued)

evaluation, and simulant evaluation tests. The TRW descent injector test series provided an adequate basis for engineering analysis.

The summary instrumentation matrix below shows the basic types of instrumentation for which data were reported during each test series.

Instrumentation	Injector Test Series			
	Bell Phase I	Bell Phase II	TRW	Rocketdyne
Pressures (0-15 psia)				
Test Chamber	No	Yes	Yes	Yes
Fuel Injector	No	Yes	Yes	No
Fuel Duct (Inj.)	No	No	Yes	No
Oxid. Injector	No	No	Yes	No
Oxid. Duct (Inj.)	No	No	Yes	No
Temperatures				
Injector Surface	No ◁	No ◁	Yes	Yes
Injector Fuel Passages	Yes	Yes	Yes	Yes
Injector Oxid Passages	No ◁	No ◁	Yes	Yes
Fuel Duct Surface	No ◁	No ◁	Yes	Yes
Fuel Duct Immersion	Yes	Yes	Yes	Yes
Oxid Duct Surface	No ◁	No ◁	Yes	Yes
Oxid Duct Immersion	No ◁	No ◁	Yes	Yes
Flow Rates	No ◁	No ◁	No ◁	No ◁

◁ Instrumentation installed, data not reported

Pressure data were adequate during the TRW and Bell Phase II tests. However, an instrumentation error resulted in the loss of usable pressure data for all Rocketdyne injector tests. Temperature data for the TRW and Rocketdyne injector tests were excellent because data on both the duct immersion and duct surface thermocouples were reported. Comparison of the data from these two types of thermocouples was used to establish the accuracy and response of the duct surface thermocouples used during the Seattle hot firing tests. (No immersion thermocouples were installed in the injector during the Seattle tests). Reported temperature data were limited to four immersion thermocouples during the Phase and Phase II Bell injector tests, although a total of 14 thermocouples were installed. No flow rate data were reported, although flow rate instrumentation was installed for all tests except the Phase I oxidizer tests.

A.6.2 Adequacy of Cold Flow Testing (Continued)

The following test facility summary shows the factors considered during the evaluation of test facility performance for each test series.

Facility Factor	Injector Test Series			
	Bell Phase I	Bell Phase II	TRW	Rocketdyne
Injector Orifices Choked	Yes [1]	Yes	[2]	Yes [1]
Initial Injector Temperature Variations				
Maximum @ 40°F Nom.	14	16	15°F	4°F
Average @ 40°F Nom.	7	10	10.5°F	2.6°F
Maximum @ 70°F Nom.	7	11	7°F	4°F
Average @ 70°F Nom.	2.4	7.5	3.5°F	3.4°F
Propellant Helium Saturation Level	[3]	[3]	[3]	[3]

[1] Based on similarity to Bell Phase II tests. Pressure data not available for these tests.

[2]. Injector unchoked between approximately 5 seconds and 200 seconds. See Figure 3-16 for typical test results.

[3] Not measured and not controlled.

The test facility pressure was sufficiently high during all TRW descent injector tests to indicate probable injector orifice unchoking during a significant period of each test. Typical duct, injector, and altitude chamber pressure data from Test #18 are shown on Figure A-13. These data show that the ratio of altitude chamber pressure to injector pressure was greater than 0.5 for the time period from 5 to 200 seconds. The significant increase in altitude chamber pressure is attributed to the relatively long propellant flow duration (3.0 to 3.7 seconds) which caused the total propellant flow to be approximately twice as large as the total ascent injector propellant flow. The facility steam ejectors could not remove the ejected propellant fast enough to maintain the required test chamber pressure. The effect of injector orifice unchoking cannot be rigorously evaluated because all descent injector tests exhibited high injector pressure ratios. However, it is apparent that the fuel-side temperature data show a significant time delay (up to 60 seconds) before evaporative cooling began. The low pressure difference across the injector orifices reduced the mass flow rate through the injector, which slowed the evaporative cooling process and increased the fuel residuals present at any time up

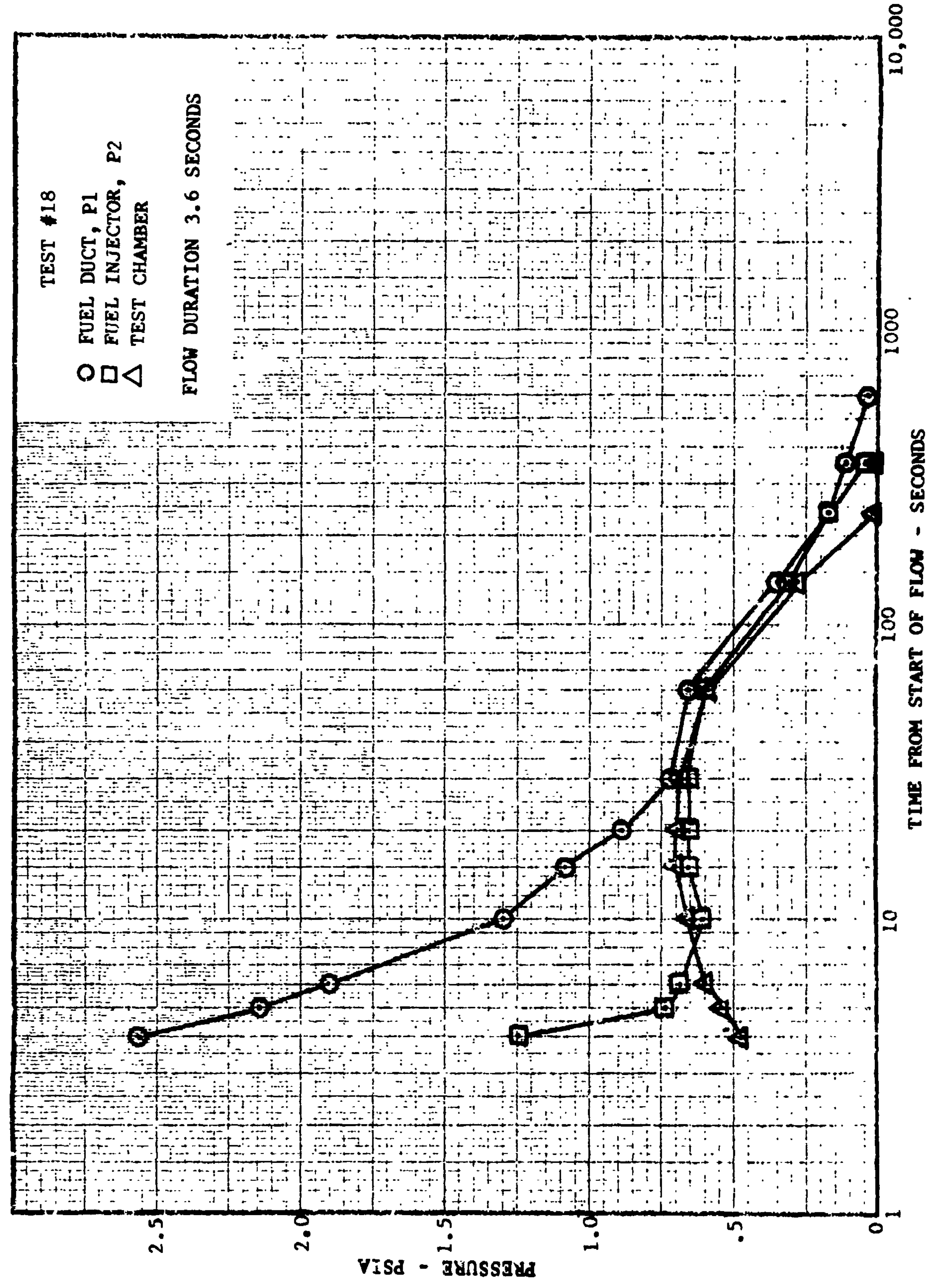


FIGURE A-13 PRESSURE HISTORY FOR TRW INJECTOR AND ARC TEST FACILITY

A.6 ^ Adequacy of Cold Flow Testing (Continued)

to residual depletion.

The nominal allowable variation in initial injector temperature was ± 5°F. This requirement was met during all of the Rocketdyne injector tests and during most of the TRW and Phase II Bell injector tests. Data were not reported for a large number of the Phase II Bell injector tests which were considered to be unsuccessful because of excessive initial temperature variations. The temperature variations may have caused some scatter in the mass residual data, but did not appear to significantly affect the phase history or temperature history data.

The test requirements for the ARC cold flow tests did not specify either control or measurement of the gas saturation level of the propellants. As a result, the effect on test data of variations in this parameter cannot be isolated. The effervescence of dissolved gases, principally helium and nitrogen, may have a significant effect on the amount of residual propellants ejected immediately after flow termination. When the propellant pressure decays below the gas saturation pressure, gas bubbles come out of solution and can force liquid out of the injector. The effect of helium and nitrogen effervescence cannot be established because the gas saturation levels are not known.

APPENDIX B - ASCENT ENGINE TESTING AT SEATTLE

B.0 GENERAL

This appendix describes the Seattle test facility and test article used in testing the ascent engine. The test series and engine instrumentation are described, and a review is made of test validity for ascent engine restarts. Detailed information on the test facility, test article, instrumentation and test series conduct is presented in References 3, 4, and 5.

B.1 TEST FACILITY AND TEST APPARATUS

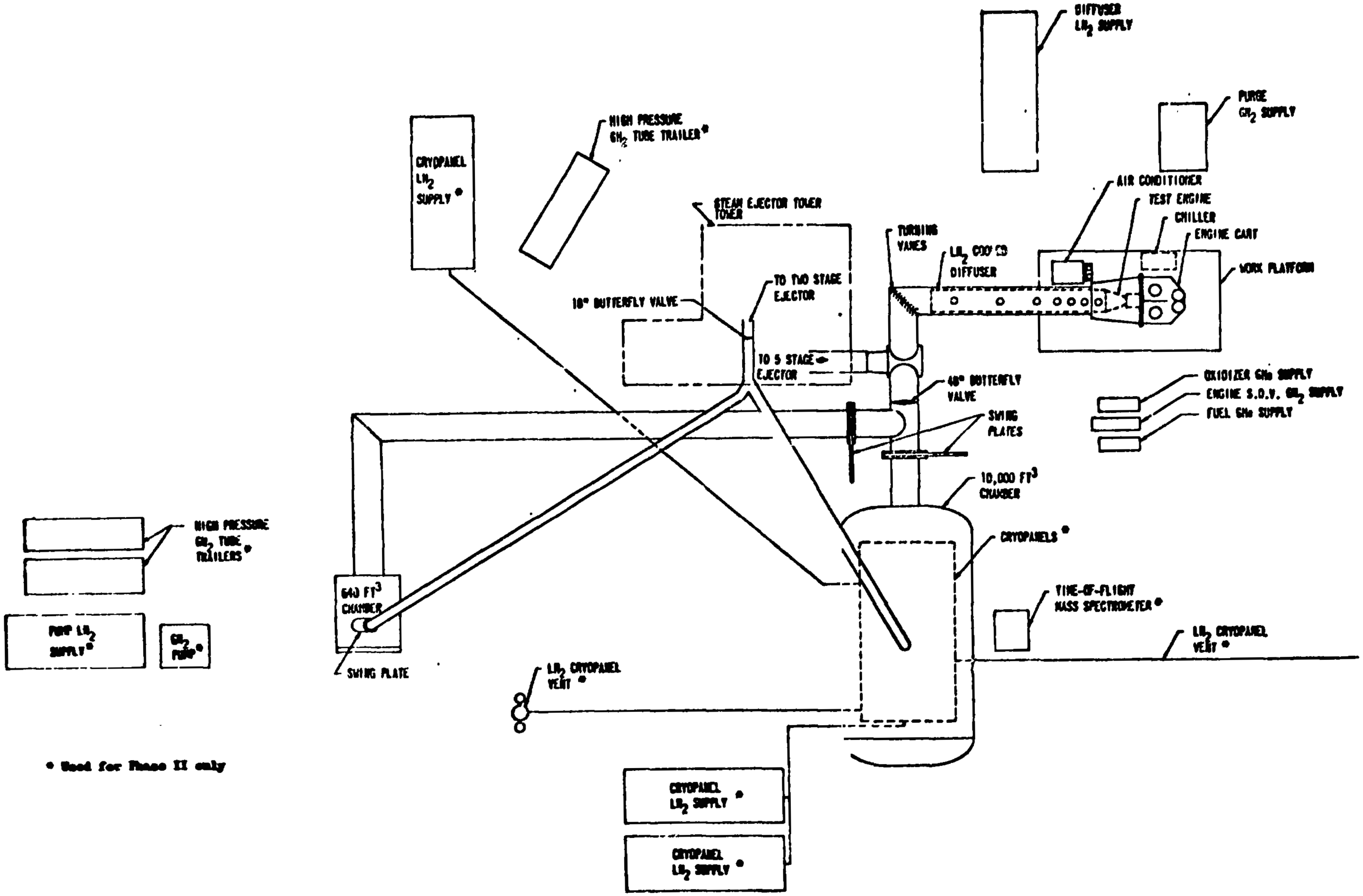
B.1.1 Test Facility

The basic test facility used for the LM Ascent and Descent Engine Restart Capabilities Test Program is the high altitude - high temperature environmental test facility located at Area 34 at the Boeing Company's Tulalip Test Site. This facility has two environmental chambers, two steam ejector systems, a steam generating plant, a cooling tower, a control building, a high bay assembly building, an air lock and air bubble structure plus other ancillary buildings and services.

Altitude pumping capability is provided by separate two stage and five stage steam ejector systems custom made for this installation by the Elliott Company. Both ejectors operate on 500 psig steam. The two stage system has a relatively high pumping capacity of 1225 pounds per hour of 70°F air at a relatively low design altitude of 96,000 feet (9.91 mm Hg). The five stage system is complementary in that its pumping capacity is 140 pounds per hour at its design altitude of 200,000 feet (0.169 mm Hg). Both systems can be run together on either or both chambers to increase the pumping capacity. The limiting conditions are the design altitude of the two stage system, or the mass flow desired. The pumping capacity is the combined mass flow of the systems at the particular altitude. The five stage system is connected to the test chambers by 48 inch diameter ducts. A remotely operated butterfly valve in the duct serves to isolate the chambers from the ejectors. The two stage system is connected to the chambers by 18 inch diameter ducts, also with a remotely operated butterfly valve installed. For control of altitude and gas flows both chambers and ducts are equipped with atmospheric and nitrogen gas bleed valves.

The basic facility was modified to meet the test requirements which specified a firing chamber volume of 600 cubic feet. It was necessary to design and build a new firing chamber and close couple it to the five stage ejector system. Schedule requirements made it mandatory to use existing hardware wherever possible to reduce the duration of the design and buildup phases. The facility layout used for the test program is shown in Figure B-1. The firing chamber is a truncated mild steel cone, 80 inches long with an 80 inch diameter at the engine end and a 69 inch diameter at the exhaust duct end. The firing chamber exhaust duct and diffuser duct were inclined 8° below the horizontal.

A liquid nitrogen cooled diffuser duct was installed inside the facility exhaust duct to increase firing chamber altitude during engine operation. The cooled duct was installed after the Phase I test series was completed. The duct assembly was 38 inches in diameter and 26 feet long, giving a length to diameter ratio of 8.2. It was



* Used for Phase II only

FIGURE B-1 - SEATTLE TEST FACILITY

B.1.1 Test Facility (Continued)

positioned so the engine nozzle exit plane was "buried" inside the diffuser inlet approximately one inch. Circumferential clearance was about one inch around the Descent engine nozzle and 2 inches around the Ascent engine nozzle.

The test engine and firing chamber closure door were bolted to an engine cart. The cart was equipped with vee-grooved wheels that matched the rails in the clean assembly air bubble and shop assembly building as well as rails on the firing chamber support stand. Installed on the cart were propellant run tanks, recirculation pumps, heat exchangers, purge and pressurant gas control systems, filters, valves, plumbing and instrumentation necessary for propellant conditioning and engine operation.

B.1.2 Test Article

The Ascent Engine used in the Phase I testing was S/N 0003, Model RS 1801A. This engine had gone through an acceptance test series (five reactive tests for mixture ratio calibration and two reactive altitude performance tests) and then was shipped to the White Sands Test Facility (WSTF). There, it underwent a checkout firing and a Mission Duty Cycle test series. A new thrust chamber was installed while the engine was at WSTF and then approximately 18 abort start tests were run. Subsequently, the engine was returned to Rocketdyne where 4 more mixture ratio tests were run. The engine was rebuilt in preparation for returning the engine to WSTF. However, before shipment to WSTF, the engine was chosen for the subject test program. A used thrust chamber was then installed, and the engine shipped to Boeing. A total of approximately 31 reactive tests had been performed on the engine prior to the 5 tests of Phase I. Test facility solenoid valves were used to control fuel flow to the engine ball valves during the Phase I tests. Prior to the Phase II tests, the facility solenoid valves were removed, and flight-type fuel prevalves were installed.

The ascent engine used in Phase II testing was the same unit tested during Phase I, except that the thrust chamber was replaced with a modified chamber containing three quartz windows. The thrust chamber windows allowed photographic observation of ignition, combustion, shutdown, coast, and restart phenomena. The test article was designated "Part Number XEOR 919740, Serial Number 003(A)" by the manufacturer, Rocketdyne Division of North American Rockwell Corporation.

B.1.3 Ascent Engine Instrumentation

The same basic instrumentation was used in both Phase I and Phase II of the Seattle restart tests. Instrumentation was provided to measure engine chamber pressure, propellant pressure and temperatures, test cell pressures and temperatures, and engine accelerations. Instrumentation locations are shown in Figure B-2 for the Phase II engine pressure measurements, and Figure B-3 for the Phase II engine temperature measurements.

Pressures in the combustion chamber, propellant feed system, test cell, and valve actuator vent lines were measured by strain gage type transducers. An Alphatron pressure transducer was installed in the large test chamber. Transducer calibration was carried out by the supplier and/or the Tulalip Calibration Facility, with recording equipment set to the calibrations. Temperatures were measured by surface bonded thermocouples which were calibrated at the Tulalip test facility. During the LMAE tests, all accelerometers were provided and calibrated by Rocketdyne. Tables B-1 and B-2 give the pertinent instrumentation information for Phase I and Phase II, respectively.

