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NAVAL POSTGRADUATE SCHOOL
Monterey, California



THESIS

AN INVESTIGATION INTO THE
PERFORMANCE CHARACTERISTICS OF A
SOLID FUEL SCRAMJET PROPULSION DEVICE

by

William J. Angus

December 1991

Thesis Advisor
Second Reader

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An Investigation into the
Performance Characteristics of a
Solid Fuel Scramjet Propulsion Device

by

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ABSTRACT

An investigation was conducted to evaluate the performance qualities of a supersonic ramjet propulsion device (SCRAMJET) using a solid fuel. The fuel grains were fabricated from Plexiglas and were cylindrical, with an axisymmetric, circular perforation that diverged in the downstream direction. A small amount of hydrogen gas was required in an initial recirculation zone in order to sustain combustion. With combustor inlet conditions of 150 psia, 1000°R, and a Mach number of 1.5, a combustor exit Mach number of approximately 1.4 was maintained. Due to poor mixing conditions, the combustion efficiency of the solid fuel was only 57%.

0.1

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NOMENCLATURE:

A - Area

C_D - Nozzle discharge coefficient

D_P - Diameter of combustor entry port

D_R - Diameter of recirculation zone

F - Thrust

g_c - constant 32.174 ft-lbm/lbf/sec²

K - Constant (Equation 1)

L - Overall length of fuel grain

L_A - Length of recirculation zone (axially)

L_C - Length of cylindrical combustion zone

M - Mach Number

m - mass flow rate

m - Molecular Weight

p - static pressure

p_t - stagnation pressure

R - Gas Constant

T_t - stagnation temperature

Z - Mass Flow Rate Correction due to gas compressibility

ϵ - Nozzle Expansion Ratio (A_E/A_t)

γ - Ratio of Specific Heats

η - Efficiency

θ - Half angle of divergence

Subscripts:

a - atmospheric, ambient

d - downstream
e - exit
F - Thrust
g - gage
HF - Heater Fuel
HO - Heater Oxidizer
i - initial
m - Measured
P - port
PF - Pilot Fuel
R - recirculation
 ΔT - Heat Addition
t - total (stagnation)
th - Theoretical
T - Throat
u - upstream

I. INTRODUCTION

Although a scramjet propulsive device has never been employed, interest in the concept has developed in light of the perceived need for hypersonic travel and "routine access to space" such as envisioned in the National Aerospace Plane (NASP).

In this scenario, air breathing propulsion is clearly required to achieve the long thrust duration needed to achieve orbit and keep the acceleration to a level that man can survive. The idea that has been chosen involves a "combined cycle" propulsion system, which integrates turbojet, ramjet and scramjet propulsion systems.

Conventional turbojet and ramjet propulsion are well understood and have been fielded in both manned flight and in tactical weapon applications. A vehicle that is powered by the ramjet is velocity limited, however, because in the combustion process, the flow is brought down to a subsonic velocity to keep the residence time of the combustion species within the combustor great enough to complete burning and also to efficiently produce thrust with high stagnation pressure. Stagnation temperature of the inlet air will reach upwards of 40,000°F if a vehicle such as NASP is to achieve orbital velocity. To bring this air to subsonic velocity (in the case

of the ramjet) implies a static temperature within the combustor of 35,000°F!

Materials for building a ramjet device that could withstand such temperatures are not known of today. Indeed, current materials technology together with dissociation of the combustion products limits a ramjet powered vehicle to velocities of Mach seven or eight. The velocity necessary to achieve earth orbit is upwards of Mach 25. Hence, the flow through the combustion device must be allowed to pass at supersonic velocities to avoid this problem with high static temperature.

From this situation springs forth the idea of a supersonic combustion process, referred to as scramjet. Most effort in this area is now focused on a liquid hydrogen fueled scramjet to power hypersonic vehicles such as NASP.

This idea of supersonic combustion and very high Mach number flight could be used in other applications besides manned flight. Weapon delivery at extremely high velocities and from much greater range could be achieved with some sort of a scramjet device. However, some consideration must be made for safety and handling in a weapon application. The higher performance liquid fuels would probably be substituted by a more stable, perhaps inherently inert solid fuel. Herein lies the idea of a solid fueled scramjet.

Witt [Ref. 1] initiated an investigation in which sustained supersonic combustion with solid fuel was

demonstrated. However, no attempt was made to determine the obtainable performance or the magnitudes of the losses. More recently, a simple one dimensional test apparatus has been used to study solid fuel combustion under supersonic crossflow conditions [Ref 2]. In addition, these combustion conditions have been investigated numerically [Ref 3].

The goal of this thesis was to continue the work of Witt [Ref 1] in order to investigate the performance characteristics of the solid fuel scramjet (SFSJ), specifically:

1. Combustion efficiency
2. Fuel regression rate
3. Flameholder (or recirculation zone) geometry optimization

One of the objectives was to alleviate the shockdown of the supersonic flow in the combustor which can occur due to friction and heat addition effects. This was apparently experienced in the initial work by Witt. To counter these effects, the supersonic flow must be accelerated through the combustor by increasing the area of it's passage. Ideally, the flow acceleration due to the divergence and the deceleration due to the heat addition and friction would cancel one another and the exit Mach number of the flow would be approximately the same as when it entered the combustor. However, obtaining this balance is quite difficult since the rate of heat release depends strongly upon the mixing process

between the boundary layer produced fuel pyrolysis products and the oxygen within the central core of air flow moving at supersonic velocity.

II. DESCRIPTION OF APPARATUS AND TEST FACILITY

A. AIR SUPPLY

A high volume, high pressure bank of air bottles, a hydrogen fired, vitiated air heater and a converging-diverging nozzle (conical, $\epsilon=1.17$, $\theta=1.20^\circ$) were used to create the high

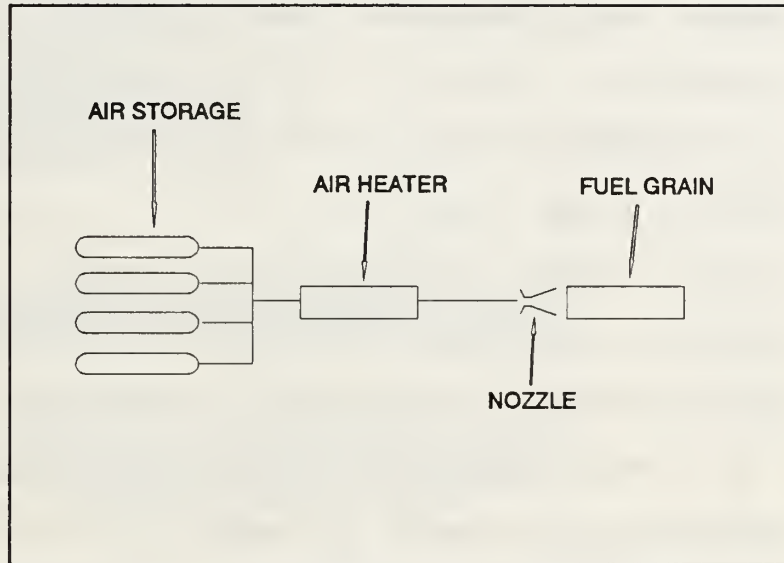


Figure 1. Diagram of Test Facility

stagnation enthalpy, supersonic airflow to simulate air entering a vehicle's inlet at flight velocity (Figure 1).

B. FUEL GRAIN

The fuel grain was an axisymmetric, axially perforated, cylindrical piece of Plexiglas. Plexiglas was chosen due to its reasonably high heat of combustion, ease of machining and of course, its availability. It consisted of a flameholding recirculation zone and a primary combustor zone. The recirculation zone was created by a rearward facing step, followed by a forward facing step. This recirculation zone provided a mixed supersonic/subsonic flow region where flame

stabilization could be achieved. Downstream of the recirculation zone began the primary combustion zone. In general, the grains had the design shown in

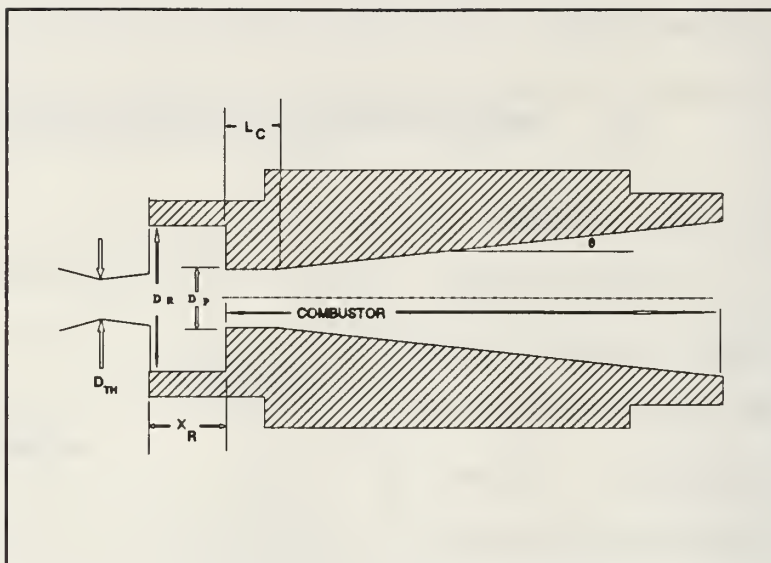


Figure 2. Specific grain dimensions are listed in Table 1.