Primary data were recorded on oscillographs, strip charts, and on an FM tape. The tape record was used to replay selected critical chamber pressure and accelerometer data onto an oscillograph at the speed necessary to resolve and separate traces. Data selected by GAC were hand reduced and plotted, and compiled to form a "data pack" for each test series. Additional data recording equipment was added for Phase II.

Photographic coverage of the Phase I tests was limited to a 16 mm Milliken camera viewing the vent lines of the engine ball valve actuators. Photographs were taken at 24 frames per second. Photographic coverage of the Phase II tests was expanded to provide high speed (1000 frames/second) and low speed (24 frames/second) photographs of the injector face. The injector face was photographed through two quartz windows installed in a modified thrust chamber. Lighting was provided by a photoflood lamp aimed through a third quartz window. The camera views, and locations of the windows, cameras, and floodlight are shown in Figures 4-1 and 4-2 (Volume 1) of this report.

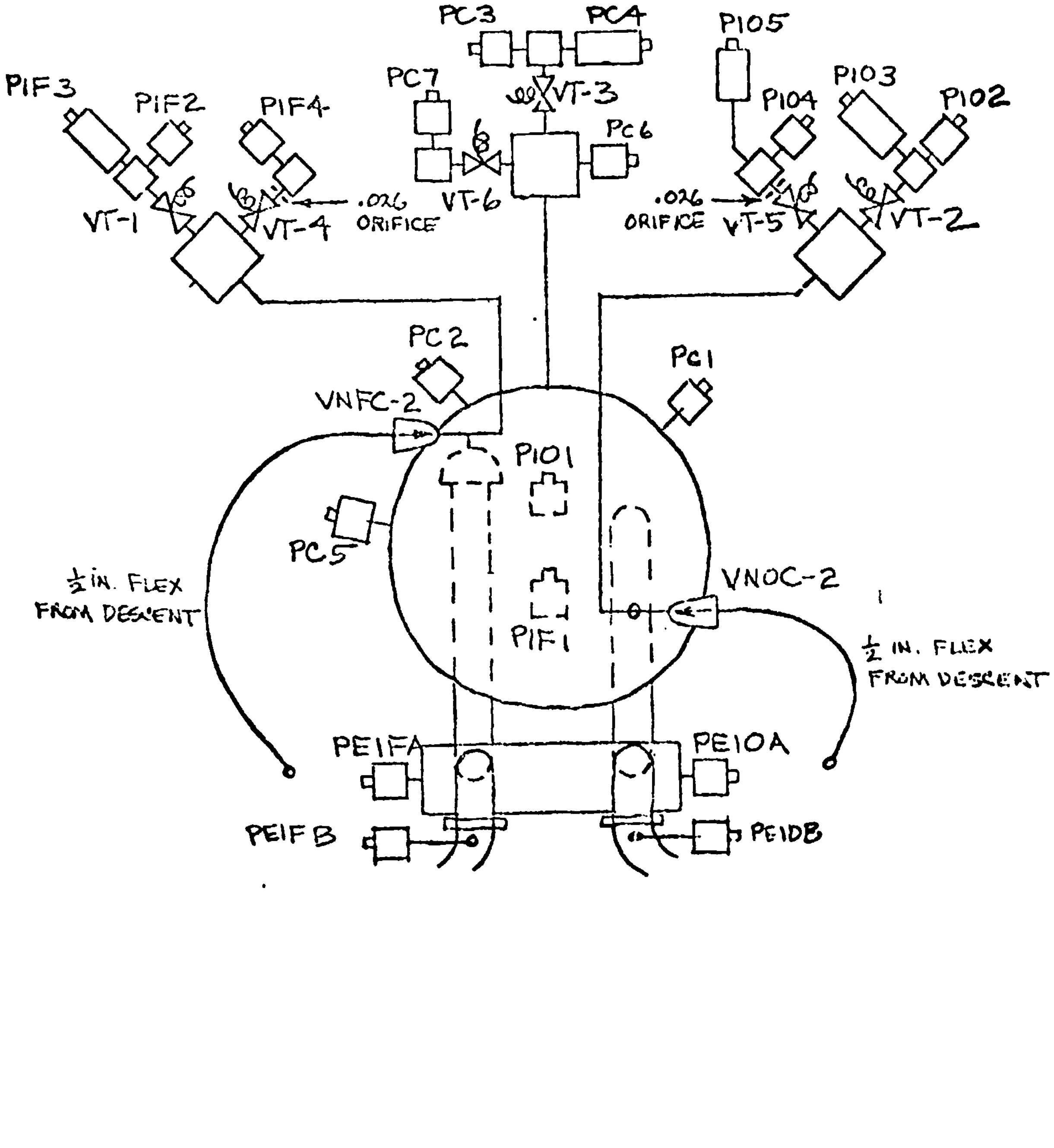
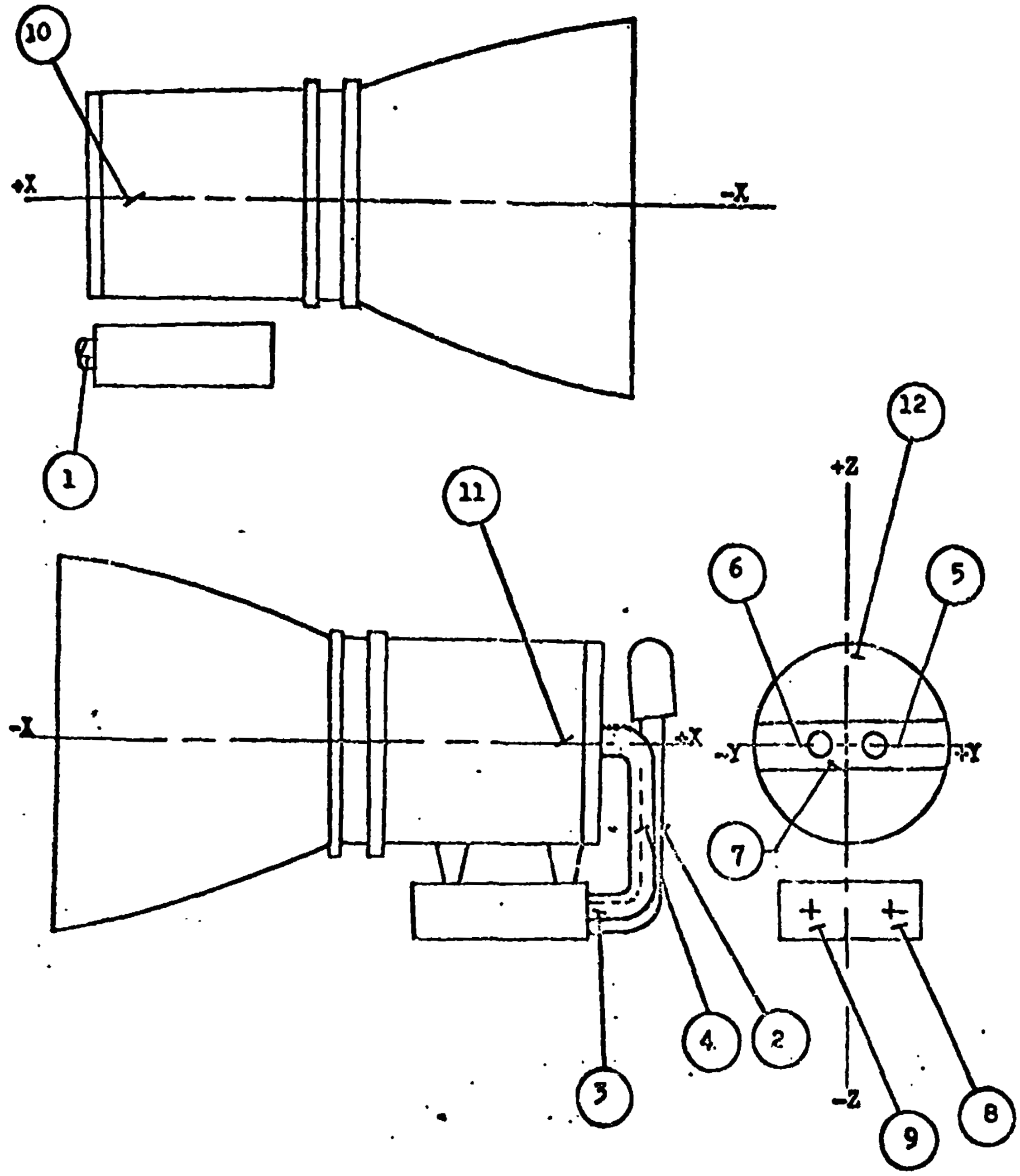


FIGURE B-2 SEATTLE APS ENGINE PRESSURE MEASUREMENTS



NUMBERS 1-12 REFER TO THERMOCOUPLES TE-1 THROUGH TE-12, RESPECTIVELY

FIGURE B-3 SEATTLE AFS ENGINE TEMPERATURE MEASUREMENTS

MEASUREMENT	SYM	INSTRUMENT	RANGE	SERIAL NO.	FREQ. RESP.	ACC.
Accelerometer	AX2	ES16412A	1000 G	2186	2000 cps	10%
Accelerometer	AY2	ES16011	1000 G	2847		
Accelerometer	AZ2	ES16011	1000 G	2850		
Ox. Manifold Press. No. 2	PI02	Data Sen. 943	0-15 psia		1 cps	5%
Fuel Manifold Press No. 2	PIF2	Data Sen 1032	0-15 psia			
Fuel Manifold Press No. 3	PIF3	PACE 26896	0-0.3 psia			
Fuel Manifold Press No. 4	PIF4	Data Sen 1000	0-200 psig		100 cps	
Chamb. Press. 3	PC3	Data Sen 854	0-15 psia		1 cps	
Chamb. Press. 4	PC4	PACE 2689	0-0.3 psia		1 cps	
Chamb. Press.	PC5	Data Sen	0-200 psig		100 cps	
Chamb. Press.	PC6	Data Sen 1211	0-200 psia			
Chamb. Press.	PC7	Data Sen 1297	0-100 psig			
Ox. Supply	PEI0	Data Sen 229	0-350 psig			
Fuel Supply	PEIF	Data Sen 862	0-350 psig			
Ox. Manifold Press. No. 3	PI03	PACE 26720	0-0.3 psia		1 cps	
Fuel Duct U/S	TE1	CU/CON TRW	-100 to 200°F		N.A.	N.A.
Fuel Duct Mid Point	TE2	CU/CON TRW	-100 to 200°F			
Oxid Duct U/S	TE3	CU/CON TRW	-100 to 200°F			
Oxid Duct Mid Point	TE4	CU/CON TRW	-100 to 200°F			
Injector Flange 9 o'clock	TE5	CU/CON TRW	-100 to 200°F			
Injector Flange 3 o'clock	TE6	CU/CON TRW	-100 to 200°F			
Injector Center	TE7	CU/CON TRW	-100 to 200°F			
Fuel Valve Outlet	TE8	CU/CON TRW	-100 to 200°F			
Oxid Valve Outlet	TE9	CU/CON TRW	-100 to 200°F			
Combustion Zone +Y	TE10	CU/CON TRW	-100 to 200°F			
Combustion Zone -Y	TE11	CU/CON TRW	-100 to 200°F			
Injector Flange 6 o'clock	TE12	CU/CON TRW	-100 to 200°F			
Ox Tank Ullage	POTA	Taber 624962	0-300 psig			
Ox Tank Ullage	POTB	Taber 624964	0-300 psig			
Fuel Tk Ullage	PFTA	Taber 652103	0-300 psig			
Fuel Tk Ullage	PFTB	Taber 652102	0-300 psig			
Test Chamber Altitude	PA1A	Datametrics	1-0MM MMHg			
Test Chamber Altitude	PA1B	Datametrics	10 MMHg			

TABLE B-1 LMAE PHASE I INSTRUMENTATION

MEASUREMENT	SYM	INSTRUMENT	RANGE	SERIAL NO.	FREQ. RESP.	ACC.
Test Chamber Altitude	PA1C	Datametrics	760 MMHg		N.A.	N.A.
10,000 Cu.Ft. Altitude	PA2	Alphatron	N.A.		↓	↓
Test Chamber Altitude	PA3	Alphatron	N.A.			
Ox. Manifold	PI01	Data Sen 2380	0-1000 psig		2000 cps	5%
Fuel Manifold	PIF1	Data Sen 2359	0-1000 psig		2000 cps	
Ox. Side Chamber Press.	PC1	Photocon Ser 6924	0-500 psia		10,000cps	
Fuel Side Chamber Press.	PC2	Photocon Ser 7098	0-500 psia		10,000cps	
Accelerometer	AX1	Flight Co.321H-HT-1	25G RMS	1737	2000 cps	10%
Accelerometer	AY1	Flight Co.321H-HT-1	25G RMS	1733		
Accelerometer	AZ1	Flight Co.321H-HT-1	25G RMS	1735	↓	↓

TABLE B-1 LMAE PHASE I INSTRUMENTATION (Continued)

MEASUREMENT	SYM	INSTRUMENT	RANGE	SERIAL NO.	FREQ. RESP.	ACC.
Oxidizer Manifold	PIO2	Data Sensor PB519A-4	0-15 psia	943	1 cps	5%
Oxidizer Manifold	PIO3	Pace Wiancko P1A-0.3 psia	0-.3 psia	26720		
Fuel Manifold	PIF2	Data Sensor PB519A-4	0-15 psia	1032		
Fuel Manifold	PIF3	Pace Wiancko P1A-0.3 psia	0-.3 psia	26721		
Chamber Pressure	PC3	Data Sensor PB519A-4	0-15 psia	854		
Chamber Pressure	PC4	Pace Wiancko P1A-0.3 psia	0-.3 psia	26719		
Oxidizer Supply Press.	PEIOB	Taber 206-SA	0-300 psig	624962		
Fuel Supply Press.	PEIFB	Taber 206-SA	0-300 psig	652103		
Oxidizer Tank Press.	POT	Taber 206-SA	0-300 psig	624964		
Fuel Tank Press.	PFT	Taber 206-SA	0-300 psig	652102		
Oxidizer Supply Press.	PEIOA	Data Sensor PB5198-4	0-350 psia	229	100 cps	
Fuel Supply Press.	PEIFA	Data Sensor PB5195-4	0-350 psia	862		
Oxidizer Manifold Press.	PIO5	Taber 226-SA	0-300 psia	624951		
Diffuser Exit Press.	PDE	Taber 254-SA	0-15 psia	651070		
Oxidizer Manifold Press.	PIO4	Data Sensor PB519S-4	0-200 psia	901		
Fuel Manifold Pressure	PIF4	Data Sensor PB519S-4	0-200 psia	612		
Chamber Pressure	PC5	Data Sensor PB519S-4	0-200 psia	1207		
Chamber Pressure	PC6	Data Sensor PB519S-4	0-200 psia	1211		
Chamber Pressure	PC7	Data Sensor PB519-S-4	0-200 psia	1297		
Ox. Manifold Pressure	PIO1	Photocon 307-2560	0-2000 psig	PRP8444	10,000 cps	
Fuel Manifold Pressure	PIF1	Photocon 307-2560	0-2000 psig	9704		
Chamber Pressure	PC1	Photocon 775-2170	0-500 psig	PRP6924		
Chamber Pressure	PC2	Photocon 775-2170	0-500 psig	PRP7098		
Test Chamber Pressure	PA1A	Datametrics 536	0-10 mmHg	1012	100 cps	
Test Chamber Pressure	PA1B	Datametrics 536	0-100 mmHg	1013		
Test Chamber Pressure	PA1C	Datametrics 536	0-1000 mmHg	1014		
Test Chamber Pressure	PA3	Alphatron	0-1000 mmHg		1 cps	10%
10,000 cu.ft. Chamber Pres	PA2	Alphatron	0-1000 mmHg			
X Plane Accel.	AX1	Columbia Research Lab. 321-H-HT-1	25G RMS	1737	10,000 cps	5%
Y Plane Accel.	AY1	Columbia Research Lab. 321-H-HT-1	25G RMS	1733		
Z Plane Accel.	AZ1	Columbia Research Lab. 321-H-HT-1	25G RMS	1733		

TABLE B-2 LMAE PHASE II INSTRUMENTATION

MEASUREMENT	SYM	INSTRUMENT	RANGE	SERIAL NO.	FREQ. RESP.	ACC.
X Plane Accel.	AX2	Electra Scientific 6412A	1000G P-P	2186	10,000cps	5%
Y Plane Accel.	AY2	Electra Scientific 6011	1000G P-P	2847		
Z Plane Accel.	AZ2	Electra Scientific 6011	1000G P-P	2850		
Fuel Duct Up Stream	TE1	CU/CON	-100 to+200°F		1 cps	±5°F
Fuel Duct Mid Point	TE2	CU/CON	-100 to+200°F			
Oxidizer Duct Up Stream	TE3	CU/CON	-100 to+200°F			
Oxidizer Duct Mid Point	TE4	CU/CON	-100 to+200°F			
Injector Flange 9 o'clock	TE5	CU/CON	-100 to+200°F			
Injector Flange 3 o'clock	TE6	CU/CON	-100 to+200°F			
Injector Center	TE7	CU/CON	-100 to+200°F			
Fuel Valve Outlet	TE8	CU/CON	-100 to+200°F			
Oxidizer Valve Outlet	TE9	CU/CON	-100 to+200°F			
Combustion Zone +Y	TE10	CU/CON	-100 to+200°F			
Combustion Zone -Y	TE11	CU/CON	-100 to+200°F			
Injector Flange 6 o'clock	TE12	CU/CON	-100 to+200°F			
OX Tank	TOT	CU/CON TTS	0 -100°F		1 cps	±5°F
Temp Fuel Tank	TFT	CU/CON TTS	0 -100°F			
Diffuser Cell End Gas Temp	TDC	CR/AL	0 to 2000°F			±20°F
Diffuser Exit Temp	TDE	CR/AL	0 to 2000°F			"
Oxidizer Supply Temp	TEIO	CU/CON	0 to 100°F			±5°F
Fuel Supply Temp	TEIF	CU/CON	0 to 100°F			
Diffuser Wall Cell End Temp	TW1	CR/AL	-320 to 2000°F			±20°F
Diffuser Wall Exhaust Duct Temp	TW2	CR/AL	-320 to 2000°F			
[illegible]tametrics Manifold Temp	TDM	W/CON TTS	0 to 200°F			±5°F

TABLE B-2 LMAE PHASE II INSTRUMENTATION (Continued)

B.2 TEST SERIES DESCRIPTION

B.2.1 Phase I Ascent Engine Tests

Phase I of the LM Ascent Restart Capabilities tests was initiated on October 28, 1968 and was completed on October 31, 1968. A checkout firing and four dual pulse firings were accomplished on the Rocketdyne Ascent Engine, Serial Number 0003. The firings were initiated at simulated altitude above 200,000 ft. and the LMAE exhaust nozzle remained choked throughout the firings. Minimum test cell altitude experienced during the first program was 78,000 ft. Initial propellant and hardware temperatures were held constant at 65 ± 5°F. A summary of initial conditions for each series is in Table 4-1, Volume I, of this report.

Engine Checkout Firing - October 28, 1968

The Phase I checkout firing was accomplished on October 28, 1968. The test cell altitude recovery was slow, requiring 150 seconds to regain 200,000 ft. This time on all other tests was 10 to 20 seconds. However, referring to Section B.3.3, it can be seen that the lesser altitude had no effect on test results.

Test Series A-1 - October 29, 1968

One restart was made following a 1500 second coast period, as scheduled.

Test Series A-2,- October 30, 1968

One restart was made following a 300 second coast period, as scheduled.

Test Series A-3 - October 30, 1968

One restart was conducted following a 90 second coast period. During the initial facility pumpdown for test series A-3, leaks were discovered in the test cell door. A new O-Ring was installed and the test proceeded. Test altitude (see Section B.3.3) was adequate for valid test results.

Test Series A-4 - October 31, 1968

One restart was conducted following a 30 second coast period. The engine experienced a chamber pressure spike in excess of 300% of steady state during the restart. The duration of this over pressure was very short, however, and no engine damage was noted during post-test inspection.

B.2.2 Phase II Ascent Engine Tests

The LMAE Phase II tests were conducted between January 15, 1969 and February 8, 1969 at simulated altitudes initially above 200,000 ft. Following the checkout firing, each test series was scheduled to include two restarts. The individual test series conduct is discussed in the following paragraphs, and a summary of initial conditions for each test series is in Table 4-1.

First Engine Checkout Firing - January 15, 1969

The first Phase II checkout firing on the Ascent engine was accomplished on January 15, 1969. An unplanned second firing occurred about 69 seconds after the first firing due to a Tally tape programmer malfunction and an operator error. During the checkout firing, altitude conditions comparable with facility checkout levels were not obtained and two of the three cameras installed malfunctioned. Grumman requested a rerun of the checkout firing.

Second Ascent Engine Checkout Firing - January 20, 1969

The second checkout firing was successfully completed. Some facility difficulties with water lines were experienced due to freezing temperatures. The facility water flow control valve to the ejector intercondenser malfunctioned just at the time the firing occurred, however test results were not affected (see Section B.3.3).

Facility freezing problems prevented continuation of countdown for follow-on test runs. Winterizing was accomplished by installing heating cables and insulation on all facility plumbing subject to freezing. The winterizing program was completed prior to test A-7.

Test Series A-5 - January 24, 1969

Two restarts were made with the required 200 second coast time between starts. Due to the long coast periods, cryopanels were not required and not used. The high speed camera did not operate. No accelerometer data were obtained for the second restart, however, one chamber window cracked.

Test Series A-6 - January 28, 1969

Two restarts with 90 second coast time, as scheduled.

Test Series A-7 - January 30, 1969

Two restarts with 50 second coast time were scheduled. The first restart was successful, the second restart was not. Grumman directed that only one restart be made on future tests. The Ascent engine

B.2.2 Phase II LMAE Tests (Continued)

was removed from the test stand on January 30, 1969 after the third test run of Phase II testing. GAC requested an engine injector screen inspection because of propellant freezing affecting the second restart of the engine. No engine damage was noted, and the engine was reinstalled on the test stand and was ready for the next test by February 2.

Test Series A-8 - February 3

This firing was made on February 3, 1969 with 30 second coast. Restart was very rough. Two engine combustion chamber windows were cracked during the engine restart. One of the engine combustion chamber pressure transducers, a water cooled Photocon furnished with the engine, failed during the test and leaked 90 gallons of water into the engine combustion chamber. The water froze and an ice plug formed in the engine throat. The transducer had been leak-checked prior to testing.

Replacement units were procured from Rocketdyne. Rocketdyne and GAC requested that the injector be flushed with deionized water and dryed with hot gas due to the possibility of water entering the injector head. The high speed Photosonics camera broke its film during the restart, which was also attributed to the hard start. The slow speed Milliken camera on the SOV vent lines broke its film at the start. Both low speed cameras were overhauled and checked in the Kent vacuum chamber.

Test Series A-9 - February 6

This firing was accomplished with a 15 second coast. One engine combustion chamber window was cracked and required replacement. The slow speed Milliken camera on the SOV vent lines again broke its film. No difficulty was experienced with this camera installation all through Phase I testing. Rubber pad shock mounts were installed on the camera mount to isolate the camera from engine shocks.

Test Series A-10 - February 6

This test was made with scheduled 10 second coast and one restart.

Test Series A-11 - February 7

This test was made with a single restart and a 2.5 second coast period.

B.2.2 Phase II LMAE Tests (Continued)

Test Series A-12 - February 8

This test was made with 1 second coast time and one restart. This completed the Ascent Engine testing. The engine was removed from the test stand and shipped to Rocketdyne.

Test Series A-5, A-6, and A-7 were made in the sub-freezing temperatures with varying problems.

1. The cameras were not functioning and this was found to be caused by film breaking and an apparent overload on the high speed camera drive motor. A heater was connected on the high speed camera and the 1200 ft. film magazine was replaced with a 400 ft. magazine. Mylar film was installed in the Milliken camera covering SOV vent lines.

2. The high altitude initial start requirement of 250,000 ft. was hard to reach because of:

   a. Ice in the diffuser.

   b. Frozen seals on the cryopanel chamber door with the possibility of ice there also.

B.3 TEST RESULTS

B.3.1 Validity of Test Results

Review of the test data showed that all of the Phase II LMAE tests are valid for restart investigations, with the exception of test AOII-1, which was considered an unsuccessful checkout. Valid comparisons with nominal engine data cannot be made for the Phase I tests, since the fuel prevalves were not of flight configuration.

Test Validity and Limitations

The validity of the Seattle LMAE tests, and the limitations on applicability of the test data are dependent on: compliance with test requirements; accurate setting of initial conditions such as propellant and hardware temperatures; helium saturation; altitude chamber pressure as it affects manifold venting characteristics. The following paragraphs discuss these criteria with respect to Seattle Phase I and Phase II LMAE tests.

The test requirements for Phase I and Phase II testing as outlined in Section 4.1.2 were met in all tests, with the following exceptions:

1. Photographic Coverage - Photographic coverage was not obtained in the following tests due to equipment malfunction or film breakage:

Photosonics	Milliken I	Milliken II
AOII-1	A-5, a, b, c	AOII-1
AOII-2		AOII-2
A-5 a, b, c		A-5 a, b, c
A-8b		A-6 a, b, c
A-12 a, b, c		A-7 a, b, c
		A-8 a, b
		A-9 a, b
		A-10 a, b
		A-12 a, b

2. Facility Requirements - The minimum Phase II initial altitude requirement of 250,000 feet was not met in any of the LMAE tests. In each case the lesser altitude attained was accepted and signed off by a GAC representative. The required altitude for restart of 200,000 feet was not met in tests A-9, 10, 11, and 12 because of the time required for facility altitude recovery following a firing, and because the cryopanels did not function as expected.

The initial propellant and hardware temperatures and propellant helium saturation requirements specified in the Phase I and Phase II test

B.3.1 Validity of Test Results (Continued)

matrices were met in all LMAE tests, with the exception of AOII-2. In that test, the initial propellant temperature of 44°F fell outside the 40 +0 -10°F range called for. Since the test was a checkout, with no restarts, the initial propellant temperature had no impact on restart test results.

The effect of the altitude chamber pressure on restarts is zero unless it is high enough to unchoke the engine nozzle. Thus, if the nozzle remains choked during the critical manifold boiling and freezing processes, the "downstream" pressure will have no effect. Referring to Table B-3, it can be seen that, for Phase II, the nozzle unchoked only in tests A-6a, A-6b, and A-6c. However, manifold pressure data indicate that the delta pressure across the injector orifices was much in excess of that required for simple theoretical choking of the orifices. Sharp-edged orifice and two-phase flow considerations preclude this pressure differential from actually isolating the manifolds from the chamber, but it is safe to assume that there was sufficient decoupling to make test series A-6 valid.

The Phase I test pressure results were not as clear as those for Phase II. GAC data reduction requirements differed from test to test, resulting in somewhat spotty coverage of the relevant parameters. The Boeing Phase I report stated, however, that "the combustion chamber throat remained choked until in the range of 200 seconds". This indicates that the injector was in fact isolated from events in the altitude chamber.

The repeatability of starts for tests having similar initial conditions is applicable only for initial starts in the Seattle LMAE tests. This is because restarts were performed only once or twice at similar coast times, and if repeated at the same coast time were run at different temperatures and degrees of helium saturation. Also, since non-flight prevalves were used in Phase I, only the Phase II initial starts are acceptable for comparison.

The initial starts for Phase II can be compared to a nominal engine on the basis of time to ignition or first Pc rise and peak Pc. For the Phase II initial starts AOII-2 through A-7-all dry starts- the time to initial Pc rise varies from .250 to .279 seconds. The Spacecraft Operational Data Book (SODB) (Reference 12) Volume II, Part I shows a range of time to initial Pc rise of .360 to .370 seconds for dry starts. For the wet initial starts A-8 through A-12 the time to initial Pc rise varies from .231 to .248 seconds, compared with .190 to .330 seconds for the SODB nominal engine. The Phase II LMAE peak initial start Pc varies from 170 to 182 psia. The SODB nominal peak Pc is 180 psia.

It can be seen from the above that the initial start characteristics of the Seattle LMAE Phase II tests compare well with nominal engine

RUN NO.	FIRST FIRING INITIAL ALT. (FT.)	FIRST FIRING UNCHOKE TIME (SEC)	FIRST FIRING TIME TO 200K FT. (SEC.)	SECOND FIRING INITIAL ALT. (FT.)	SECOND FIRING UNCHOKE TIME (SEC.)	SECOND FIRING TIME TO 200K FT. (SEC.)	THIRD FIRING INITIAL ALT. (FT.)	THIRD FIRING UNCHOKE TIME (SEC.)	THIRD FIRING TIME TO 200K FT. (SEC.)	COAST TIME (SEC)	POST RUN COAST (30 min ALT. (FT.)
AO-II-1	222,000	7.5	170	-	-	-	-	-	-	-	220,000
AO-II-2	221,000	(1)	67	-	-	-	-	-	-	-	220,000
A-5	212,000	(1)	27	210,000	(1)	20	215,000	(1)	12	200	219,000
A-6	228,000	18	50	205,000	16	78	203,000	13	1100	90	228,000
A-7	232,000	(2)	11	208,000	(2)	14	206,000	(1)	11.5	60	208,000
A-8	245,000	(2)	23	219,000	(1)	14	-	-	-	30	228,000
A-9	212,000	(2)		187,000	(1)	48	-	-	-	15	208,000
A-10	222,000	(2)		190,000	(1)	17	-	-	-	10	245,000
A-11	224,000	(2)		190,000	(1)	16	-	-	-	2.5	245,000
A-12	215,000	(2)		160,000	(1)	26	-	-	-	1.0	219,000

(1) Did not unchoke prior to reaching 200,000 ft. altitude
(2) Did not unchoke before the next firing
(3) Remained choked thereafter until above 200,000 ft. altitude
(4) Remained choked thereafter before the next firing.

TABLE B-3 ASCENT ENGINE NOZZLE CHOKING SUMMARY

B.3.1 Validity of Test Results (Continued)

performance, except for dry start Pc initial rise. However, the dry Pc rise time exhibits the correct behavior, that is, being slower than that of a wet start.

Thus, the Seattle Phase II LMAE initial starts appear consistent with those of "nominal" engines. The Phase I initial starts, using non-flight configuration prevalves, cannot be so compared with nominal flight configuration hardware results.

APPENDIX C - DESCENT ENGINE TESTING AT SEATTLE

C.0 GENERAL

This appendix describes the Seattle test facility and test article used in testing the descent engine. The test series and engine instrumentation are described, and an assessment is made of test validity for descent engine restarts. Detailed information on the test facility, test article, instrumentation and test series conduct is presented in References 3, 4, and 5.

C.1 TEST FACILITY AND TEST APPARATUS

C.1.1 Test Facility

The basic facility used for the LM Ascent and Descent Engine Restart Capabilities Test Program is described in Appendix B.1.1.

C.1.2 Test Article

The engine used in the Phase I portion of the test program was S/N 1033 (P/N SK-403936-1). This engine had undergone extensive developmental testing in association with other programs both at the TRW Capistrano Test Site (CTS) and at the White Sands Test Facility (WSTF) prior to being used in the subject test program. Prior to being shipped to Boeing by TRW, several hardware and instrumentation changes were made to the 1033 engine. These changes are detailed in Reference 13.

The descent engine used in Phase II testing was the same as the unit tested during Phase I except that the thrust chamber was replaced with one containing three quartz windows. These windows allowed photographic observation of ignition, combustion, shutdown, coast, and restart phenomena. The Phase II test article was designated "Part Number SK403936-1-2, Serial Number 1033" by the manufacturer, TRW, Inc.

C.1.3 Descent Engine Instrumentation

The instrumentation used for the LMDE tests was the same as that used for the LMAE tests (See Section B.1.3) with the exceptions that the flight instrumentation was different and Boeing accelerometers were used on the LMDE along with the flight accelerometers. Instrument locations for Phase I and Phase II engine system measurements are shown in Figure C-1 and Tables C-1 and C-2 give the instrument characteristics pertinent to the LMDE tests.

Photographic coverage of the Phase I and Phase II LMDE tests was the same as that used for the LMAE tests (See Section B.1.3), with the exception that the longer duration LMDE firings (2 to 3.5 seconds) precluded high-speed coverage of an initial start plus two restarts. The Phase II high-speed (1000 fps) coverage was therefore restricted to the two restart firings for all restart tests except D-10. High speed coverage was provided during the Phase II checkout firing, D-O-II.