A small amount of gaseous hydrogen was bled into the recirculation zone at the head end of the fuel grain in order to maintain flame stability. A rake traverse was mounted at the aft-end of the fuel grain (not attached to the thrust stand) where water-cooled pressure and thermocouple probes could be arranged to measure stagnation pressure and temperature profiles.

Ignition was achieved with an ethylene/oxygen fueled torch that was fired for a two second duration into the recirculation zone.

The forward facing step was used to reduce the flow area of the primary combustor entrance. This was required to keep the supersonic inlet airstream velocity low in order to permit sufficient residence time to mix and burn the fuel vapor within the motor. The Mach number at the entrance to

the flame stabilization region was 1.50. Inlet air temperature was approximately 1000°R.

Seven grains were fabricated and burned over the course of this experiment (See Table I). One grain, (#7) utilized a step-change geometry instead of that shown in Figure 2 (See Figure 3).

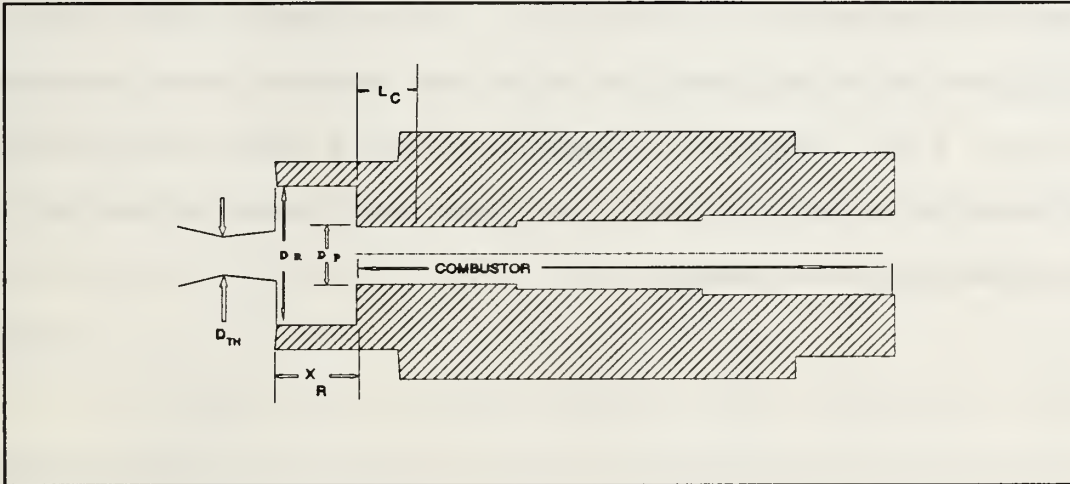


Figure 3 Step Divergent Grain

Table I SUMMARY OF GRAIN GEOMETRIES (inches)

#	D_R	D_P	L_C	X_R	L	θ (°)	D_{th}
1	2.0	0.68	0	1.2	11.0	4.0	0.504
2	2.0	0.60	2.0	1.5	11.0	2.67	0.504
3	2.0	0.70	2.0	1.13	9.0	2.67	0.504
4	1.5	0.57	10.5	1.5	12.0	0.0	0.504
5	1.5	0.57	10.5	1.5	12.0	0.0	0.504
6	1.5	0.59	3.0	1.5	11.5	1.5	0.504
7	1.5	0.59	4.0	0.75	11.5	N/A	0.504

C. DATA ACQUISITION AND TEST CONTROL

Various temperature and pressure transducers and a thrust load cell were used to determine the performance, and to accurately compute flow rates of all gases delivered to the thrust stand. The data was gathered using a Kaye Instruments Modular Data Acquisition System Model 7000 (MDAS) at a sampling interval of 0.05 sec. From there, the data was displayed on the screen of a PC and printed to a hard copy printer for analysis. Control of the test sequence was provided by a Hewlett-Packard 9836S computer together with an HP 3054A control system.

III. EXPERIMENTAL PROCEDURE AND TEST MATRIX

A. PRE-FIRING PROCEDURES

Before the firing event, all pressure transducers were calibrated. The slope of the pressure vs. voltage plots, voltage at p_g equals zero, and the ambient atmospheric pressure were entered into the MDAS software so that all pressures could be directly displayed in psia.

Chromel/Alumel thermocouples with electronic ice points were used and within MDAS, the temperatures were automatically converted to °R.

All gases were passed through sonic chokes and their upstream stagnation pressures and temperatures were measured. Using the continuity equation for choked flow (Equation 1), the mass flow rates were evaluated.

$$\dot{m} = K \frac{P_t A_t}{\sqrt{T_t}} C_D \frac{1}{Z} \quad (1)$$

The fuel grain was weighed before and after the test. The weight loss and recorded burn time were used to calculate the average fuel mass flow rate.

B. TEST MATRIX

Table II lists gas flow rates for all tests conducted as part of this thesis.

Table II MASS FLOW RATES ($\times 10^4$ lbm/sec)

GRAIN #	RUN	m_{air}	m_{HF}	m_{HO}	m_{PF}	m_{PMM}
1	1	3360	8.6	63	3.4	176
1	2	3360	8.6	66	7.4	176
2	3	3370	8.5	73	6.5	76
2	6	3360	7.8	74	7.7	76
3	5	3400	11.0	110	8.3	120
3	6	3330	11.0	110	10.0	120
4	7	4590	18.0	145	3.7	314
4	8	4610	18.0	146	3.7	314
6	9	4620	19.0	150	6.8	218
7	10	4640	19.0	150	7.7	148

C. MOTOR FIRING TEST SEQUENCE

All the valves on the SFSJ test stand were electromechanically or electropneumatically actuated. Their operation was controlled remotely using the HP 9836S/3054A system. The correct sequencing of these valves was provided by a computer program that was tailored for each experiment. The sequence proceeded in the following manner:

1. Airflow was initiated and directed through an overboard dump.
2. The air heater was ignited and allowed to reach a steady state exit temperature of 1100°R.
3. The airflow was directed through the motor for four seconds to permit the inlet air temperature to stabilize to approximately 1000°R and to warm the fuel to "flight conditions".
4. The pilot hydrogen flow was initiated and ignited.
5. After a predetermined interval (approximately three seconds), the rake traverse was actuated, bringing the watercooled thermocouple and pitot probes across the motor exit plane.
6. The motor was burned for approximately seven seconds.
7. The pilot hydrogen valve was closed.
8. The airflow was again directed through the overboard dump.
9. Nitrogen was dumped into the motor to purge reactants and halt the reaction.
10. Airflow was maintained until the air heater was cool.
11. All gases were turned off at the bottle sources.
12. The recorded data was produced in tabular form at the PC and was saved to the hard drive.

IV. RESULTS

The data obtained during this investigation are included in the Appendix.

A. PRESSURE VS. TIME TRACES

Static pressure taps were drilled in the motor grains and extended from the exterior surface of the fuel grain radially into the center perforation. During the run, the pressure at these stations were sampled and recorded at a frequency of 20 Hz. This data was used for off-line analysis of shock position, flow separation from the fuel surface, and pressure gradients within the motor.

The pressure displayed within the motor grains typically had a distribution that began high at the motor inlet and dropped off in the downstream direction. Motor pressure at each tap point fell off at a rate of approximately one psi per second during the actual burn interval. This can be explained by the decrease in static pressure in an accelerating isentropic flow, the acceleration resulting from the opening of the grain perforation as the surface regressed, forcing the supersonic flow to accelerate.

B. EXIT PLANE DATA

The motor exit stagnation pressure profiles typically revealed higher pressure at the outer region of the flow than in the center of the flow.

The stagnation temperature profiles typically displayed the same shape as the stagnation pressure. It was found, however, that the thermocouple in this probe did not have sufficient response time to yield useable data for analysis.

C. FUEL SURFACE REGRESSION

Fuel surface regression was analyzed both qualitatively and quantitatively after each run. This was done to evaluate the uniformity of the combustion and consequently the fuel regression rate, as well as to determine the average mass flow rate of the Plexiglas during the run.

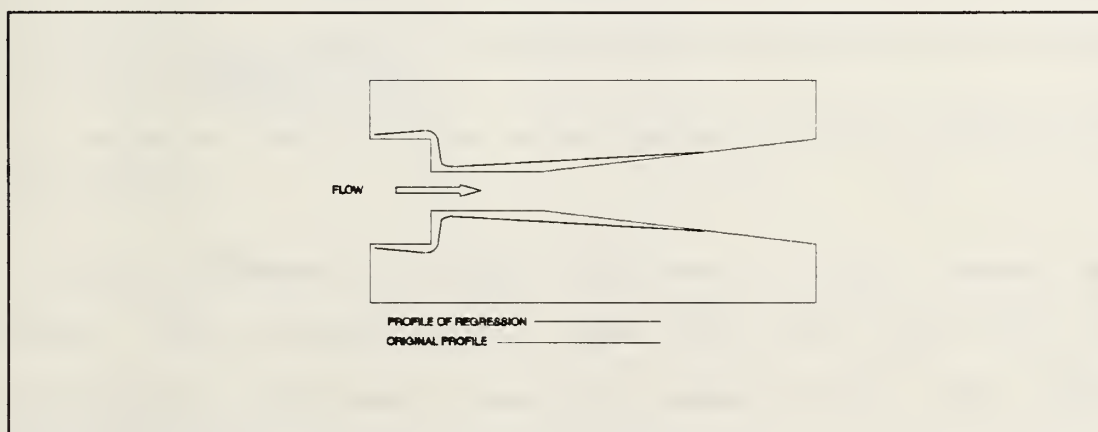


Figure 4 Typical Regression Profile

1. Divergent Combustors

The fuel grains with divergent combustors all exhibited a regression rate that diminished with distance travelled downstream. All had nearly zero fuel consumption at the exit plane and in the immediate area upstream of the exit plane. Figure 4 illustrates the typical regression profiles exhibited in these grains.