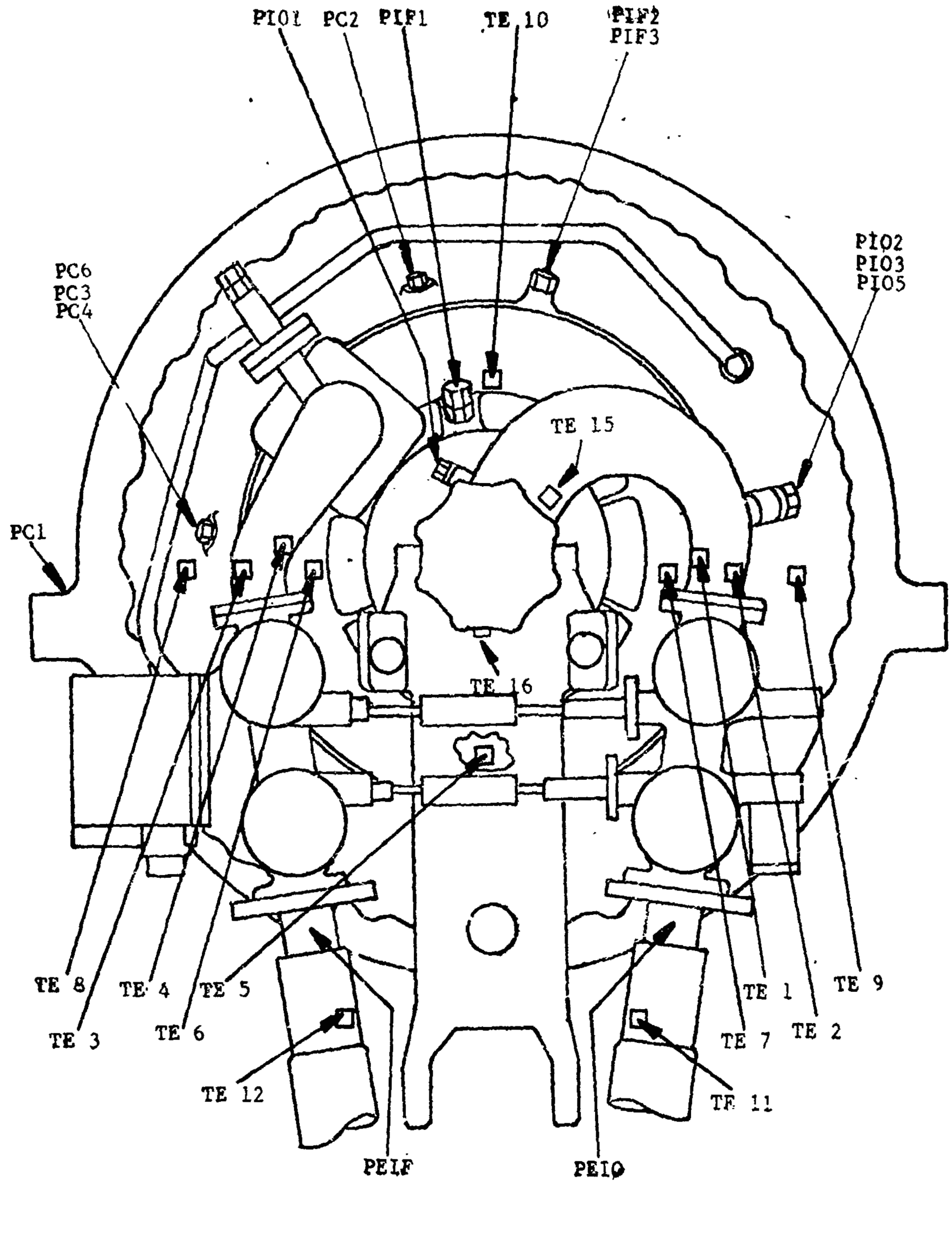


FIGURE C-1 SEATTLE DPS INSTRUMENTATION LOCATIONS

MEASUREMENT	SYM	INSTRUMENT	RANGE	SERIAL NO.	FREQ. RESP.	ACC.
Test chamber altitude	PA1B	Datametrics	10 MMHg		100 cps	5%
Test chamber altitude	PA1C	Datametrics	760 MMHg		100 cps	5%
10,000 Cu.Ft. Altitude	PA2	Alphatron			5 cps	100%
Test chamber altitude	PA3	Alphatron			5 cps	10%
Ox. manifold press. No.1	PI01	Photocon 100033	0 - 200 psia		10,000cps	5%
Fuel manifold press No.2	PIF1	Photocon 100028	0 - 200 psia		10,000 cps	
Chamb. press 1	PC1	Photocon 100055	0 - 500 psia		2,000 cps	
Accelerometer injectory body	AX1	Flight 0-5V	0 - 100 G			10%
Accelerometer injector body	AY1	Flight 0-5V	0 - 100 G			
Accelerometer injector body	AZ1	Flight 0-5V	0 - 100 G			
Accelerometer injector body	AX2	Endevco 2242	0 - 1000 G			
Accelerometer injector body	AY2	Endevco 2242	0 - 1000 G			
Accelerometer injector body	AZ2	Endevco 2242	0 - 1000 G			
Ox. manifold press No.2	PI02	Pace 27036	0 - 10 psia		200 cps	5%
Ox. manifold press No.3	PI03	Pace 26721	0 - .3 psia			
Fuel manifold press No.2	PIF2	Pace 27037	0 - 10 psia			
Fuel manifold press No.3	PIF3	Pace 26722	0 - .3 psia			
Chamb. press. 3	PC3	Page 27035	0 - 10 psia			
Chamb. press. 4	PC4	Page 26719	0 - .3 psia			
Chamb. press.	PC2	Micro systems	0 - 200 psia		400 cps	
Oxidizer duct	TE1	CU/CON TRW	-100 to 200°F		1 cps	±5°F
Oxidizer duct	TE2	CU/CON TRW	-100 to 200°F			
Fuel duct	TE3	CU/CON TRW	-100 to 200°F			
Fuel duct	TE4	CU/CON TRW	-100 to 200°F			
Fuel manifold +Z side	TE5	CU/CON TRW	-100 to 200°F			
Fuel manifold +Y side	TE6	CU/CON TRW	-100 to 200°F			
Fuel manifold =Y side	TE7	CU/CON TRW	-100 to 200°F			
Injector mtg. flange ±Y side	TE8	CU/CON TRW	-100 to 200°F			

TABLE C-1 LMDE PHASE I INSTRUMENTATION

MEASUREMENT	SYM	INSTRUMENT	RANGE	SERIAL NO.	FREQ. RESP.	ACC.
Oxidizer elbow +Y side	TE11	CU/CON TRW	-100 to 200°F		1 cps	±5°F
Fuel elbow -Y side	TE12	CU/CON TRW	-100 to 200°F			
Combustion chamb. +Y side	TE13	CU/CON TRW	-100 to 200°F			
Combustion chamb. -Y side	TE14	CU/CON TRW	-100 to 200°F			
Ox tank temp	TOT	CU/CON TTS	0 - 100°F			
Fuel tank temp.	TFT	CU/CON TTS	0 - 100°F			
Diffuser cell end gas temp	TDC	CR/AL TTS	-100 to 2500°F			
Diffuser ejector end gas temp	TDE	CR/AL TTS	0 to 2000°F			
Diffuser wall temp test chamb.	TW1	CR/AL TTS	-320 to 2000°F			
Diffuser wall temp exhaust duct	TW2	CR/AL TTS	-320 to 2000°F			
10,000 Cu.Ft.chamb. gas temp.	TC	CR/CON TTS	0 to 500°F			
Ox interface temp.	TEIO	CU/CON TTS	0 to 100°F			
Fuel interface temp.	TEIF	CU/CON TTS	0 to 100°F			
Ox tank ullage	POTA	Taber 624962	0 - 300 psig		300 cps	5%
Ox tank ullage	POTB	Taber 624964	0 - 300 psig			
Fuel tk ullage	PFTA	Taber 652103	0 - 300 psig			
Fuel tk ullage	PFTB	Taber 652102	0 - 300 psig			
Ox interface	PEIO	Taber 652101	0 - 300 psia			
Fuel interface	PEIF	Taber 624961	0 - 300 psia			
Test chamber altitude	PA1A	Datametrics	1.0MM MMHg			

TABLE C-1 LMDE PHASE I INSTRUMENTATION (CONTINUED)

MEASUREMENT	SYM	INSTRUMENT	RANGE	SERIAL NO.	FREQ. RESP.	ACC.
Oxidizer duct	TE1	CU/CON TRW	$-100^{o}F$ to $200^{o}F$		1 cps	$5^{o}F$
Oxidizer duct	TE2	CU/CON TRW	$-100^{o}F$ to $200^{o}F$			
Fuel duct	TE3	CU/CON TRW	$-100^{o}F$ to $200^{o}F$			
Fuel duct	TE4	CU/CON TRW	$-100^{o}F$ to $200^{o}F$			
Fuel manifold +Z side	TE5	CU/CON TRW	$-100^{o}F$ to $200^{o}F$			
Fuel manifold +Y side	TE6	CU/CON TRW	$-100^{o}F$ to $200^{o}F$			
Fuel manifold -Y side	TE7	CU/CON TRW	$-100^{o}F$ to $200^{o}F$			
Injector mntg flange +Y side	TE8	CU/CON TRW	$-100^{o}F$ to $200^{o}F$			
Oxidizer elbow +Y side	TE11	CU/CON TRW	$-100^{o}F$ to $200^{o}F$			
Fuel elbow -Y side	TE12	CU/CON TRW	$-100^{o}F$ to $200^{o}F$			
Combustion chamber +Y side	TE13	CU/CON TRW	$-100^{o}F$ to $200^{o}F$			
Combustion chamber -Y side	TE14	CU/CON TRW	$-100^{o}F$ to $200^{o}F$			
Oxid. duct near injector	TE15	CU/CON TRW	$-100^{o}F$ to $200^{o}F$			
Injector body	TE16	CU/CON TRW	$-100^{o}F$ to $200^{o}F$			
Ox tank	TOT	CU/CON TTS	$0 - 100^{o}F$			$\pm 5^{o}F$
Temp fuel tank	TFT	CU/CON TTS	$0 - 100^{o}F$			"
Diffuser cell end gas temp	TDC	CR/AL	0 to $2000^{o}F$			$\pm 20^{o}F$
Diffuser exit temp	TDE	CR/AL	0 to $2000^{o}F$			"
Ejector gas temp	TEG	CR/AL	0 to $2000^{o}F$			$\pm 5^{o}F$
Oxidizer supply temp	TEIO	CU/CON	18 to $150^{o}F$			
Fuel supply temp	TEIFF	CU/CON	18 to $150^{o}F$			
10 KFT^3 chamber gas temp	TC	CU/CON	0 to $500^{o}F$			
Diffuser wall cell end temp	TW1	CR/AL	-320 to $2000^{o}F$			$\pm 20^{o}F$
Diffuser wall exhaust duct temp	TW2	CR/AL	-320 to $2000^{o}F$			$\pm 20^{o}F$

TABLE C-2 LMDE PHASE II INSTRUMENTATION (CONTINUED)

MEASUREMENT	SYM	INSTRUMENT	RANGE	SERIAL NO.	FREQ. RESP.	ACC.
Chamber Pressure	PC6	Data Sensor PB519A-28	0-25 psia	1175	100 cps	5%
Oxidizer Manifold	PI01	Photocon 5307-5875	0-200 psia	100033	10,000cps	
Fuel Manifold	PIF1	Photocon 5307-5875	0-200 psia	100028		
Chamber Pressure	PC1	Photocon 5307-5875	0-500 psia	100055		
Chamber Pressure	PC2	Micro Systems 1025-0016G	0-200 psia	Q423		
Test Chamber Pressure	PA1A	Datametrics 536	0-10 mmHg	1012	100 cps	
Test Chamber Pressure	PA1B	Datametrics 536	0-100 mmHg	1013		
Test Chamber Pressure	PA1C	Datametrics 536	0-1000 mmHg	1014		
10,000 Cu.Ft.Chamber Pres	PA3	Datametrics 536	0-100 mmHg	1070		
10,000 Cu.Ft.Chamber Pres	PA2	Alphatron 530	0-1000 mmHg	1	1 cps	10%
Oxidizer Manifold	PI02	Data Sensor PB519A-4	0-15 psia	943		5%
Oxidizer Manifold	PI03	Bace Wiancko P1A-0.3 psia	0-.3 psia	26720		
Fuel Manifold	PIF2	Data Sensor PB519A-4	0-15 psia	1032		
Fuel Manifold	PIF3	Pace Wiancko P1A-0.3 psia	0-.33 psia	26722		
Chamber Pressure	PC3	Data Sensor PB519A-4	0-15 psia	854		
Chamber Pressure	PC4	Pace Wiancko P1A-0.3 psia	0-.3 psia	26719		
Oxidizer Supply Pressure	PEIOB	Taber 206-SA	0-300 psig	624962		
Fuel Supply Pressure	PEIFB	Taber 206-SA	0-300 psig	652103		
Oxidizer Tank Pressure	POT	Taber 206-SA	0-300 psig	624964		
Fuel Tank Pressure	PFT	Taber 206-SA	0-300 psig	652102		
Oxidizer Supply Pressure	PEIOA	Data Sensor PB5198-4	0-350 psis	229	100 cps	
Fuel Supply Pressure	PEIFA	Data Sensor PB519S-4	0-350 psis	862		
Oxidizer Manifold Press	PI05	Alinco 151-C2B	0-50 psia	26207		
Diffuser Exit Pressure	PDE	Taber 254-SA	0-15 psia	651070		
X Plane Accel.	AX1	Columbia Research Lab. 321-H-HT-I	100 G RMS		10,000cps	5%
Y Plane Accel.	AY1	Columbia Research Lab. 321-H-HT-I	100 G RMS			
Z Plane Accel.	AZ1	Columbia Research Lab. 321-H-HT-I	100 G RMS			
X Plane Accel.	AX2	Endevco 2242	0-1000 G P-P	CB42		
Y Plane Accel.	AY2	Endevco 2242	0-1000 G P-P	7612		
Z Plane Accel.	AZ2	Endevco 2242	0-1000 G P-P	7805		

TABLE C-2 LMDE PHASE II INSTRUMENTATION

C.2 TEST SERIES DESCRIPTION

C.2.1 Phase I Descent Engine Tests

Phase I of the LM Descent Restart Capabilities tests was initiated on October 19, 1968 in the Boeing Tulalip Test Site, Area 34 High Altitude Facility and completed on October 24, 1968. A checkout firing and four dual pulse firings were accomplished on the TRW Descent Engine, Serial Number 1033. The firings were initiated at simulated altitudes above 200,000 ft. and the LMDE exhaust nozzle remained choked throughout the firings. Minimum test cell altitude experienced during the first program was 78,000 ft. Initial propellant and hardware temperatures were held constant at 65 $\pm$ 5°F. In all tests the LMDE was started and run in the 10% throttle position, since this is the normal starting mode, and also any higher throttle setting would exceed the pumping capabilities of the steam ejector system at the desired simulated altitudes. Minor problems were encountered in the facility operation. The initial system cleanliness verification procedure was prolonged by recirculating pump seal leakage problems. These pumps were replaced with units from the White Sands Test Facility. After several attempts, the system finally met the TRW cleanliness standards. A summary of initial test conditions is shown in Table 5-1, Volume I, of this report.

<u>Descent Checkout Firing - October 19, 1968</u>

The Descent engine checkout was satisfactorily completed on October 19, 1968.

<u>Test Series D-1 - October 21, 1968</u>

One restart was made following a 1819 second coast period. After the test series, a crack in the exhaust duct was discovered. The crack, caused by liquid nitrogen impingement, was repair-welded and the LN_2 overflow from the diffuser cooling coils rerouted prior to the next firings. The crack (see Section C.3.3) did not compromise test results.

<u>Test Series D-2 - October 22, 1968</u>

One restart was made following a 316 second coast period. This was the hardest DPS start in Phase I tests.

<u>Test Series D-3 - October 23, 1968</u>

One restart was made following a 120 second coast period.

<u>Test Series D-4 - October 24, 1968</u>

One restart was made following a 43.5 second coast period.

C.2.2 Phase II Descent Engine Tests

The Descent Engine Phase II tests were conducted between February 12, 1969 and February 21, 1969 at simulated altitudes initially above 200,000 ft. Following the checkout firing, each test included two restarts. The individual test series conduct is discussed in the following paragraphs, and a summary of initial conditions for each test series is shown in Table 5-1, Volume I, of this report.

Descent Engine Checkout Firing - February 12, 1969

The descent engine checkout firing was completed on February 13, 1969. The firing was satisfactory except that the Milliken camera on the engine vent lines broke its film. Mylar film had not been used in loading the camera due to a separation or cracking of the film emulsion noted on previous tests using this film. Cryopanels were not used on this test, since there were no restarts, and the butterfly-valve was cycled to obtain optimum altitude recovery for the post-firing coast.

Test Series D-5 - February 13, 1969

Two restarts were made with a 120 second coast period, and a firing time of 3.5 seconds. The cryopanels were used, and the 48" butterfly valve was cycled to optimize altitude recovery characteristics.

Test Series D-6 - February 15, 1969

Two restarts were made with a 90 second coast period. The 48" butterfly valve was cycled, and cryopanels were used. The cryopanels did not function as expected, but did not compromise test results (see Section C.3.3). Due to the problem, however, two modifications were incorporated in the cryopanels for D-6 and subsequent tests. First, 3/4" spacer blocks were installed along the longitudinal edges of the LN_2 shroud, providing 20% more flow to the LH_2 panels. Second, electric shaker motors were attached to the panel support in an effort to shake any frozen gases entrapped on the LN_2 panels off onto the LH_2 panels. Following test D-6, flat flow deflector extensions were added to further shield the cryopanels from direct impingement by exhaust gases.

Test Series D-7 - February 17, 1969

Two restarts were made with a 50 second coast period. The butterfly valve and cryopanels were used. The initial firing was "wet", the first such initial start in Phase II DPS tests. The second restart was made at an altitude of only 195,000 feet. Following the test, a crack was found in the diffuser duct, explaining the altitude problem. The cause of the cracking was thought to be the LN_2 cooling for the diffuser duct, and it was decided to use no more LN_2 diffuser

C.2.2 Phase II Descent Engine Tests (Continued)

duct cooling. This would lead to a zero shift in the heat sensitive Datametric pressure transducers in the altitude chamber following the second restart, but this problem was not considered serious. The altitude problem did not affect test results (see Section C.3.3).

Test Series D-8 - February 18, 1969

Two restarts were made with a 15 second coast period. The firing time was changed by GAC to 3.0 seconds. Cryopanels were used, but LN_2 diffuser duct cooling was not. The butterfly valve was cycled as before, but apparently did not open for the second restart due to a broken pilot opening line on the butterfly valve actuator. This failure resulted in a post-firing maximum altitude chamber pressure of 42 mmHg for the second restart. This problem did not affect test results (see Section C.3.3).

Test Series D-9 - February 19, 1969

Two restarts were made with a coast period of 5 seconds. Cryopanels were used and the butterfly valve cycled as before. Due to lack of communication between Test Conductors, the LN_2 cooling was again used for the diffuser duct, which cracked again. Poor altitude recovery times resulted. The LN_2 cooling was not to be used again. It was also discovered that the fuel was depressurized by accident to about 10 psi for about 1 to 2 minutes, just prior to the test. Thus, there was probably not 100% helium saturation for the test. However, (see Section C.3.3) test results were not compromised.

Test Series D-10 - February 19, 1969

Two restarts were made with a coast period of 2 seconds. Cryopanels were used, and the butterfly valve was closed at T + 15.

Test Series D-11 - February 20, 1969

Two restarts were made with a coast period of 375 seconds. The firing times were changed by GAC to 3.0 seconds for the initial start and 2.0 seconds for the restarts. Cryopanels were not used, and the two stage steam ejector was used to pump the 10,000 ft^3 chamber while the butterfly valve was closed during coast. An engine low voltage condition existed during the second restart, with low voltages noted to the propellant valves and flight accelerometers. The accelerometers acted very erratically. A reduction in mass flow and peak chamber pressure for the third firing indicated that perhaps the propellant valves did not open fully.

C.2.2 Phase II Descent Engine Tests (Continued)

Test Series D-12 - February 21, 1969

Two restarts were made with a coast period of 170 seconds. Cryopanels were not used, and the two stage steam ejector was used as in Test D-11. The test was to have been conducted on February 20, 1969, but an unknown facility problem prevented the attainment of sufficient altitude to run the test. No reasons for the problem could be found following facility shutdown. With the dawn of February 21, 1969 another attempt to reach altitude was made successfully, with no reoccurrence of the previous night's problems. This test completed Descent Engine testing.

C.3 TEST RESULTS

C.3.1 Validity of Test Results

Review of the Phase I and Phase II descent test data indicates that all of the descent test results are valid for restart investigations.

Test Validity and Limitations

The validity of the Seattle LMDE tests, and the limitations on applicability of the test data are dependent on: compliance with test requirements; accurate setting of initial conditions such as propellant and hardware temperatures; helium saturation; altitude chamber pressure as it affects manifold venting characteristics. The following paragraphs discuss these criteria with respect to Seattle Phase I and Phase II LMDE tests.