2. Non-Diverging Combustors

These grains had fairly uniform regression through their entire lengths. Post run inspection of the downstream ends revealed a rough surface that showed no particular pattern or symmetry that would be indicative of phenomena arising from supersonic flow.

V. DISCUSSION OF RESULTS

The testing proceeded in a logical sequence; each subsequent test attempted to correct a shortcoming (shockdown, poor ignition, etc.) experienced in the previous test. A detailed analysis was performed only for the final run (#12) of the matrix. The data used in the following discussion is contained in the Appendix. The goal was to evaluate exit Mach number and the combustion efficiency of the Plexiglas fuel.

The Mach number at the exit plane was evaluated by using the measured values of thrust, mass flow rate, and the isentropic relations at the exit plane of the combustor.

The theoretical chemical equilibrium composition and temperature of the combustion products were evaluated using the MICROPEP computer code [Ref. 4].

To bracket the possible theoretical performance, two distinct situations were analyzed:

1. All the pyrolyzed Plexiglas mass burned prior to exiting from the motor.
2. None of the Plexiglas mass burned prior to exiting from the motor.

It was assumed that all hydrogen, in both cases, was burned. This established an upper and lower limit for the theoretically obtainable reaction temperature. The inputs into MICROPEP were:

1. The mass flow rates of all the gases introduced into the motor (air, heater fuel, heater oxidizer, and pilot hydrogen).
2. The Plexiglas mass flow rate, using the average value computed over the course of the run.
3. The average pressure at which the combustion took place, as obtained from the pressure/time plots.

For the case where all the Plexiglas burned:

$$\gamma \cong 1.3, T_i = 2394^\circ\text{R}, m = 28.2 \Rightarrow R = 54.8 \text{ ft-lbf/lbm-}^\circ\text{R}.$$

For the case where no Plexiglas burned:

$$\gamma = 1.4, T_i = 1317^\circ\text{R}, m = 28.1 \Rightarrow R = 55.0 \text{ ft-lbf/lbm-}^\circ\text{R}.$$

A. EXIT MACH NUMBER

The initial task was to determine whether or not the exit Mach number was greater or less than one, e.g. whether or not the flow remained supersonic within the combustor.

1. Exit Mach Number from Pressure Ratio at Exit Plane

The pitot probe static pressure transducer recorded gage pressure and, therefore, could only accurately measure pressures greater than atmospheric. Pressures less than atmospheric were detected by the transducer (negative

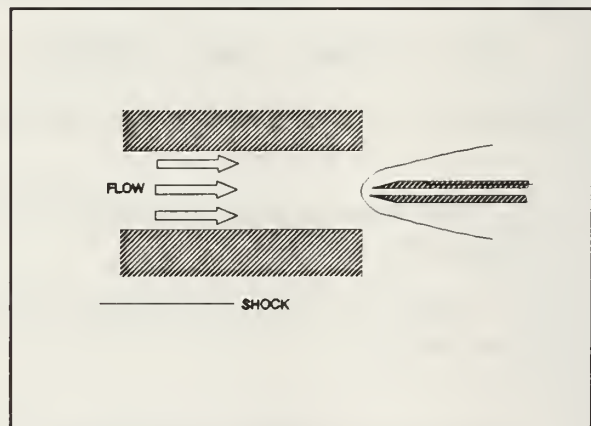


Figure 5 Shock Formation on Probe

voltage), but the accuracy was questionable. Static pressures

within the flow at the grain exit could be greater or less than atmospheric, depending upon the presence of oblique shocks or expansion waves. Assuming that the exit flow was subsonic, the ratio of p_i/p would be less than 1.8 (from isentropic tables, $\gamma = 1.3$). The average probe stagnation pressure was 47 psia, from the probe: $p_w/p_e \geq 47/14.7 = 3.19$. Thus, the flow at the grain exit could not have been subsonic.

2. Exit Mach Number from Thrust Measurement

The exit plane Mach number can also be estimated using the thrust equation. Since the inlet air was injected perpendicular to the thrust direction,

$$F = \frac{\dot{m}}{g_c} M_e \sqrt{\frac{\gamma g_c R T_{te}}{1 + \frac{\gamma-1}{2} M_e^2}} + (P_e - P_a) A_e \quad (2)$$

During the interval before motor ignition the following data were applicable: $T_{e(m)} = 1000^\circ R$, $F_m = 28$ lbf, $\gamma = 1.4$, $A_e = 0.94$ in², $R = 53.3$ ft-lbf/lbm- $^\circ R$, $p_e \leq 14.7$, and $\dot{m}_e = 0.480$ lbm/sec.

If $p_{e(m)} = 14.7$ psia, $M_e = 2.08$.

If $p_{e(m)} < 14.7$ psia, $M_e > 2.08$.

If $M_e = 1.0$, $p_{e(m)} = 28$ (too high).

The wall pressure upstream in the center section was a maximum of 20 psia during this portion of the run. Thus, attaining 28 psia was not plausible.

The preliminary conclusion to be drawn from this (based on thrust) was that M_e must have been greater than 1.0 before ignition. The upper limit for M_e was 3.1, based upon isentropic expansion from the inlet nozzle throat area to the grain exit area, with no heat addition.

During the interval after motor ignition the following data were applicable: $T_{te(\min)}$ (only H_2 burned) = $1317^\circ R$, $T_{te(\max)}$ (all H_2 and Plexiglas burned) = $2394^\circ R$, $F_m \cong 42$ lbf, $p_{cm} \approx 14.7$ psia, $\gamma = 1.3$, $R = 54.7$ ft-lbf/lbm/ $^\circ R$, $A_{c(\text{avg})} = 0.989$ in² (the average of the exit area between the initial area and the area after motor burn) or, $A_c = 0.600$ (in the event of flow separation at the first step upstream of the exit plane) and $\dot{m}_c = 0.495$ lbm/sec. In addition to satisfying the momentum (thrust) equation, continuity must also be satisfied:

$$\dot{m}_e = P_e M_e A_e \sqrt{\frac{\gamma g_c (1 + \frac{\gamma-1}{2} M_e^2)}{RT_{te}}} C_D \quad (3)$$

Assuming $C_D = 1.0$, $T_{te} = 1317^\circ R$ (only H_2 burned), then $M_e = 0.4$ and $p_e = 40$ psia, or $M_e = 3.9$ and $p_e = 1.0$ psia. The first solution was not possible, since the exit static pressure was

approximately 15 psia. The second solution was not possible since the Mach number was higher than the Mach number for isentropic expansion with no heat addition (3.1).

If $T_{t_e} = 2394^\circ\text{R}$ (all H_2 and Plexiglas burned), then $M_e = 0.5$ and $p_e = 37$ psia, or $M_e = 1.2$ and $p_e = 15$ psia. Again, the first solution was not possible due to the exit pressure, but the second solution was possible. Thus, in order to satisfy the measured thrust and the measured mass flow rate, the exit Mach number should be between 1.2 and 3.1.

In the thrust-time trace, the measured thrust fell off to an average of 7 lbf during passage of the probes across the exit plane. This low a value of thrust could not be explained with one-dimensional flow at the grain exit (for sub or supersonic flows). Thus, either the flow was diverted away from the axial direction at the exit plane or the side force affected the bearings and/or load cell of the thrust stand. The former would require a shock to be present around the probe which would affect the upstream flow, inside the combustor. Nevertheless, all indications were that supersonic flow was maintained through the motor.

B. COMBUSTION EFFICIENCY

Combustion efficiency is the ratio of the observed heat addition from combustion to the maximum heat addition attainable from the fuel expended:

$$\eta_{\Delta T} = \frac{T_{te_m} - T_{t_i}}{T_{te_{th}} - T_{t_i}} \quad (4)$$

Since the exit stagnation temperature measured at the exit was found to be invalid (due to failure of the cooling water jacket), the "experimental" exit stagnation temperature had to be calculated based upon other measured data.

The stagnation pressure ratio across a normal shock is given by:

$$\frac{P_{t_e}}{P_{t_u}} = \left(\frac{\frac{(\gamma+1)M_e^2}{2}}{1 + \frac{(\gamma-1)M_e^2}{2}} \right)^{\frac{\gamma}{\gamma-1}} \left(\frac{2\gamma M_e^2}{\gamma+1} - \frac{\gamma-1}{\gamma+1} \right)^{\frac{1}{1-\gamma}} \quad (5)$$

The stagnation pressure downstream of the normal shock on the probe had an average value of 47 psia. With $\gamma = 1.3$ and an initial estimate for M_e , Equation 5 yields p_w . Then,

$$\frac{P_{t_e}}{P_e} = \left(1 + \frac{\gamma-1}{2} M_e^2 \right)^{\frac{\gamma}{\gamma-1}} \quad (6)$$

yields p_e . Equation 3 can then be used to solve for $T_{t_c(m)}$, assuming that $C_D \approx 1.0$ for the cylindrical grain. Equation 2 then yields F for the assumed value of M_e . M_e can be iterated until the calculated value of $F = F_m$.

Figure 6 presents the possible values which satisfy Equations 2,3,4 and 6 and the measured data. It shows that the final combustion stagnation temperature is very sensitive to the exit Mach number (through the thrust) and that the best performances (high thrust, high combustion efficiency) are attained if the flow is slowed to a Mach number of near unity. Because of the sensitivity of the solution to small errors in measured thrust it was decided to let $p_c = 14.8$ psia, the average value during the run. With $p_c = 14.8$ psia, Figure 6 shows that $F = 41.5$ lbf, $M_c = 1.48$, $T_{tc} = 1930^\circ\text{R}$ and $p_{tc} = 50.4$ psia.

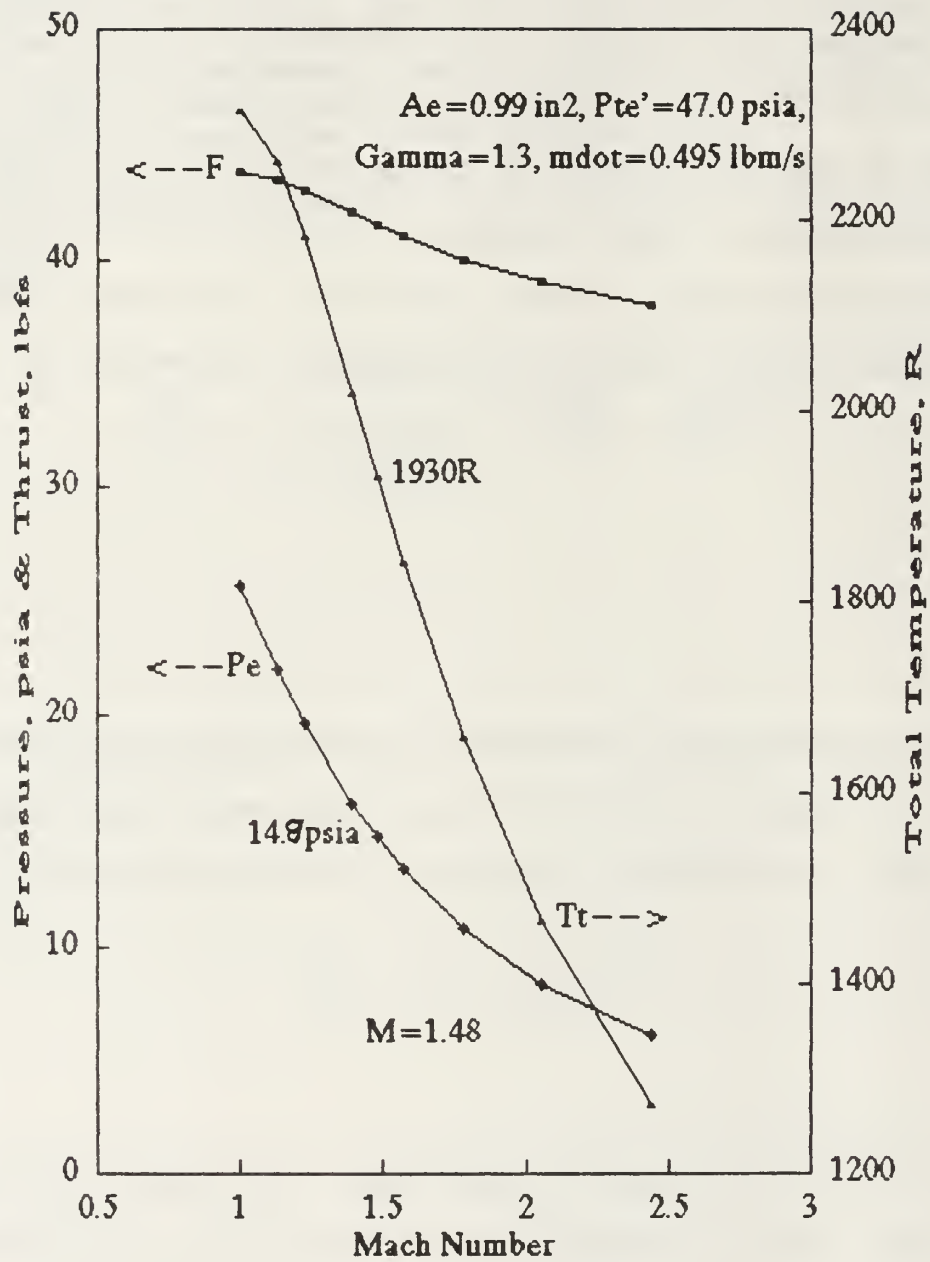


FIGURE 6 Relationship Between T_{te} , M_e , F , p_e and Measured Data

For these conditions, the drop in the stagnation pressure across the combustor was about $p_c/p_t = 0.34$, revealing the existence of additional losses (weak shocks, boundary layer detachment, etc) to those of friction and heat addition.

$\eta_{\Delta T}$ can then be estimated using Equation 4. For all of the fuel utilized ($H_2 + \text{Plexiglas}$):

$$\eta_{\Delta T} = \frac{1930 - 1000}{2394 - 1000} = 0.67 \quad (7)$$

The combustion efficiency of the solid fuel alone can also be computed:

$$\eta_{\Delta T} = \frac{1930 - 1317}{2394 - 1317} = 0.57 \quad (8)$$

This low combustion efficiency indicates a major portion of the Plexiglas vapor did not burn. Since kinetic rates are quite fast and the Mach number was only approximately 1.5, the poor combustion efficiency probably resulted from poor mixing of the fuel (which is pyrolyzed at the wall) with the oxygen in the core air flow. Improved mixing techniques are needed which at the same time do not greatly increase losses in stagnation pressure.

VI. CONCLUSIONS AND RECOMMENDATIONS

The major conclusions reached in this investigation were:

1. Supersonic Mach numbers were maintained in the SFSJ utilized.
2. With inlet and exit Mach numbers of approximately 1.4, the solid fuel combustion efficiency was approximately 0.57.
3. To improve combustion efficiency, mixing of the pyrolyzed PMM with the core air must be enhanced. This must be done without significantly increasing losses in stagnation pressure.
4. The concept of a simple missile propulsion device based upon the solid fuel scramjet appears feasible. Of course, as the grain burns, the Mach number through the motor increases. This may reduce $\eta_{\Delta T}$.
5. Optimum flameholding and combustor geometries need to be determined which minimize p_t losses and the amount of pilot hydrogen required.
6. Recommendation 1: To better estimate the exit Mach number of the SFSJ, schlieren photography should be taken of the shocks formed by placing a wedge in the exit flow. Using two different wedge angles, and measuring the oblique shocks formed on each side of the wedge, the exit Mach number could be very accurately determined over a broad range (Figure 7).

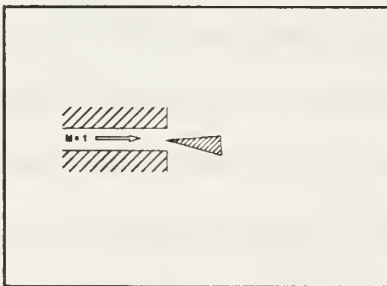
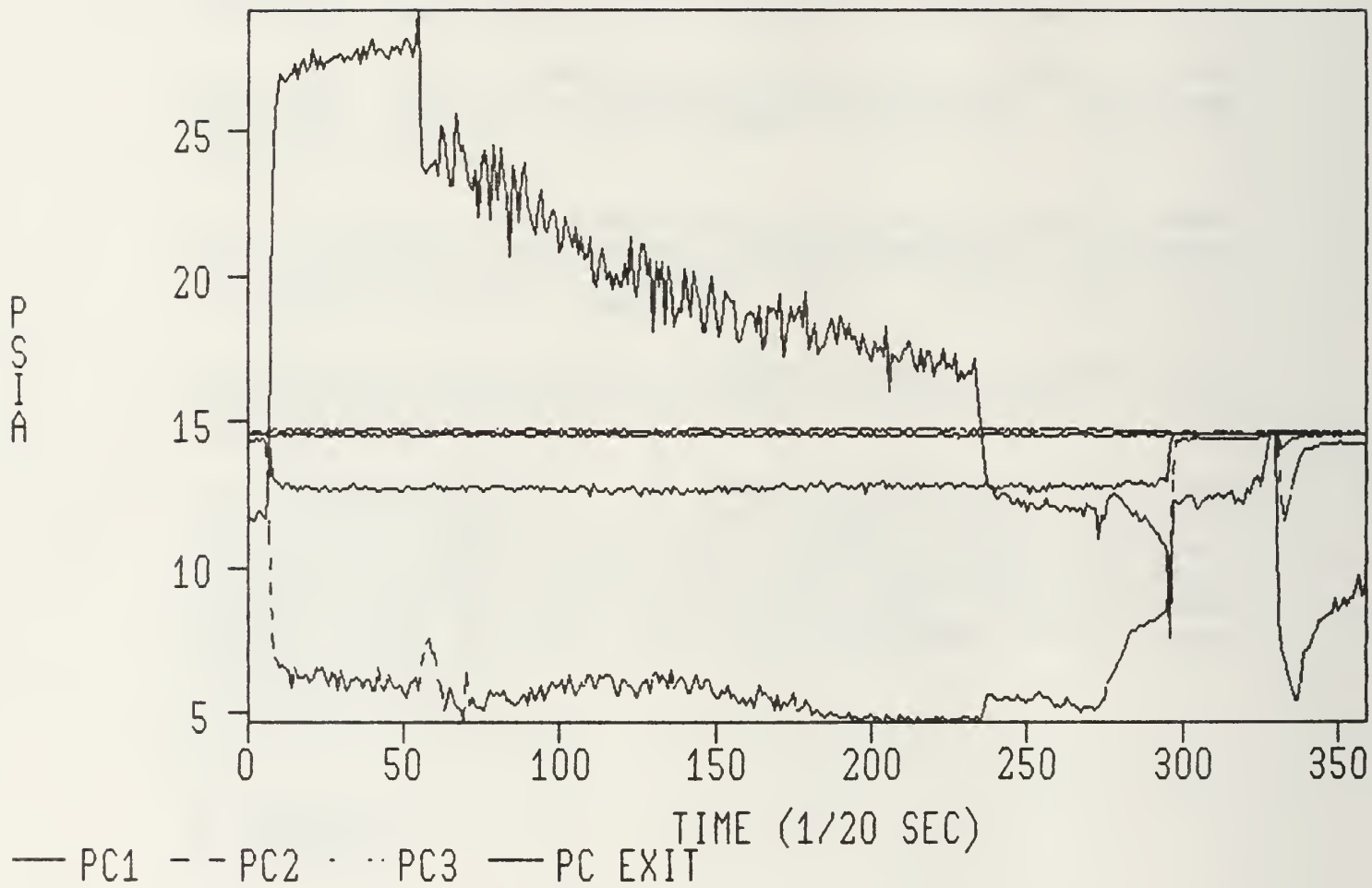