The test requirements for Phase I and Phase II testing as outlined in Section 5.1.2 were met in all tests, with the following exceptions:

1. Photographic Coverage - Photographic coverage was not obtained in the following tests due to equipment malfunction or film breakage:

Photosonics	Milliken I	Milliken II
		DO-II

2. Facility Requirements - The minimum Phase I initial altitude requirement of 200,000 feet was not met in LMDE test D-1. The minimum Phase II initial altitude requirement of 250,000 feet was not met in tests DO-II, D-6, D-7, D-10, D-11, and D-12. In each case the lesser altitude attained was accepted and signed off by a GAC representative. The required altitude for restart of 200,000 feet was not met in tests D-16, D-7C, D-8C, D-9b, c, and D-10b, c because of the time required for facility altitude recovery, and because the cryopanels did not work as expected.

3. Electrical Requirements - The voltage supplied to the LMDE valves was below limits during the third firing of test series D-11 due to a GSE power supply malfunction.

4. Propellant Helium Saturation - The fuel was depressurized to 10 psi for 1 to 2 minutes during test series D-9, resulting in less than 100% helium saturation.

Initial Test Conditions

The initial propellant and hardware temperature requirements specified in the Phase I and Phase II test matrices were met in all LMDE tests.

C.3.1 Validity of Test Results (Continued)

Altitude Chamber Pressure Effects

As explained in Section B.3.3, the altitude chamber pressure will have no effect on restart events so long as the engine nozzle is choked. Referring to Table C-3, it can be seen that the nozzle remained choked for the duration of the coast, except in tests DO-II, D-5b, D-7b, c, D-8b, c, D-9c, and D-10c. However, in these tests, manifold pressure data indicate that there was sufficient delta pressure across the injector to reasonably isolate the manifolds from the chamber. Again, Phase I data are sketchy, but follow the same pattern as Phase II.

Initial Start Repeatability

The repeatability of starts for tests having similar initial conditions is applicable only for initial starts in the Seattle descent tests. This is because restarts were performed only once or twice at similar coast times, and if repeated at the same coast time were run at different temperatures and degrees of helium saturation.

The initial starts for Phase II can be compared to a nominal engine on the basis of time to ignition or first Pc rise only, since that is the only parameter listed in the SODB, Volume II, Part I (Reference 12). That document indicates that the time to first Pc rise can vary from 0.375 to 3.0 seconds for a "nominal" engine. In Phase II tests, the initial start time to first Pc rise varied from 1.3 to 1.6 seconds, well within the "nominal" limits.

RUN NO.	FIRST FIRING INITIAL ALT. (FT.)	FIRST FIRING UNCHOKE TIME (SEC)	FIRST FIRING TIME TO 200K FT. (SEC.)	SECOND FIRING INITIAL ALT. (FT.)	SECOND FIRING UNCHOKE TIME (SEC.)	SECOND FIRING TIME TO 200K FT. (SEC.)	THIRD FIRING INITIAL ALT. (FT.)	THIRD FIRING UNCHOKE TIME (SEC.)	THIRD FIRING TIME TO 200K FT. (SEC.)	COAST TIME (SEC)	POST RUN COAST (30 min ALT. (FT.)
DO-II	212,000	9-15, (3)	20	-	-	-	-	-	-	-	212,000
D-5	266,000	(2)	11.5	239,000	9-10, (4)	13.5	233,000	(1)	14.5	120	232,000
D-6	208,000	(2)	12	232,000	(2)	15	216,000	(1)	17	90	217,000
D-7	214,000	(2)	13	210,000	8-15, (4)		196,000	9-15, (3)	28	50	210,000
D-8	256,000	(2)	16.5	205,000	9-13, (4)		181,000	6-30	30	15	222,000
D-9	222,000	(2)		132,000	(2)		116,000	8.5-11 (3)	42	5	208,000
D-10	255,000	(2)		121,000	(2)		92,000	7-22, (3)	42	2	245,000
D-11	228,000	(1)	17	232,000	(1)	14	239,000	(1)	10	375	227,000
D-12	222,000	(1)	17	231,000	(1)	16	232,000	(1)	14	170	232,000

(1) Did not unchoke prior to reaching 200,000 ft. altitude
(2) Did not unchoke before the next firing
(3) Remained choked thereafter until above 200,000 ft. altitude
(4) Remained choked thereafter before the next firing.

TABLE C-3 DESCENT ENGINE NOZZLE CHOKING SUMMARY

APPENDIX D - SPS ENGINE TESTING AT AEDC

D.0 GENERAL

This appendix describes the AEDC test facilities and test articles used during the SPS engine restart tests and the SPS injector cold flow tests. The test series and engine instrumentation are described, and an assessment is made of test validity for the SPS engine restarts.

D.1 TEST FACILITY AND TEST APPARATUS (ENGINE RESTART)

D.1.1 AEDC Test Facility

The SPS engine restart tests were conducted at the Rocket Test Facility, Arnold Engineering Development Center, Air Force Systems Command, Arnold Air Force Station, Tennessee. This testing was conducted at a simulated pressure altitude in test cell J-2A.

Test Cell

The J-2A Propulsion Engine Test Cell is a rocket engine test chamber capable of a simulated pressure altitude in excess of 350,000 feet. The test chamber consists of an 18-1/3 foot diameter by 30 feet long, cryogenically cooled liner within a 20 foot diameter basic test chamber. The interior of the liner and all major test chamber components are painted black to increase the absorptivity for thermal radiation.

Hardware Temperature Conditioning

The exteriors of the test hardware were maintained at desired temperatures. while radiating to the test chamber walls, by 12 rows of infrared lamps equally spaced around the test chamber liner.

Exhaust Canister

An exhaust canister was attached to the test engine at the nozzle extension mounting flange and extended to the 1.5 area ratio station in the nozzle. The canister consisted of three pneumatically operated doors which isolated the test engine from the 80,000 feet pressure altitude of the exhaust duct and maintained it in excess of 200,000 feet pressure altitude during ignition, shutdown transients, and coast periods.

Exhaust Gas Diffuser System

The exhaust canister is connected to the J-2A test cell diffuser inlet bulkhead. The diffuser section consists of a 90-in.-diam, LN_2-cooled section 20.5 ft long; a 72-in.-diam, water-cooled section 18 ft long; a hydraulically operated, 72-in.-diam, multiple frangible disc changer; and a hydraulically actuated mechanism containing up to 26 frangible discs 20 mils thick. These discs are used to isolate the test chamber, which can be pumped down to a pressure altitude in excess of 300,000 ft by the cell pumping system, from the Rocket Test Facility (RTF) exhauster system. The exhauster ducting system is maintained at a pressure altitude of 80,000 ft by the RTF plant. The disc is pyrotechnically ruptured on a signal from the engine as the combustion chamber pressure increases to a predetermined level during the ignition transient. After the firing, the exhaust door of the canister is again closed, isolating the engine from the plant exhaust system.

D.1.1 AEDC Test Facility (Continued)

Propellant System

The propellant system (Fig. D-1) consisted of a gaseous-nitrogen (GN_2) pressurizing system, 1000-gal supply tanks, 40-μ filter systems, temperature conditioning and recirculation circuits, bellows-type accumulators and accumulator charging system, and associated valves, pumps, and piping.

The propellants were conditioned by circulating through heat exchangers. Two recirculation circuits were used in each propellant system. The primary circuit allowed propellant recirculation from the supply tanks to near the engine Thrust Chamber Valve (TCV). The secondary circuits allowed recirculation from the supply tanks to near the propellant prevalves. Temperature conditioning of the propellant lines was accomplished using heater tape and radiation lamps.

The oxidizer and fuel supply systems were designed to produce dynamic effects similar to those expected in the Apollo system.

A small parallel propellant system was used to introduce propellants downstream of the TCV during the inbleed portion of the test program as shown in Figure D-1.

D.1.2 Test Article

The test article was a full-scale, flight type Apollo SPS Block I engine (AJ10-137) without a nozzle extension. The engine was mounted horizontally in the test chamber and was attached to a thrust abutment at the engine gimbal mounts. The gimbal linkages were replaced with rigid counterparts. The engine was attached to and exhausted into the canister system used to obtain the required altitude conditions. The SPS engine, supplied by NASA-MSC, was modified to obtain optical coverage of the injector and to permit both high response pressure and temperature probe installations.

Engine

The Apollo SPS engine is a pressure-fed, liquid-propellant rocket engine consisting of a bipropellant thrust chamber valve assembly, an injector, a combustion chamber, a nozzle extension, and a gimbal actuator-ring mount assembly. (The nozzle extension and gimbal actuators were not used during this test program).

The design vacuum performance of the engine is 21,500 lb_f of thrust at a propellant mixture ratio of O/F = 2.0 and a combustion chamber pressure of 100 psia. The engine is designed to be capable of 50 restarts over the design operating life of 750 secs. using hypergolic, storable propellants. Nitrogen tetroxide (N_2O_4) is the oxidizer; AZ-50, a 50-50 weight blend of hydrazine (N_2H_4) and unsymmetrical dimethylhydrazine ($(CH_3)_2N_2H_2$), is the fuel.

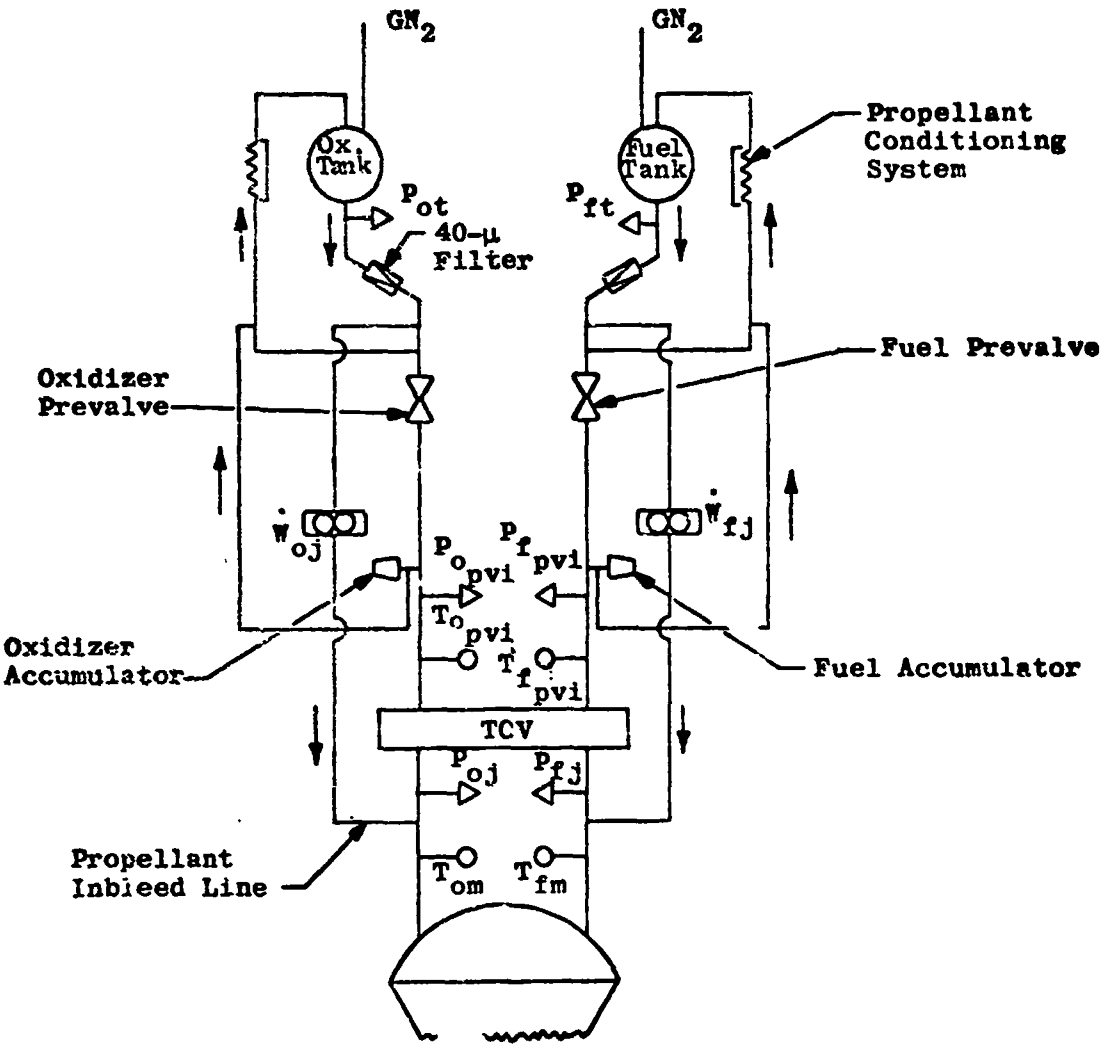


FIGURE D-1

PROPELLANT SYSTEM SCHEMATIC AND INSTRUMENTATION LOCATIONS

D.1.2 Test Article (Continued)

A list of all components used during this test program is presented in Table D-1.

Thrust Chamber Valve

The thrust chamber valve (TCV) consists of a pneumatically operated system of eight ball valves: two in each of two parallel fuel passages and two in each of the two parallel oxidizer passages. One fuel passage and one oxidizer passage constitute an independent valve bank; thus, the TCV has two valve banks, designated valve Banks A and B. Two independent pressure sources operate a pair of actuators in each bank, therefore, the engine can be fired using either Bank A, Bank B, or both. All firings during this test program were conducted using valve Banks A and B together.

Injector

The injector (Block I) used for this test program was a doublet orifice configuration arranged in a concentric ring pattern and was baffled (Fig. D-2) for improved combustion stability. The injector baffles were regeneratively cooled with fuel, which routed through the baffles and back to the injection ring passages. A small portion of the fuel was discharged from each baffle extremity. Film cooling of the combustion chamber was provided by fuel flow from orifices in the extreme outer ring of the injector, adjacent to the injector mounting flange. Approximately 5 percent of the engine fuel flow was injected for film cooling. The injector was modified in order to provide instrumentation taps for high response pressure measurements in the manifold and temperature measurements in the manifold and on the injector face.

Combustion Chamber

The combustion chamber was constructed with an ablative liner, an asbestos insulating liner, and an external wrap. The ablative liner consists of a silica glass fabric tape impregnated with a phenolic resin compound. The chamber was constructed so that the maximum ablative thickness was obtained at the throat section. Several layers of resin-impregnated fiber glass wrap (glass fabric and glass filament) were bonded over the asbestos insulation. The mounting flanges for the injector and nozzle extension were attached to the chamber by bonding the flange lips to the ablative material and overwrapping with fiber glass.

The chamber was modified at the RTF to provide motion-picture coverage of the interior of the chamber during engine operation. This coverage was obtained through two special ports in the chamber wall fitted with quartz windows. The ports were contoured in a manner to minimize any local flow variations within the combustion chamber zone. One port was used to allow chamber interior lighting (1000 watts) by a quartz iodide light, and the other port was used for mounting a fiber optics. The quartz windows were

TABLE D-1
TEST ARTICLE COMPONENTS

Component	Serial Number
Engine Assembly	0000030
Combustion Chamber	0000259
Injector	0000111
Propellant Valve	0000113 (AA through AE)
	0000115 (AG only)

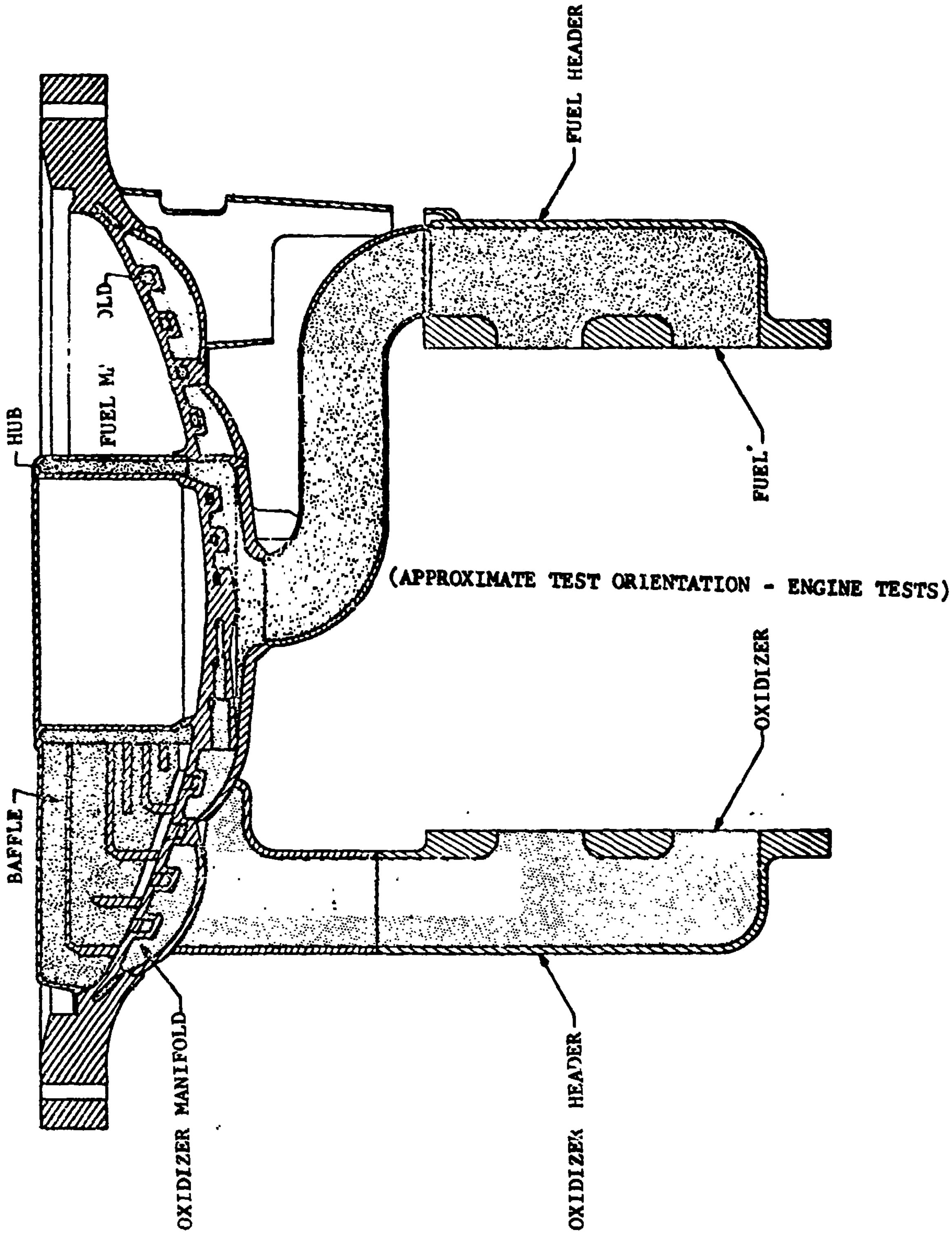


FIGURE D-2

CROSS SECTION SPS ENGINE INJECTOR

D.1.2 Test Article (Continued)

replaced with stainless steel plates for the last test series. Instrumentation ports were drilled to measure temperature and pressure at different axial stations in the chamber.