FIGURE 7 Shock Angles Due to Wedge in Exit Flow

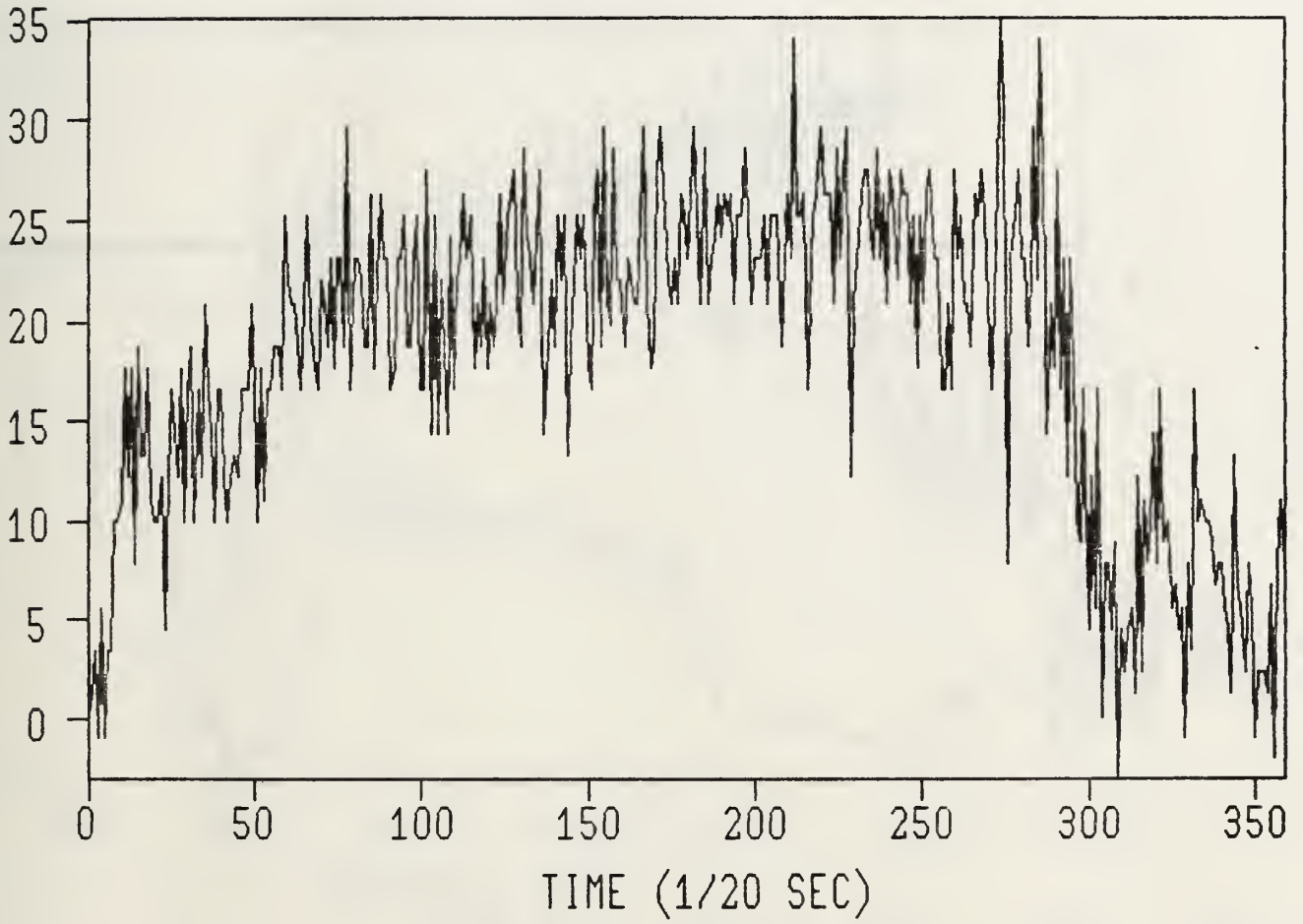
7. Recommendation 2: An improved stagnation temperature probe should be incorporated into the instrumentation in order to attain useable exit temperature data.