D.1.3 Instrumentation

Instrumentation systems were provided to obtain measurements of pressures, temperatures, inbleed flow rates, and accelerations of the Apollo SPS engine (Fig. D-3) and its propellant system (Fig. D-1), as well as temperatures and pressures of its environment. The outputs of all of the sensing devices were recorded either on magnetic tape (using the commutating digital or the analog-to-frequency conversion system), light beam oscillograph recorders, or null-balance potentiometers (strip charts). The means of data acquisition are shown in Table D-2 for the various parameters. Visual coverage was also provided by motion picture and television cameras.

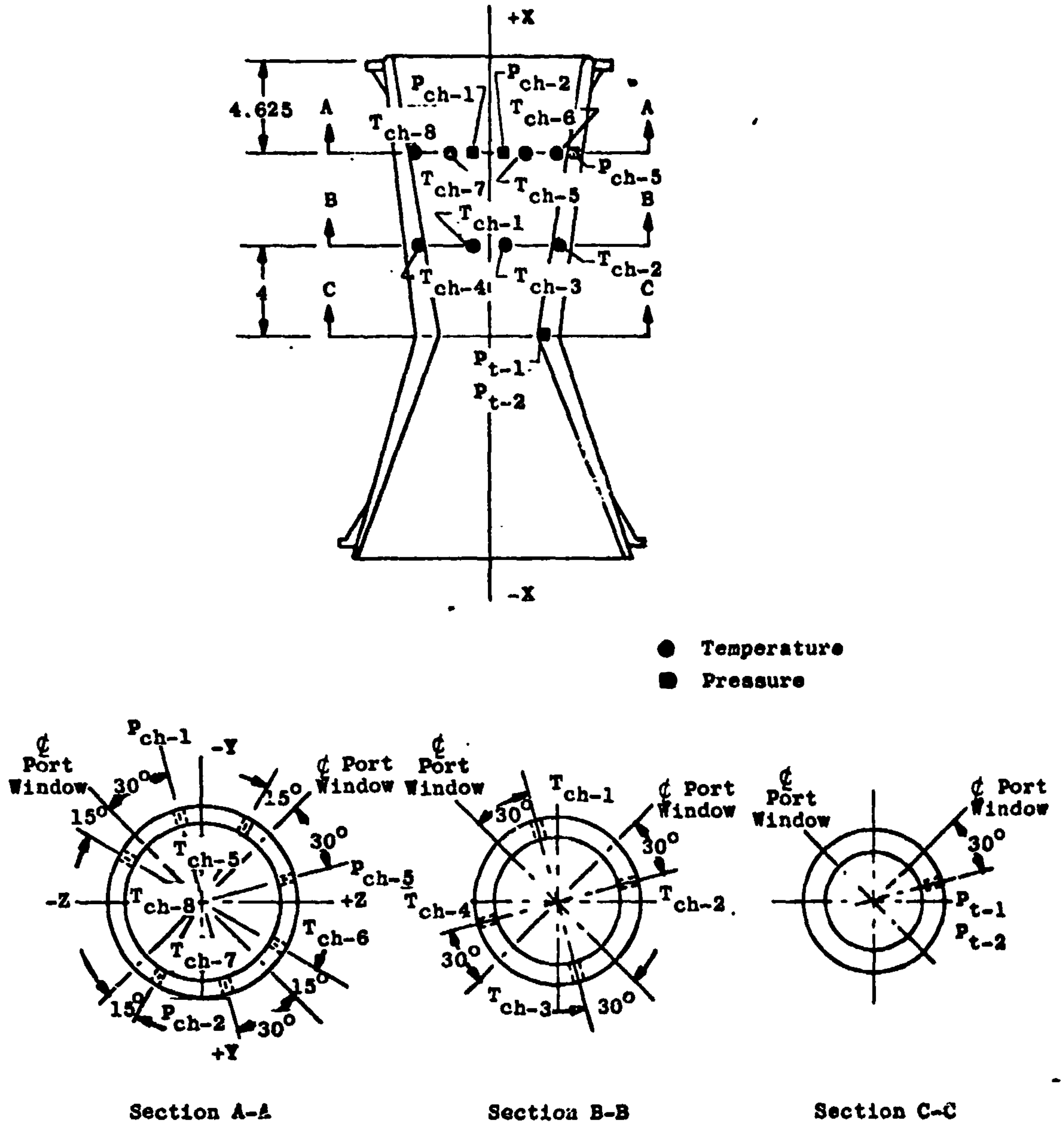
Pressure

Pressure measurements were obtained from the test engine and associated systems with both strain-gage and piezoelectric transducers. The strain-gage-type transducers were used to measure low response and quasi-steady-state pressure. The transducers, mounted on brackets, were connected to the pressure ports with standard 0.25-in. A/N tubing. A close-coupled combustion chamber pressure measurement, through the ablative wall, was made during the AG test series. The ablative chamber was modified for obtaining pressure measurements at both the injector baffle and throat locations. All strain-gage-type transducers were laboratory calibrated before and after the completion of testing, and the calibrations were traceable to the National Bureau of Standards (NBS). The piezoelectric transducers were used to measure high response phenomena. The injector and ablative chamber wall were modified such that the transducers were flush mounted to reduce the response time in the measurement. These transducers and recording devices, in addition to electrical step calibrations, were calibrated in the test cell using a transient step function pressure source with a precision transducer traceable to NBS.

Test chamber pressures were measured with capacitance-type and ionization-type vacuum gages. These transducers and gages were laboratory calibrated with traceability to NBS.

Temperature

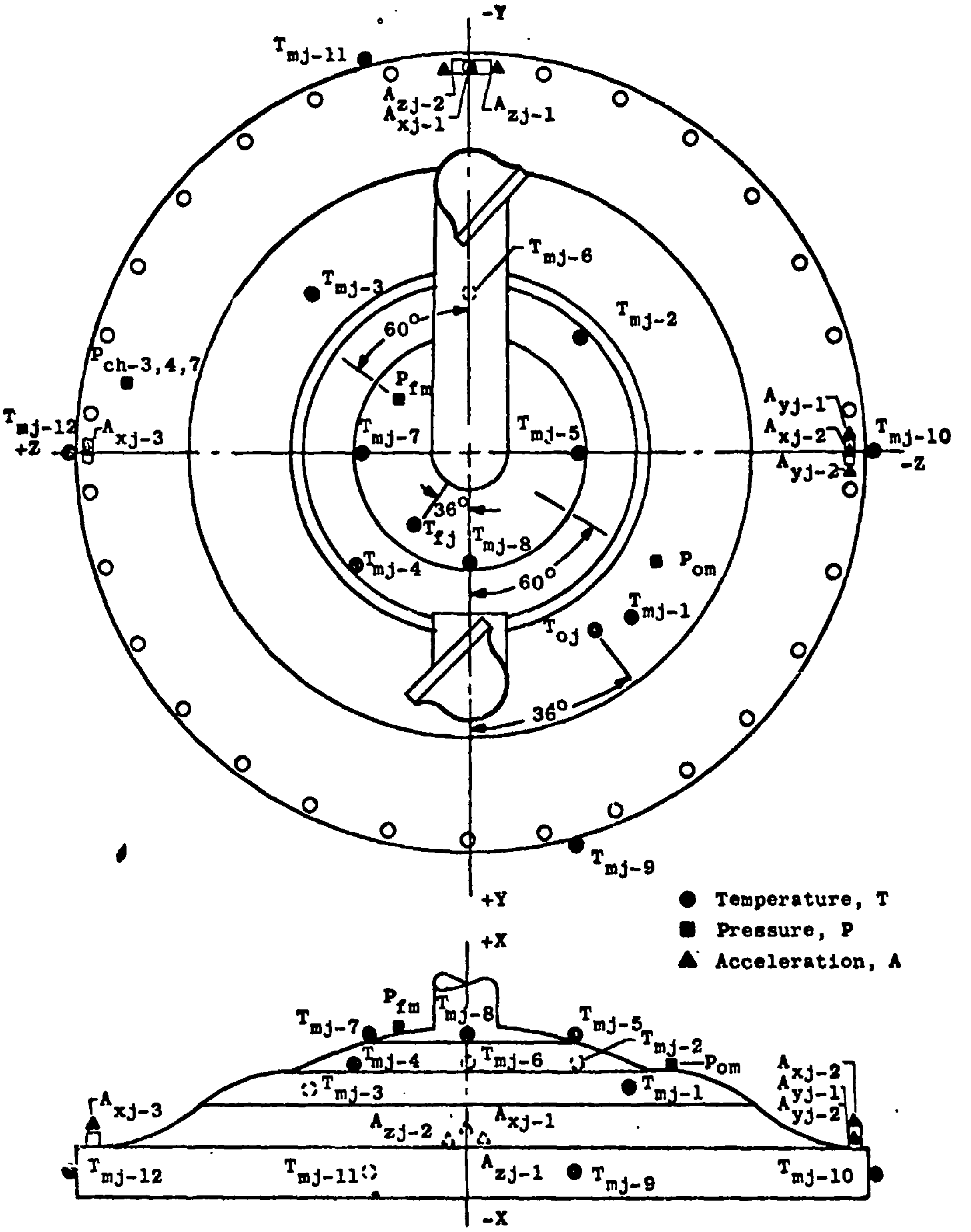
Temperatures were measured using both Chromel -Alumel (CA) thermocouples and resistance temperature transducers (RTT). The RTT's were used to measure propellant and injector manifold temperatures, and the CA thermocouples were used to measure the surface temperatures. Special mountings of thermocouples were made in the injector, on the baffle walls, and on the chamber interior surface to provide high response temperature data during engine operation.



a. Combustion Chamber

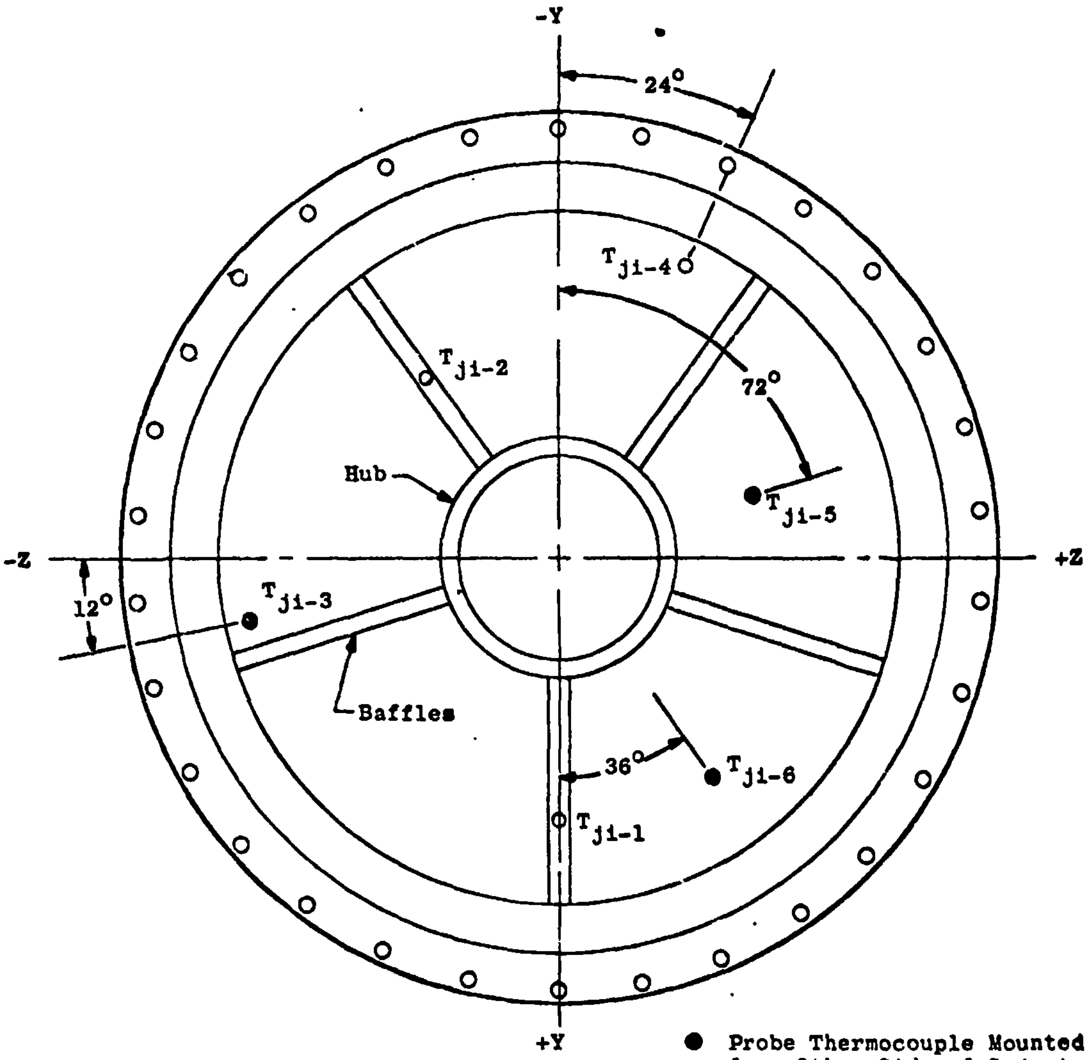
FIGURE D-3

ENGINE INSTRUMENTATION LOCATIONS



b. Injector

FIGURE D-3 (CONTINUED)



● Probe Thermocouple Mounted from Other Side of Injector

○ Surface Thermocouple on Hot Surface

c. Injector Face and Baffle

FIGURE D-3 (CONCLUDED)

TABLE D-2

INSTRUMENTATION LIST

Parameter	Type	Range	Data Acquisition System			
			Magnetic Tape		Strip Chart	Oscillograph
			Analog-to-Digital	Analog-to-Frequency		
Oxidizer Tank Pressure	Strain Gage	0 to 300 psia		X		
Oxidizer Valve Inlet Pressure	Strain Gage	0 to 300 psia		X		X
Oxidizer Manifold Pressure	Piezoelectric	0 to 1000 psia		X		X
Oxidizer Manifold Pressure, Low	Piezoelectric	0 to 200 psia		X		X
Oxidizer Injector Header Pressure No. 1	Strain Gage	0 to 300 psia		X		X
Oxidizer Injector Header Pressure No. 2	Strain Gage	0 to 5 psia	X	X		X
Fuel Tank Pressure	Strain Gage	0 to 300 psia		X		
Fuel Valve Inlet Pressure	Strain Gage	0 to 300 psia		X		X
Fuel Manifold Pressure	Piezoelectric	0 to 1000 psia		X		X
Fuel Manifold Pressure, Low	Piezoelectric	0 to 200 psia		X		X
Fuel Injector Header Pressure No. 1	Strain Gage	0 to 300 psia	X	X		X
Fuel Injector Header Pressure No. 2	Strain Gage	0 to 5 psia	X	X		X
Chamber Throat Pressure No. 1	Strain Gage	0 to 5 psia	X		X	
Chamber Throat Pressure No. 2	Strain Gage	0 to 50 psia	X	X		
Chamber Pressure No. 1, High	Piezoelectric	0 to 1000 psia		X		X
Chamber Pressure No. 1, Low	Piezoelectric	0 to 100 psia		X		X
Chamber Pressure No. 2, High	Piezoelectric	0 to 1000 psia				
Chamber Pressure No. 2, Low	Piezoelectric	0 to 100 psia				
Chamber Pressure No. 3	Strain Gage	0 to 100 psia		X		X
Chamber Pressure No. 4	Strain Gage	0 to 100 psia		X	X	X
Chamber Pressure No. 5	Strain Gage	0 to 100 psia	X		X	
Chamber Pressure No. 7	Strain Gage	0 to 5 psia		X		X
Diffuser Cannister Pressure No. 1	Strain Gage	0 to 1 psia	X	X	X	X
Diffuser Cannister Pressure No. 2	Strain Gage	0 to 50 psia	X		X	X
Cell Pressure No. 1	Ionization	0 to 760 mm Hg			X	
Cell Pressure No. 2	Ionization	0 to 760 mm Hg			X	
Cell Pressure No. 7	Capacitance	0 to 10 mm Hg	X	X		X
Cell Pressure No. 8	Capacitance	0 to 30 mm Hg	X	X		X
Cell Pressure No. 9	Capacitance	0 to 1[illegible] mm Hg	X	X		X
Oxidizer Valve Inlet Temperature	Resistance	0 to 180°F	X			
Oxidizer Manifold Temperature	Resistance	-100 to +100°F	X			
Oxidizer Injector Header Temperature	CA Thermocouple	80 to 370°F	X			
Fuel Valve Inlet Temperature	Resistance	0 to 180°F	X			
Fuel Manifold Temperature	Resistance	-50 to +150°F	X			
Fuel Injector Header Temperature	CA Thermocouple	-80 to 370°F	X			
Propellant Valve Temperature	CA Thermocouple	-80 to 370°F	X			
Combustion Chamber Temperature (Eight)	CA Thermocouple	-100 to 1000°F	X			
Injector Face, Baffle Temperature (Six)	CA Thermocouple	-100 to 1000°F	X			
Injector Exterior Temperature (12)	CA Thermocouple	-150 to +370°F	X			
Accelerometer XX, Injector No. 1	Piezoelectric	0 to 2000 g		X		X
Accelerometer XX, Injector No. 2	Piezoelectric	0 to 4000 g		X		X
Accelerometer XX, Injector No. 3	Piezoelectric	0 to 500 g		X		
Accelerometer YY, Injector No. 1 and 2	Piezoelectric	0 to 2000 g		X		X
Accelerometer ZZ, Injector No. 1 and 2	Piezoelectric	0 to 2000 g		X		X
Oxidizer Flow Rate Injection	Turbine Flowmeter	0 to 5 gpm			X	
Fuel Flow Rate Injection	Turbine Flowmeter	0 to 0.1 gpm			X	

D.1.3 Instrumentation (Continued)

Flow

The flow rates of the propellants inbled into the injector were measured with one flowmeter in each inbleed line. The flowmeters were rotating, permanent magnet, turbine-type, axial flow, volumetric flow sensors with induction coil signal generators. The flowmeters were laboratory calibrated in water with traceability to NBS.