APPENDIX

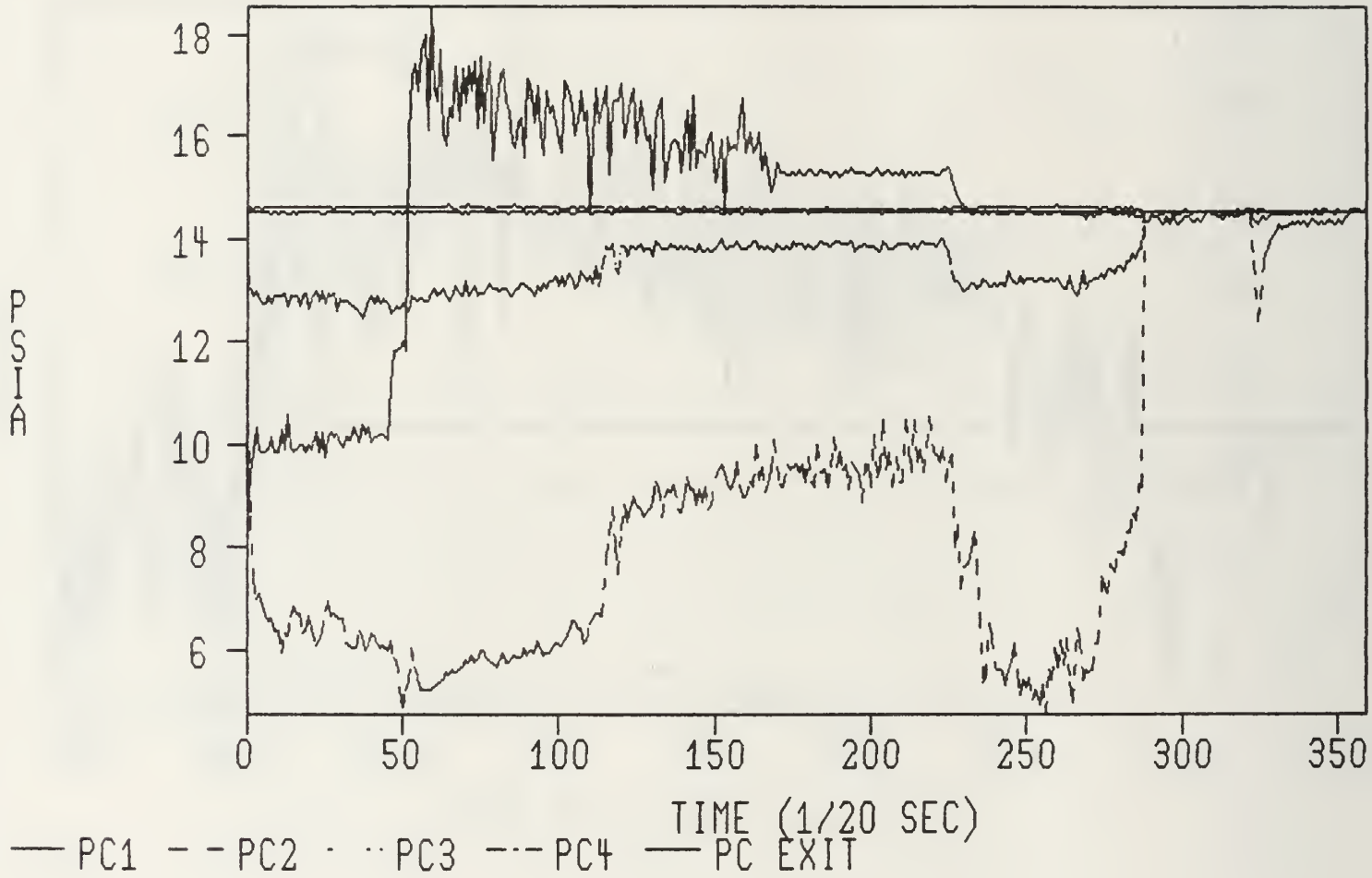
RUN #1 STATIC PRESSURE WITHIN MOTOR



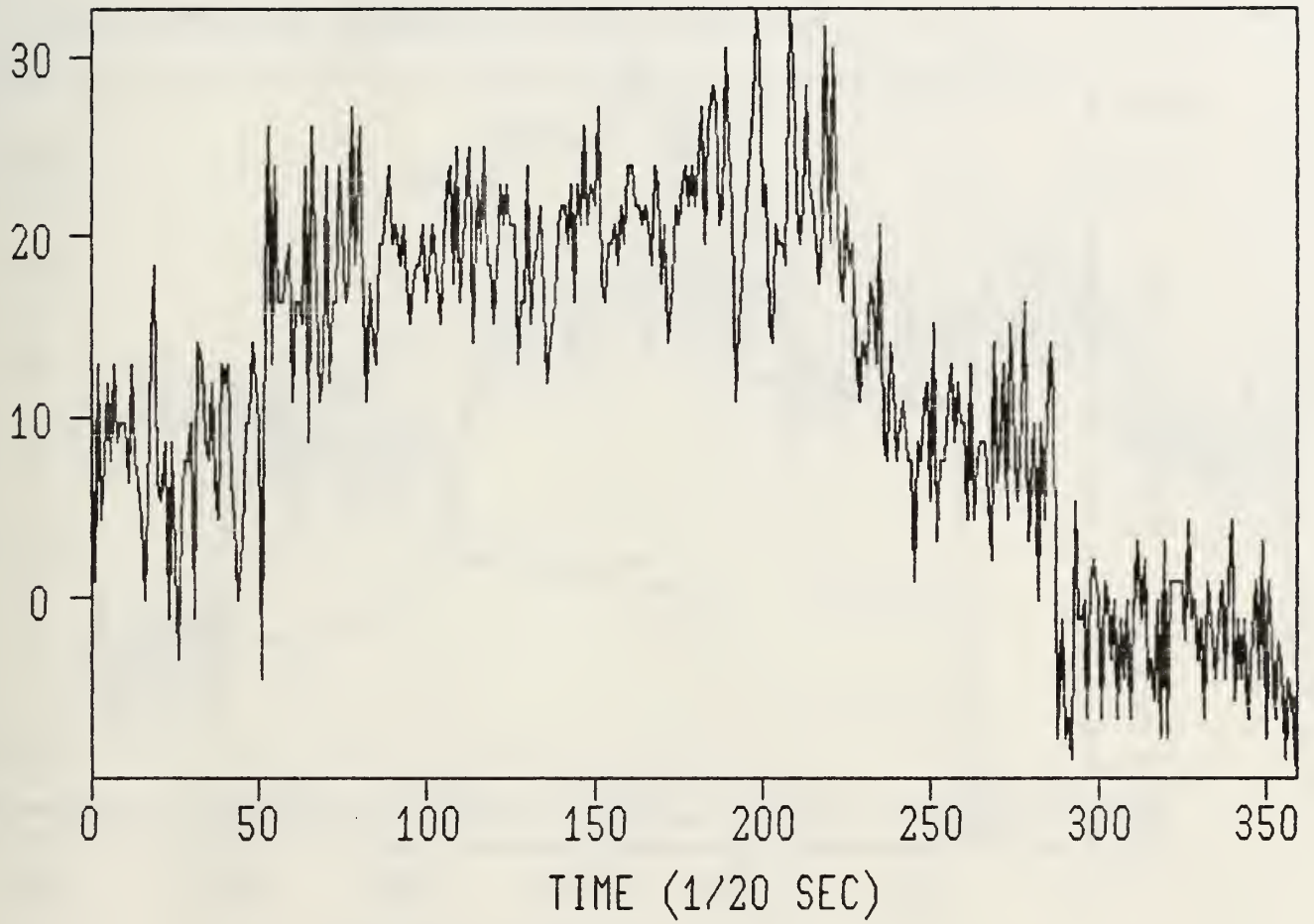
RUN #1 THRUST



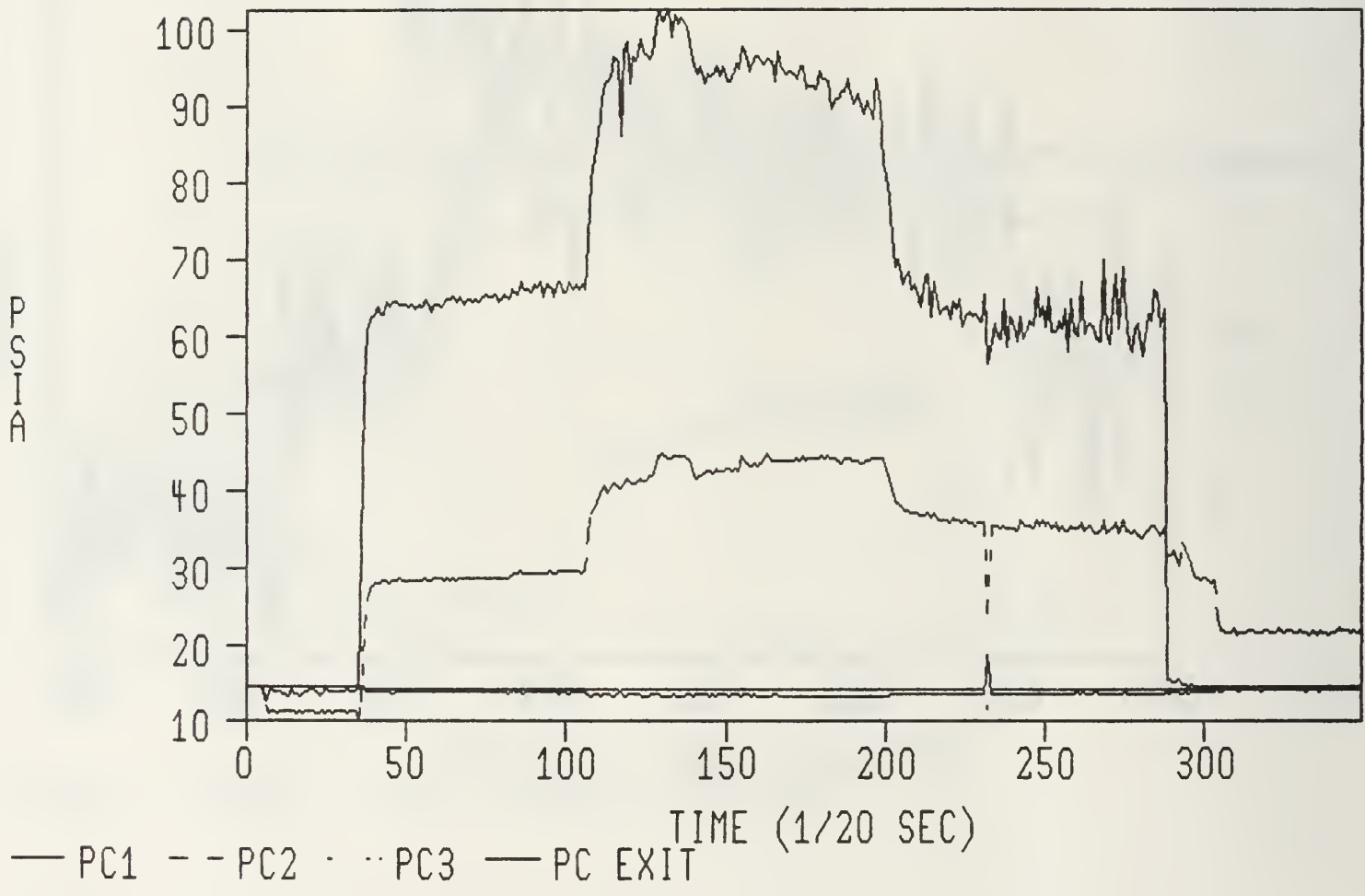