Acceleration

Engine acceleration data were provided using several piezoelectric-type accelerometers in three orthogonal planes. The accelerometers were mounted on the injector with insulated studs inserted into aluminum blocks that had been welded to the injector flange.

Visual

Three motion picture cameras and two closed-circuit television cameras were used to provide visual coverage of the test program (test series AG used only two motion picture cameras). Two of the motion picture cameras and the two television cameras were used to provide exterior coverage of the test hardware. The third motion picture camera was used in conjunction with a fiber optics system to obtain visual coverage of the interior of the chamber during engine operation.

D.2 TEST SERIES DESCRIPTION (ENGINE RESTART)

D.2.1 General

The SPS engine restart tests were conducted over seven "air periods" which included test series AA through AG. These tests were conducted between May and September 1968. All firings were made in the dual (AB) bore valve mode. The test series are broadly categorized in Table D-3.

Series AA - this test series was facility and test specimen checkout. During these tests propellant feed line pressure drops were unacceptable. This series was terminated due to a malfunction of the canister vent doors. Test data were not analyzed.

Series AB - this test series was facility and test specimen checkout and consisted of three firings. Accumulators were installed in the propellant feed lines to eliminate the high line pressure drops. Test data were not analyzed.

Series AC - this test series consisted of five hot firings to evaluate engine restarts. Test duration was 0.37 seconds, and test hardware and propellant were temperature conditioned at 35 and 55°F respectively. Coast periods between engine starts was approximately 22 minutes. Test films showed ice visible on the combustion chamber and injector face during start and shutdown. Testing was terminated when the canister doors malfunctioned.

Series AD - this series consisted of five hot firings to evaluate engine restarts. During the second firing the canister doors malfunctioned, and it was decided to continue the "air period" by conducting three oxidizer inbleed tests. Volumes of N_2O_4 equivalent to 25, 50, and 100 percent of the oxidizer manifold volume were inbled into the injector at a rate to promote propellant freezing. Test durations were 0.37 seconds, and test hardware and propellant were temperature conditioned to 35 and 35 - 42°F respectively. Coast periods were 158 - 1242 minutes.

Series AE - this test series consisted of seventeen hot firings to evaluate engine restarts. One start was conducted at a low (80,000 ft.) pressure altitude. Test duration was 0.37 seconds and test hardware and propellant were temperature conditioned at 35 ±5°F. Coast periods between engine starts were 20 seconds to 462 minutes. During this test series the quartz window in the combustion chamber in which the fiber optics was located was completely blown out, and the window in which the camera light was located was cracked. No other hardware damage was found.

D.2.1 General (Continued)

Series AF - this test series was terminated when the SPS engine bi-propellant ball valve failed to function. Data do not exist for this series.

Series AG - this test series consists of thirty-six hot firings to evaluate engine restarts. Three tests were fuel inbleed tests and two starts were conducted at low (80,000 ft.) pressure altitudes. Test durations were 0.37 - 0.52 seconds and test hardware and propellant were temperature conditioned at 35-40 and 35-65°F respectively. Coast periods between engine starts varied from 7 seconds to 412 minutes.

TABLE D-3

AEDC TEST SUMMARY
SPS RESTART

TEST SERIES	TYPE OF TEST	NO. OF TESTS
AA	Facility Checkout	2
AB	Facility Checkout	3
AC	Restart	5
AD	Oxidizer Inbleed	5
AE	Restart	17
AF	Malfunction - No Data	---
AG	Restart	33
	Fuel Inbleed	3
	TOTAL	68

D.3 TEST RESULTS (ENGINE RESTART)

Review of the test data showed hat all of the test data are valid for restart investigations within the limitations of data acquisition during specific test sequences.

D.3.1 Test Validity and Limitations

The validity of the AEDC SPS engine tests, and the limitations on the applicability of the test data are based on compliance with test requirements, use of Block I hardware, and use of GN_2 as the propellant pressurant.

Although the test data is valid for engine restart investigations there is an apparent lack of data acquired during specific tests. It was not possible to establish an oxidizer lead/lag relationship, based on injector manifold priming times, during any "air period" because either the oxidizer or fuel injector manifold pressure transducers did not function properly. Steady state chamber pressure was not available until the AG test series when a low response strain gage (Taber) transducer was installed.

Although the absolute data appears to be somewhat random for all tests there is a definite correlation of trends for tests having similar initial conditions. Because of the firing duration (0.37 seconds), it is difficult to compare these restart tests to other SPS engine data or to a nominal engine. The data base is also limited when trying to evaluate repeatability for tests having similar initial conditions, since restarts were performed only once or twice at identical coast times.

The test requirements as outlined in Section 6.1.2 were met in all tests, with the following exceptions:

1. Facility Requirements

   The minimum pressure altitude of 180,000 feet was not met in tests AC-03, AD-02, AD-03, AD-04, AD-05, AE-06C, and AG-16.

D.4 TEST FACILITY AND TEST APPARATUS (COLD FLOW)

D.4.1 AEDC Test Facility

The SPS injector cold flow tests conducted at the Aerospace Environment Facility, Arnold Engineering Development Center, Air Force Systems Command, Arnold Air Force Station, Tennessee. This testing was conducted at a simulated pressure altitude in Aerospace Research Chambers ARC (8V) and ARC (7V).

The basic requirement of this test program, rapid evacuation of the SPS injector, was accomplished by installing the injector in a relatively small antechamber (A/C) and attaching it by means of a large duct and valve to either of several large-volume cryogenic pumping chambers available in the Aerospace Environment Facility. As shown in Figure D-4, the injector was mounted inside the A/C adjacent to the pumping or test support chamber. A 10 inch duct with an air operated 10 inch valve connected the A/C to the pumping chamber.

<u>ARC (8V)</u>

The ARC (8V) is a stainless steel vacuum chamber 10 feet in diameter and 20 feet in length containing 600 ft^2 of liquid-nitrogen (LN_2)-cooled cryo-surface and 120 ft^2 of gaseous-helium (GHe)-cooled cryoarray. This chamber is capable of holding a vacuum of 10^{-2} torr through the test.

This chamber provided vacuum pumping through the 10 inch diameter connecting valve. During the oxidizer tests, the liquid oxidizer supply tank was confined within the test support chamber to provide maximum safety conditions in the test area. The complete test support system is diagrammed in Figure D-4.

A toxic vent system was provided for use during chamber repressurization cycles so that the vapors of N_2O_4 could be safely exhausted outside the chamber laboratory building.

<u>ARC (7V)</u>

Phases II and III of the Apollo SPS injector cold flow test, using Aerozine-50 (in Phase II), and Aerozine-50 plus Freon MF, a "simulated" oxidizer (in Phase III), were conducted in the Aerospace Research Chamber (7V).

The ARC (7V) is a stainless steel vacuum chamber 7 feet in diameter and 12 feet in length. In Phase II, 64 ft^2 of the tube-in-sheet paneling served as the LN_2 cryosystem but later, in Phase III, the 480 ft^2 LN_2 chamber liner was required to carry the additional load imposed by the Freon MF in addition to the Aerozine-50.

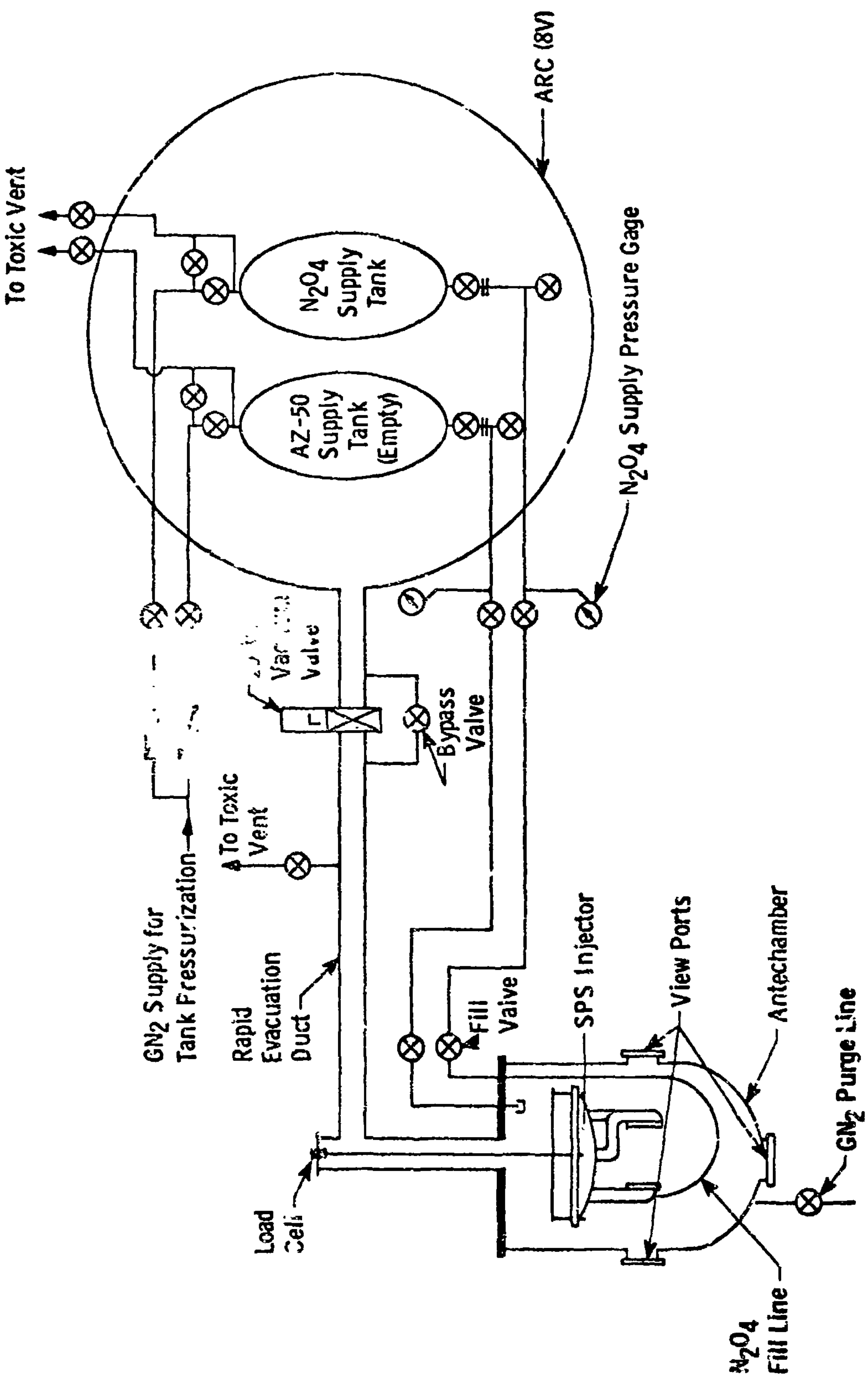


FIGURE D-4
APOLLO SPS INJECTOR COLD FLOW TEST SYSTEM SCHEMATIC

D.4.2 Test Article

General Description

The test article used for these tests was a modified full-scale Block I SPS injector. The modifications were for instrumentation and photographic coverage during the tests.

This aluminum alloy injector was designed and built by the Aerojet-General Corporation. The injector receives propellants through individual manifolds to separate orifice arrays. After passing through these orifice arrays, the propellants are mixed producing hypergolic ignition. In these tests. the injector was oriented with the orifice arrays pointing up.

Modifications

Modifications for temperature and pressure measurement devices and for view ports and light ports consisted of aluminum extensions and attachment flanges welded to the injector and penetrations cut to provide access to the inside of the injector.

D.4.3 Instrumentation

Instrumentation systems were provided to obtain pressure, temperatures, and change in injector weight. The outputs of all of the sensing devices were recorded on magnetic tape. The means of data acquisition, ranges, and channel numbers are shown in Table D-4. Visual coverage was provided by high-and-low-speed motion picture cameras.

Temperature Measurements	Position Number	Type Sensor	Readout Channel Numbers
Injector Skin			
Fuel Duct	1, 2, 3	CATC(1)	UDS(2) 1, 20, 21
Fuel Cover	4, 5, 6, 7		M-1(3)/UDS 22, 23, 24, 25
Oxidizer Manifold (Inboard)	8, 9, 10, 11		UDS 26, 27, 30, 31
Oxidizer Manifold (Outboard)	12, 13, 14, 15		UDS 32, 47, 50, 51
Oxidizer Duct	17, 18, 19		UDS 53, 54, 55
Flange Edge	16		UDS 52
Liquid Temperature Probes			
Fuel Duct	20	PRTP(4)	12, 43, 74, 125
Fuel Manifold	21		13, 44, 75, 126
Oxidizer Duct	22		14, 45, 76, 127
Oxidizer Manifold	23		15, 46, 77, 130
Front Side of Injector			
Injector Face (Center Fuel Hole)	24	CATC	UDS 56
Injector Face (Center Oxidizer Hole)	25		57
Injector Face (Center Baffle Edge)	26		60
Injector Face (Radial Baffle Side)	27		61
Fuel Baffle (Feed into Transducer Fitting)	28	CATC Ceramo Probe	62
Fuel Channel (Most Inboard)	29	CATC	63
Fuel Channel	30		100
Fuel Channel	31		101
Fuel Channel (Most Outboard)	32		102
Oxidizer Channel (Inboard)	33		103
Oxidizer Channel (Outboard)	34		104
Antechamber, Inside Skin	35	CATC	M-2/UDS 105
Antechamber, Inside Skin	36		UDS 106
Antechamber, Inside Skin	37		UDS 107
Antechamber, Inside Skin	38		UDS 110
Antechamber, Inside Gas	39		M-3
Antechamber, Inside Gas	40		111

Pressure Measurements	Position Number	Type Sensor	Pressure Range, psia	Readout Channel Numbers
Fuel Injector Cavity				
Fuel Inlet (High Pressure)	41	Statham 15 psia	0-10	Oscillograph - 1
Fuel Manifold (High Pressure)	42		0-10	Oscillograph - 3/UDS 2, 33, 64, 115
Fuel Inlet (Low Pressure	43	CEC 1 psia	0-0.5	UDS 4, 35, 66, 117
Fuel Manifold (Low Pressure)	44		0-0.5	UDS 5, 36, 67, 120
Oxidizer Injector Cavity				
Oxidizer Inlet (High Pressure)	45	Taber 50 psia	0-30	Oscillograph - 5
Oxidizer Manifold (High Pressure)	46		0-30	Oscillograph - 7/UDS 3, 34, 65, 116
Oxidizer Inlet (Low Pressure)	47	CEC 1 psia	0-0.5	UDS 6, 37, 70, 121
Oxidizer Manifold (Low Pressure)	48		0-0.5	UDS 7, 40, 71, 122
Antechamber, Top Flange	49	Taber 100 psia	0-30	Oscillograph - 11/UDS 11, 42, 73, 124
Antechamber, Top Duct	50	CEC 1 psia	0-0.5	UDS 10, 41, 72, 123
Antechamber, Side	51	CEC 1 psia	0-0.5	Oscillograph - 9
Antechamber, Top Duct	52	CEC 0.5 psia	0-0.1	UDS 17

Force Measurements	Position Number	Type Sensor	Range	Readout Channel Numbers
Injector Weight	53	BLH/load cell 100 lb	0-70 lb	UDS 16

NOTES. (1) Chromel®/Alumel® Thermocouple

(2) Universal Data System Readout Number

(3) Multipoint Readout Number (Redundant Readout in Chamber Control Area)

(4) Platinum Resistance Thermometer Probes

TABLE D-4

APOLLO SPS INJECTOR COLD FLOW INSTRUMENTATION SUMMARY

D.5 TEST SERIES DESCRIPTION (COLD FLOW)

D.5.1 General

The SPS injector cold flow tests were conducted in three phases; phase I - oxidizer, phase II - fuel, and phase III - fuel and simulated oxidizer. These tests were conduc'. from December 15, 1967 to February 2, 1968. The AEDC cold flow tests are broadly categorized as follows:

FLUID	NO. OF TESTS	INITIAL PROPELLANT TEMPERATURE
N_2O_4	11	15-65°F
A-50	10	34-79°F
A-50 and Freon MF	6	40-85°F Fuel 23-80°F Freon MF

Phase I Tests - this phase consisted of eleven oxidizer injector cold flow tests using N_2O_4. A data recording malfunction voided the data for the first test. The remaining tests were conducted with initial propellant temperatures of 16 to 65°F.

Phase II Tests - this phase consisted of ten fuel injector cold flow tests using A-50. A data recording malfunction voided two, three, and four. The remaining tests were conducted with initial propellant temperatures of 35 to 80°F.

Phase III Tests - this phase consisted of six injector cold flow tests using A-50 and Freon MF as an oxidizer simulant. These tests were conducted with initial propellant temperatures of 35 to 80°F for the Freon MF and 35-85°F for the A-50.

D.6 TEST RESULTS (COLD FLOW)

Review of the test data showed that the cold flow test data were valid for determining propellant phenomena in the injector and inlet ducts when exposed to a vacuum. .

The test requirements as outlined in Section 6.3.2 were met in all tests.

APPENDIX E RESTART COLD-FLOW RECOMMENDATIONS

E.0 GENERAL

This appendix provides a recommended approach for future cold-flow test programs which may be conducted to support restart hot-firing tests. The recommendations in this Appendix are based on an evaluation of cold-flow and hot-firing test programs conducted to define the restart characteristics of the Apollo SPS, APS, and DPS engines. Cold-flow test programs were conducted at AEDC on the SPS injector and at ARC on the APS and DPS injectors. Hot-firing tests were conducted at AEDC on the SPS engine and at Boeing/Seattle on the APS and DPS engines. Although the recommendations are based on an evaluation of tests which used N_2O_4/A-50 propellants, the basic test philosophy can be applied to other propellant combinations.

The overall engine restart problem can be separated into three phases, the engine shutdown phase, the coast phase, and the restart phase. The shutdown phase begins when the engine propellant valves begin to close, and ends when combustion ceases. The coast phase begins when the injector manifold pressures decay to the fuel and oxidizer vapor pressures, and ends when the next engine start signal is given. The restart phase begins when the engine start signal is given, and ends when steady-state engine operation has been reached. The boundary between the shutdown and coast phases is not precise, because combustion may continue after one, or both, of the propellants reach their respective vapor pressures. However, these definitions are conceptually straight-forward since the shutdown phase emphasizes combustion tailoff, which the cold flow tests cannot accurately simulate, while the coast phase emphasizes nonreactive propellant behavior, which the cold flow tests can simulate.

The table below defines the major propellant phenomena occurring during the engine shutdown, coast and restart phases.

	TEST PHASE		
	SHUTDOWN	COAST	RESTART
Phenomena	Effervescence of dissolved gases	Propellant boiling	Injector manifold priming
	Propellant drainage	Propellant freezing	
	Beginning of propellant boiling	Propellant sublimation	

The definitions assume that the duration of the first engine firing is short enough to minimize engine heat effects. During engine firings, the thrust chamber and injector are heated by the combustion process.

E.0 GENERAL (Continued)

Heat transfer from the thrust chamber and injector can affect the amount of propellant residuals present as a function of time. Current Apollo engines experience peak heat soak back effects approximately 1/2 to 1 hour after a long duration (mission duty cycle) engine firing. By this time, the residual N_2O_4/A-50 propellants would be largely depleted even if there were no heat soak back. Hence, the definition of a "short duration" engine firing is not precise, and must be determined from temperature histories obtained for varying engine firing durations. The Apollo engine restart tests were conducted with firing durations of .37 to 3.5 seconds. These firing durations were "short" because no significant engine heating effects were observed. For example, frozen oxidizer was observed on the DPS thrust chamber wall within a few seconds after 3.5 second test firings.

E.1 REQUIRED COLD FLOW TEST RESULTS

A cold flow test program can provide basic engineering data so that the effect of operational variables, such as propellant gas content, propellant temperatures, time interval between firing, etc. on engine start characteristics can be predicted. Cold flow tests are also much less expensive than hot firing tests, and therefore can be used to explore a wide range of test variables in the search for potential engine operating problems. Fuel and oxidizer side characteristics can be evaluated separately, and the results can be combined to predict and correlate engine hot firing results. A limited number of hot firing tests can then be conducted at carefully selected test conditions to confirm the presence or absence of undesirable engine operating characteristics.

A specific set of desired test results is provided in the table E-1 Current Apollo engines (APS, DPS, SPS) use the N_2O_4/A-50 propellant combination. Smooth ignition of this combination typically requires an oxidizer lead or a long fuel lead. Nearly simultaneous injection of propellants usually results in large pressure spikes at ignition. Therefore, the desired results emphasize the determination of the fuel and oxidizer lead/lag relationship. Other factors, such as propellant temperature and propellant gas content, are also included because they can affect ignition characteristics. If the cold flow tests are conducted with a new injector or engine design, or a different propellant combination, the required test results can be established only after the injector and/or propellant ignition characteristics have been analyzed.

E.2 EVALUATION OF APS, DPS, AND SPS COLD FLOW TEST PROGRAMS

The APS, DPS, and SPS cold flow programs were evaluated to determine how well they predicted operational problems, or provided additional data to explain the problems observed during subsequent hot firing tests. The advantages and disadvantages of the cold flow tests, with respect to the

TABLE E-1

REQUIRED COLD-FLOW TEST RESULTS

SIGNIFICANT VARIABLES	MANIFOLD PRIMING TIME (FUEL & OXID)	INJECTOR FACE PRIMING TIME (FUEL & OXID)	RESIDUAL VOLUMES (FUEL & OXID)	PROP. HISTORIES PRESS. VS. TEMP. (FUEL & OXID)	INJECTOR PRESSURE DECAY (FUEL & OXID)
COAST TIME	X	X	X		
PROPELLANT TEMPERATURE	X	X	X	X	X
PROPELLANT DISSOLVED GAS CONTENT	X	X	X	X	X
NUMBER OF PROPELLANT FLOW PULSES	X	X	X	X	
INJECTOR ORIENTATION	X	X	X	X	X

E.2 EVALUATION OF APS, DPS, AND SPS COLD FLOW TEST PROGRAMS (Continued)

hot firing tests, were determined and the results used to prepare a recommended test program which will retain the advantages and eliminate the disadvantages of the previous cold flow programs. Table E-2 summarizes the significant advantages and disadvantages of the cold flow test programs. This summary is based on the evaluation of the ARC cold flow test program in Appendix A, and on the evaluation of hot firing tests in Section 7 (Volume 1).

At the time the Apollo cold flow tests were conducted, the basic restart phenomena were not completely understood, and could not be analyzed. As a result, the cold flow tests should have been broad enough in scope and provided a realistic enough simulation to isolate potential problem areas. It is apparent from the summary in Table E-2 and from the evaluation in Appendix A, that these cold flow tests emphasized the coast period between engine starts. The cold flow tests were generally adequate to characterize the coast period, but were not adequate to define the effect of coast phase phenomena on the engine restart characteristics. In the specific case of the LM ascent engine, the cold flow tests were not adequate to predict the oxidizer side problems experienced during the subsequent hot firing tests. In addition, no data on the shutdown phase was available from any of the test programs. The result is that the effect of the cold flow shutdown characteristics on the coast phase phenomena cannot be established.

Cold flow tests can provide a good simulation of the coast phase following short duration engine firings. During the coast phase, chamber pressure effects on boiling, freezing and sublimation are negligible. Short duration firings result in minimum heat soakback with the result that the coast phase phenomena are not significantly affected by the heat soakback. Simulation of the shutdown and restart phases is not as accurate because the combustion chamber pressure history cannot be simulated. During the shutdown phase of an engine hot firing, chamber pressure delays the onset of boiling, and tends to reduce the rate of boiling. During the restart phase of an engine hot firing test, the increase in chamber pressure following ignition results in significant changes in propellant flow rates and pressures in the injector. Important engineering data on the shutdown and restart phases can be obtained even though the cold flow simulation will not be exact. The cold flow data can be used to determine the relative importance of the phenomena even though the magnitude of the effects may not be correct.

E.3 RECOMMEND TEST PROGRAM

Data from a cold flow test program, when combined with basic data on engine operation and propellant ignition characteristics (i.e. the effect of propellant temperature, oxidizer lead/lag, etc.) should provide the basis for a hot firing verification test program. This can

TABLE E-2

COLD FLOW TEST EVALUATION

INJECTOR	ADVANTAGES	DISADVANTAGES
Ascent (Rocketdyne)	Mass residuals measured at several coast times (fuel side)	No oxidizer side tests
		No oxidizer simulant evaluation tests
	Flight configuration ball valves used to control propellant flow	No high response instrumentation
Descent (TRW)	Mass residuals measured at several coast times (fuel side)	No measurement of oxidizer residuals
		No high response instrumentation
	Flight configuration ball valves used to control propellant flow	Altitude chamber pumping capacity not adequate
SPS (Aerojet)	High speed photographic coverage of internal injector and duct flow paths	Static-type test, no engine propellant valves used.

E.3 RECOMMENDED TEST PROGRAM (Continued)

be accomplished if the test article, test facility, instrumentation, and test matrix are designed and selected to provide the required data. The following is a presentation of the essential requirements for a restart cold flow test program.

TEST SEQUENCE

TEST PHASE	REQUIREMENT	EXPECTED RESULTS OR RATIONALE
COAST	LONG DURATION COAST (1 HOUR MINIMUM) FUEL AND OXIDIZER TESTS NOMINAL, COLD AND WARM PROPELLANT TEMPERATURES	DETERMINE ONSET OF BOILING AND FREEZING CALCULATE MASS RESIDUALS DETERMINE RECOMMENDED COAST PERIODS FOR REMAINING TESTS
COAST	EVALUATION OF OXIDIZER SIMULANT OXIDIZER FLOW SIMULANT FLOW FUEL (ONLY) FLOW FUEL AND OXIZIDER SIMULANT FLOW	DETERMINE EFFECTIVENESS OF OXIDIZER SIMULANT DETERMINE EFFECT OF OXIDIZER COOLING ON FUEL SIDE, AND FUEL COOLING ON OXIDIZER SIDE OF INJECTOR
COAST	CALIBRATION OF PROPELLANT RESIDUAL MEASUREMENT APPARATUS AND TECHNIQUES (FUEL AND OXIDIZER)	DETERMINE ACCURACY AND REPEATABILITY OF RESIDUAL DISTILLATION APPARATUS AND TECHNIQUES
SHUTDOWN, COAST	SINGLE-PULSE TESTS, MEASURE RESIDUALS NOMINAL, COLD, AND WARM PROPELLANT TEMPERATURES 0% AND 100% GAS SATURATION (HE. N_2) VARY ORIENTATION (UP, DOWN, HORIZONTAL)	DETERMINE EFFECT OF TEST VARIABLES ON VOLUME AND THERMODYNAMIC STATE OF PROPELLANT RESIDUALS EVALUATE RELATIVE IMPORTANCE OF GRAVITY EFFECTS AND PROPELLANT PHENOMENA (BOILING ETC.) FOR ENGINEERING PREDICTION OF "0-g" FLIGHT CONDITIONS
RESTART	DUAL-PULSE TESTS (FUEL AND OXIDIZER) NOMINAL, COLD, AND WARM PROPELLANT TEMPERATURES	DETERMINE MANIFOLD AND INJECTOR PRIMING TIMES AND PRIMING CHARACTERISTICS AS AFFECTED BY VOLUME AND THERMODYNAMIC STATE OF PROPELLANT RESIDUALS

TEST SEQUENCE (Continued)

TEST PHASE	REQUIREMENT	EXPECTED RESULTS OF RATIONALE
RESTART (Continued)	0% AND 100% GAS SATURATION (HE, N_2) VARY INJECTOR ORIENTATION (UP, DOWN, HORIZONTAL)	
RESTART	MULTIPLE-PULSE TESTS (3-5 FLOW PULSES, FUEL AND OXIDIZER) RUN AT TEST CONDITIONS WHICH MAXIMIZE AMOUNT OF FROZEN PROPELLANT RESIDUALS	EVALUATE "WORST-CASE" RESTART CONDITIONS

TEST ARTICLE CONFIGURATION

TEST PHASE	REQUIREMENT	EXPECTED RESULTS OR RATIONALE
ALL	INJECTOR CONFIGURATION IDENTICAL TO CONFIGURATION PLANNED FOR HOT FIRING TESTS	ELIMINATE, WHEREVER POSSIBLE, THE REQUIREMENT TO PREDICT THE EFFECT OF CONFIGURATION CHANGES ON ENGINE OPERATIONAL CHARACTERISTICS.
SHUTDOWN, RESTART	UTILIZE FLIGHT TYPE PROPELLANT VALVES AND ACTUATION SYSTEM TO CONTROL PROPELLANT FLOW	PROVIDE ACCURATE SIMULATION OF PROPELLANT TRANSIENT FLOW CHARACTERISTICS.
ALL	PROVIDE SIMULATED COMBUSTION CHAMBER AND THROAT	PROVIDE BACK PRESSURE AT INJECTOR FACE. SIMULATION WILL NOT BE ACCURATE DURING SHUTDOWN, BUT WILL BE REASONABLE DURING COAST AND DURING THE MANIFOLD PRIMING IN THE RESTART PHASE.

TEST FACILITY

TEST PHASE	REQUIREMENT	EXPECTED RESULTS OR RATIONALE
COAST	DISTILL RESIDUALS (FUEL AND OXIDIZER)	MEASURED MASS RESIDUALS AS FUNCTION OF TEST CONDITIONS (COAST TIME, PROPELLANT TEMPERATURE, PROPELLANT GAS CONTENT, ETC.)
SHUTDOWN, COAST	CONTROL AND MEASURE PROPELLANT GAS CONTENT (He, N_2, ETC.)	DETERMINE EFFECT OF GAS CONTENT ON VOLUME AND THERMODYNAMIC STATE OF RESIDUAL FUEL AND OXIDIZER
SHUTDOWN, COAST, RESTART	PROVIDE CAPABILITY FOR CHANGING INJECTOR ORIENTATION (UP, DOWN, HORIZONTAL)	DETERMINE RELATIVE IMPORTANCE OF GRAVITY FORCES (PROPELLANT DRAINAGE, RETENTION OF RESIDUALS) AND PROPELLANT PHENOMENA (BOILING, GAS EFFERVESCENCE, SUBLIMATION) ON VOLUME AND THERMODYNAMIC STATE OF RESIDUAL FUEL AND OXIDIZER
SHUTDOWN, COAST, RESTART	ALTITUDE CHAMBER PUMPING CAPACITY ADEQUATE TO MAINTAIN CHOKED FLOW IN THRUST CHAMBER NOZZLE UNTIL PRESSURE IN ALTITUDE CHAMBER DECREASES BELOW FUEL (HYDRAZINE) AND OXIDIZER TRIPLE POINT PRESSURES	CHOKED NOZZLE FLOW WILL ISOLATE INJECTOR FROM THE ALTITUDE CHAMBER, ALLOWING VALID SIMULATION OF SPACE CONDITIONS. ULTIMATE TEST CHAMBER PRESSURE MUST BE BELOW TRIPLE POINT PRESSURES TO PRODUCE PROPELLANT FREEZING AND SUBLIMATION

INSTRUMENTATION

TEST PHASE	REQUIREMENT	EXPECTED RESULTS OR RATIONALE
RESTART	HIGH FREQUENCY RESPONSE (1000 CPS) CLOSE-COUPLED PRESSURE TRANSDUCERS: ENGINE INTERFACE, FUEL AND OXIDIZER INJECTOR MANIFOLDS, FUEL AND OXIDIZER	DETERMINE MANIFOLD PRIMING TESTS DETERMINE IF FROZEN FUEL OR OXIDIZER HAS BLOCKED INJECTOR AND/OR DUCT FLOW PASSAGES
SHUTDOWN	HIGH FREQUENCY RESPONSE (1000 CPS) CLOSE-COUPLED PRESSURE TRANSDUCERS ON FUEL AND OXIDIZER INJECTOR MANIFOLDS	DETERMINE INJECTOR SHUTDOWN CHARACTERISTICS WITH NO COMBUSTION (COMPARE TO ENGINE HOT FIRING CHARACTERISTICS)
COAST PHASE	LOW RANGE INJECTOR MANIFOLD PRESSURE TRANSDUCERS, WITH A RANGE OF ZERO PSIA TO PROPELLANT VAPOR PRESSURE (0-5 PSIA, FUEL; 0-15 PSIA OXIDIZER)	DETERMINE THERMODYNAMIC STATE OF PROPELLANT RESIDUALS, WHEN CORRELATED WITH TEMPERATURE MEASUREMENTS CALCULATE MASS RESIDUALS AS A FUNCTION OF COAST TIME. (COMPARE TO MEASURED MASS RESIDUAL DATA)
	THERMOCOUPLES, SURFACE AND IMMERSION, LOCATED WHERE PROPELLANT RESIDUALS CAN BE TRAPPED IN THE INJECTOR, DUCTS, AND BETWEEN VALVES	DETERMINE THERMODYNAMIC STATE OF PROPELLANT RESIDUALS, WHEN CORRELATED WITH PRESSURE MEASUREMENTS DETERMINE CORRELATION BETWEEN SURFACE AND IMMERSION THERMOCOUPLES FOR APPLICATION TO HOT FIRING TEMPERATURE DATA. (IMMERSION THERMOCOUPLES MAY NOT BE USED DURING HOT FIRING TESTS)

INSTRUMENTATION (Continued)

TEST PHASE	REQUIREMENT*	EXPECTED RESULTS OR RATIONALE
	THERMOCOUPLES ON THE INJECTOR AND VALVES	DETERMINE MAGNITUDE OF EVAPORATIVE COOLING EFFECTS
RESTART, SHUTDOWN	HIGH SPEED (1000 FRAMES/SECOND) MOVIES OF THE INJECTOR FACE, WITH EVENT LIGHTS MARKING "FIRE SIGNAL" AND "SHUTDOWN SIGNAL"	DETERMINE FUEL AND OXIDIZER PRIMING TIMES AT THE INJECTOR FACE, AND DETERMINE IF INJECTOR FLOW PATHS ARE BLOCKED BY FROZEN PROPELLANT RESIDUALS
		DETERMINE INJECTOR SHUTDOWN CHARACTERISTICS WHEN CORRELATED WITH HIGH RESPONSE PRESSURE MEASUREMENTS
		DETERMINE IF INJECTOR ORIFICES ARE BLOCKED BY FROZEN PROPELLANT RESIDUALS DURING ENGINE SHUTDOWN
SHUTDOWN, COAST	MOVIES OF INTERNAL FLOW PATHS (DUCTS, INJECTOR MANIFOLDS)	DETERMINE ONSET OF BOILING AND FREEZING, CORRELATE WITH PRESSURE-TEMPERATURE DATA