RUN #2 STATIC PRESSURE WITHIN MOTOR



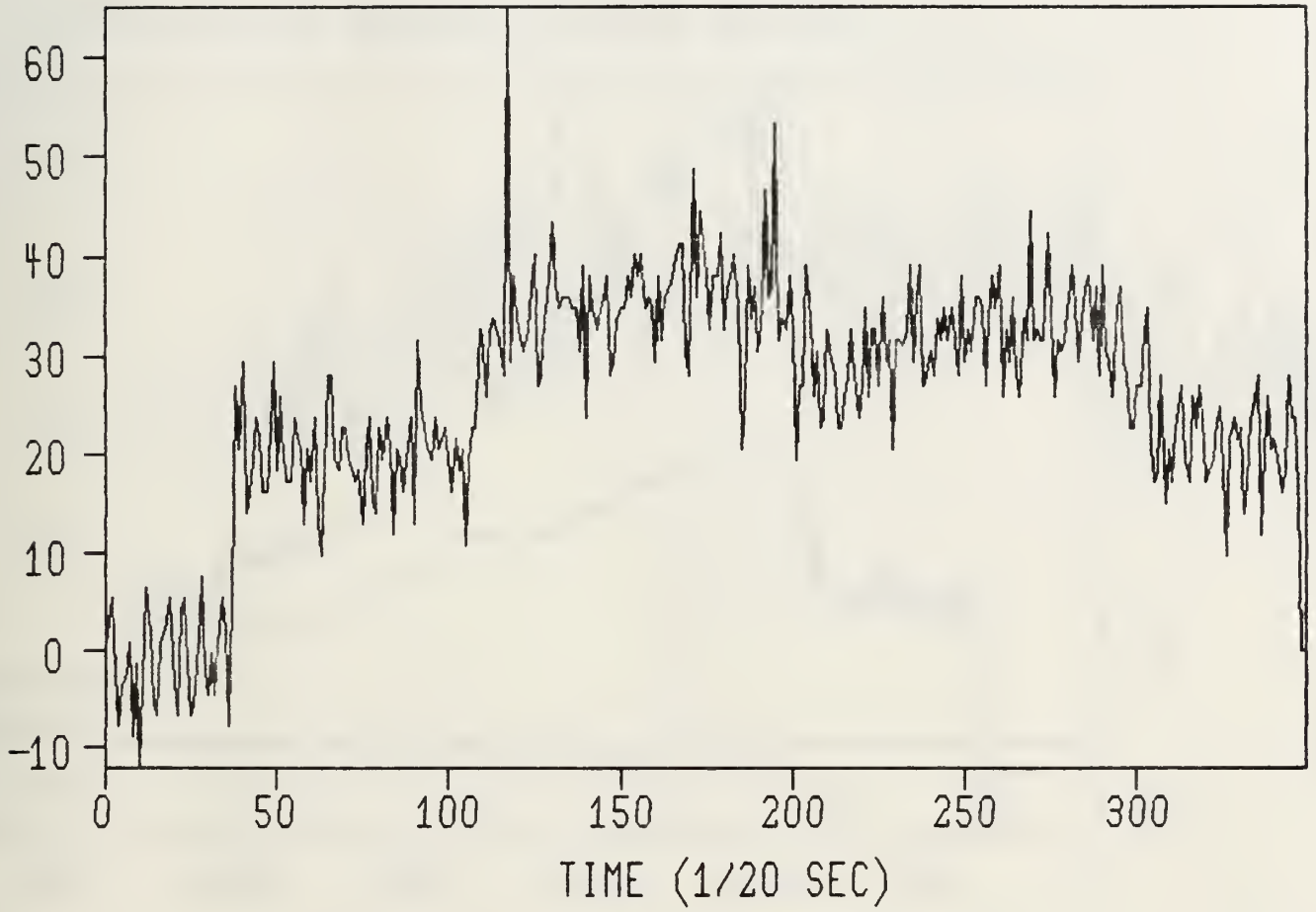
RUN #2 THRUST



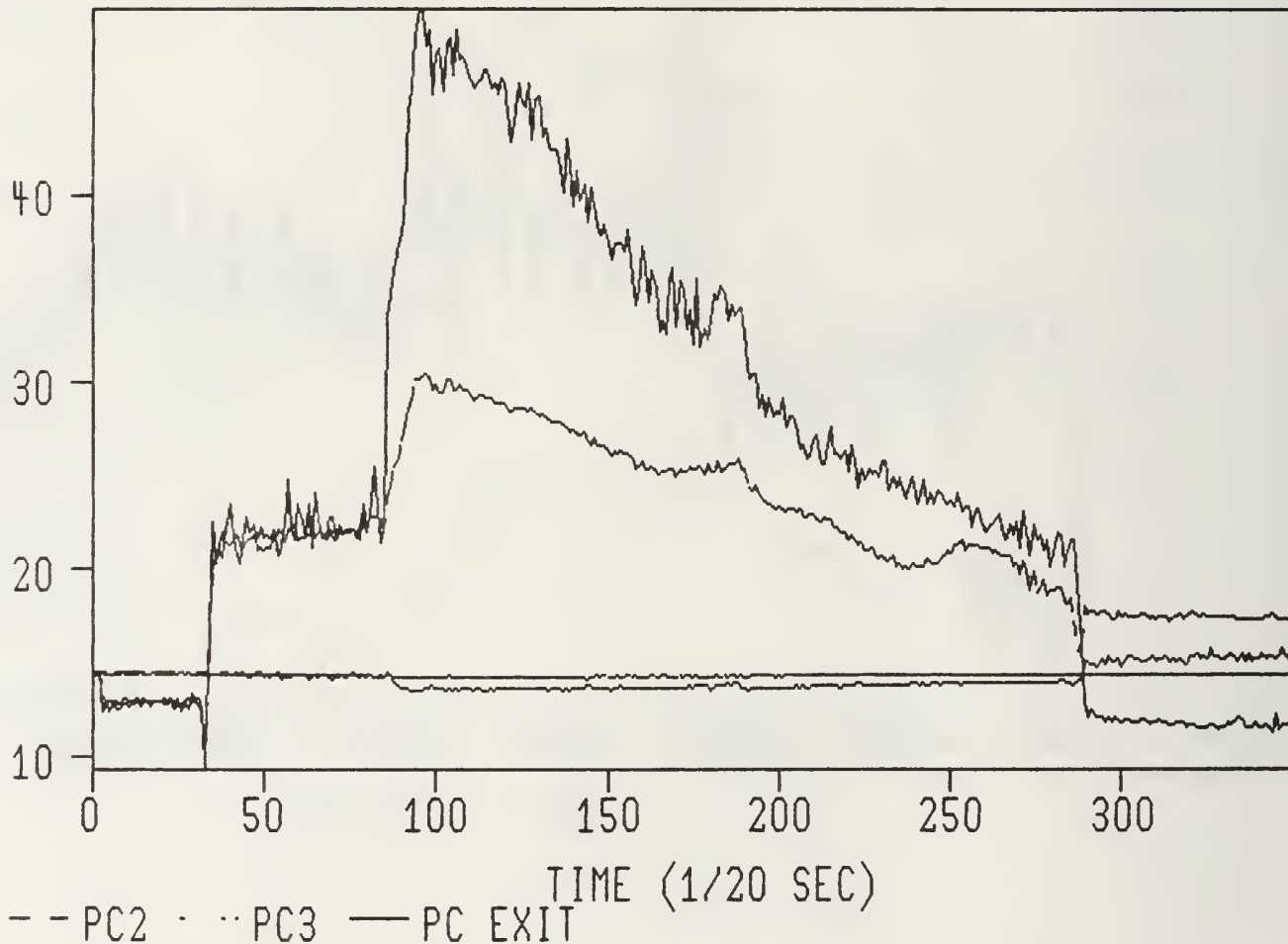
RUN #3 STATIC PRESSURE WITHIN MOTOR



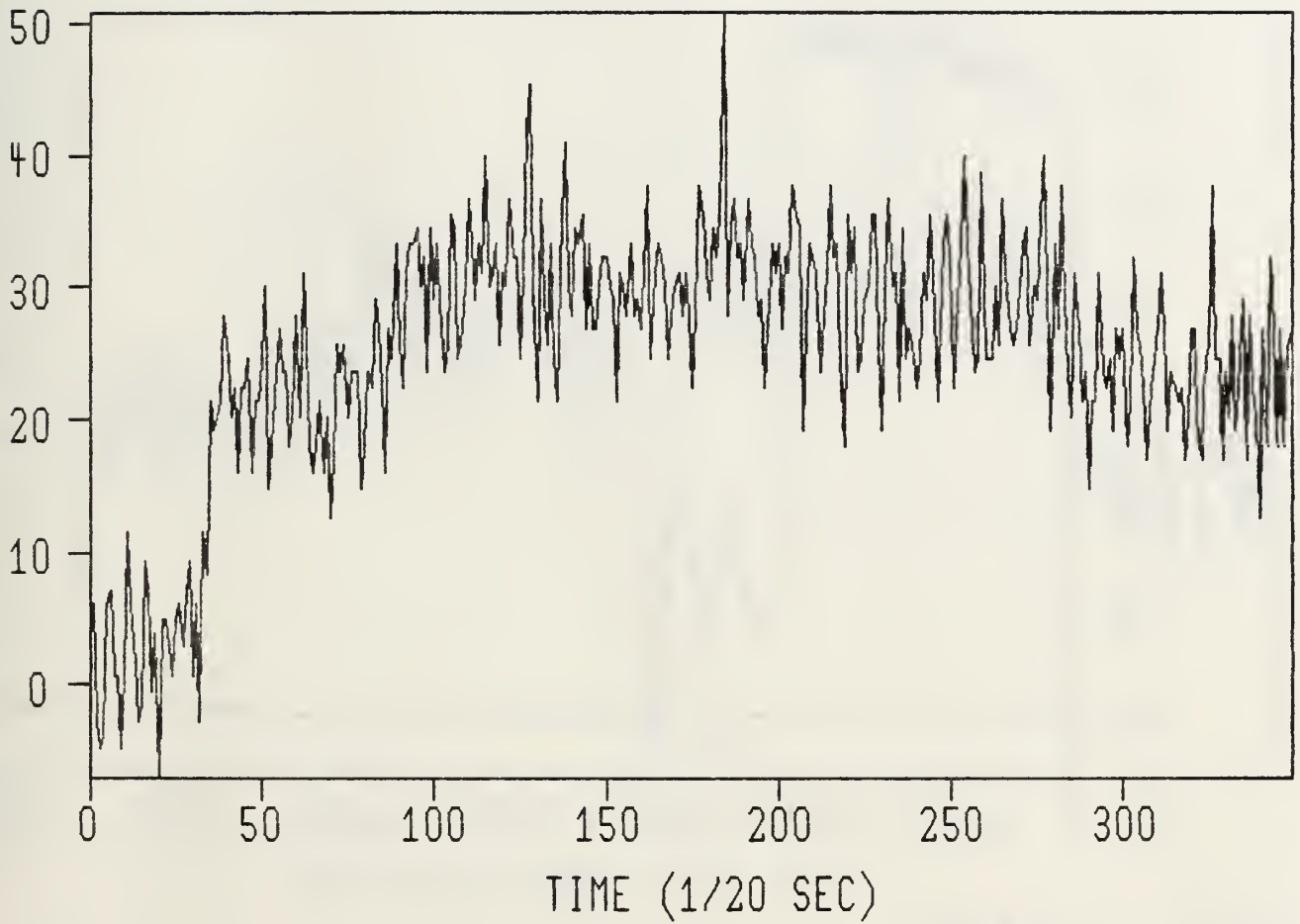
RUN #3 THRUST



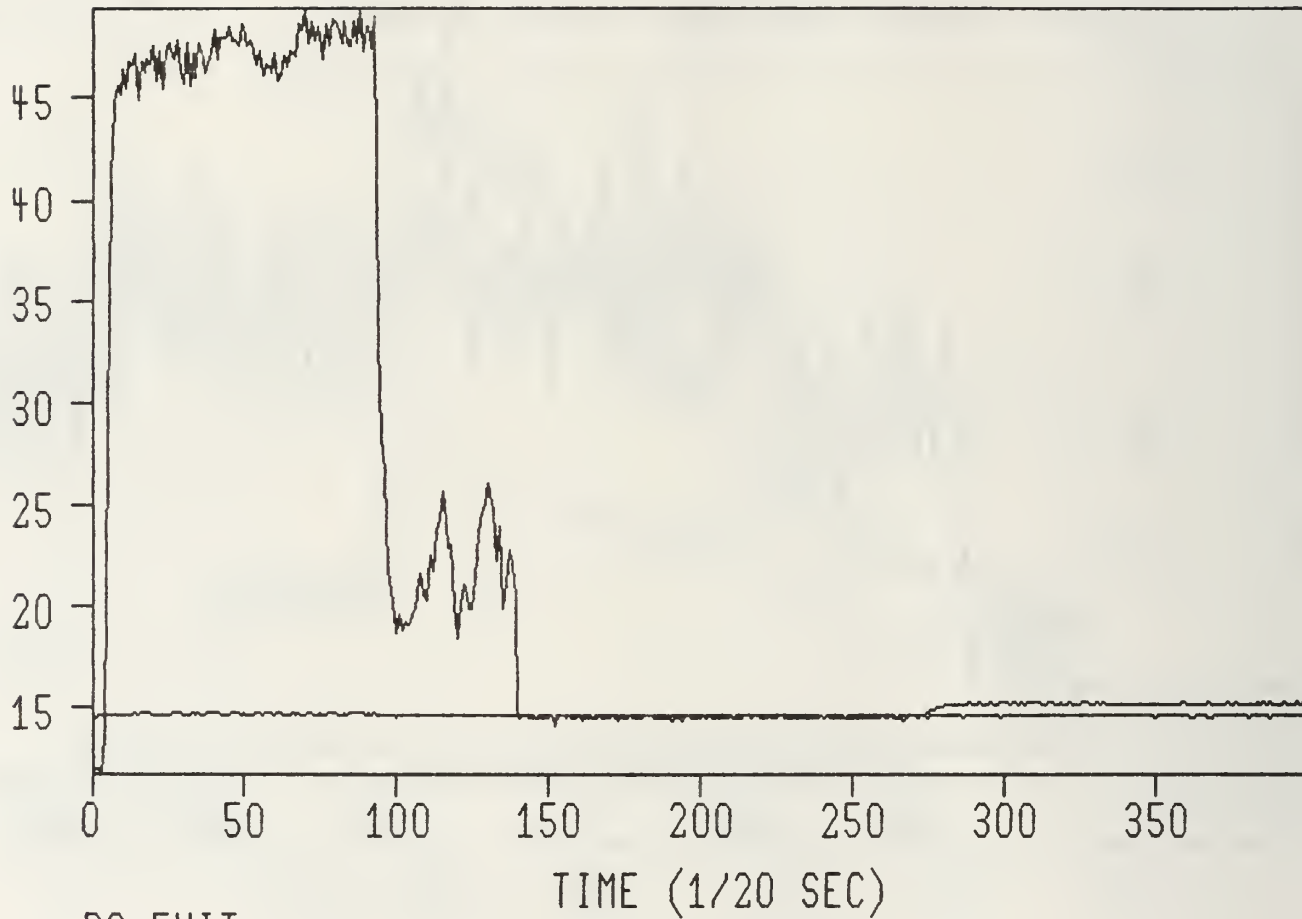
RUN #4 STATIC PRESSURE WITHIN MOTOR



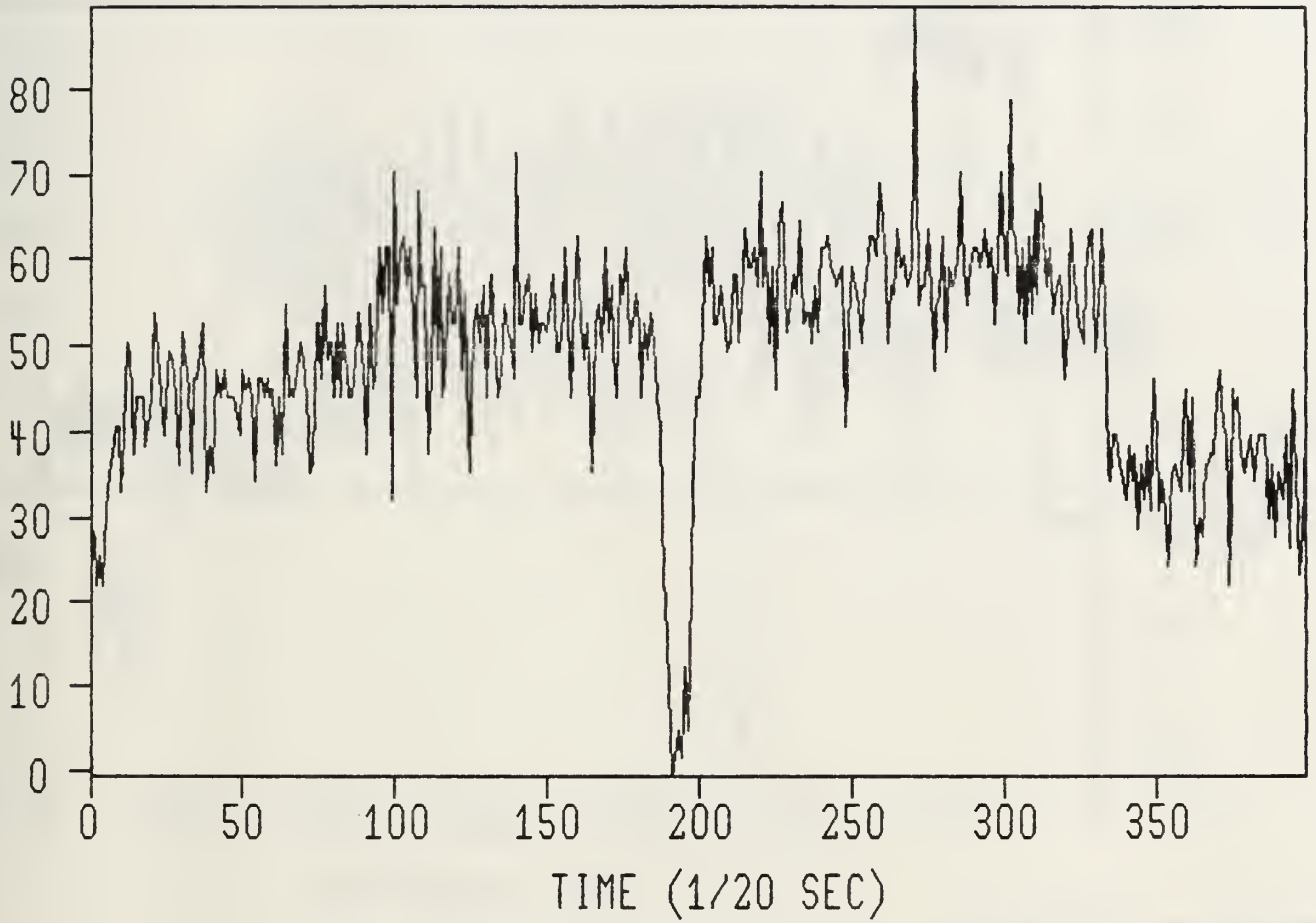
RUN #4 THRUST



RUN #5 STATIC PRESSURE WITHIN MOTOR

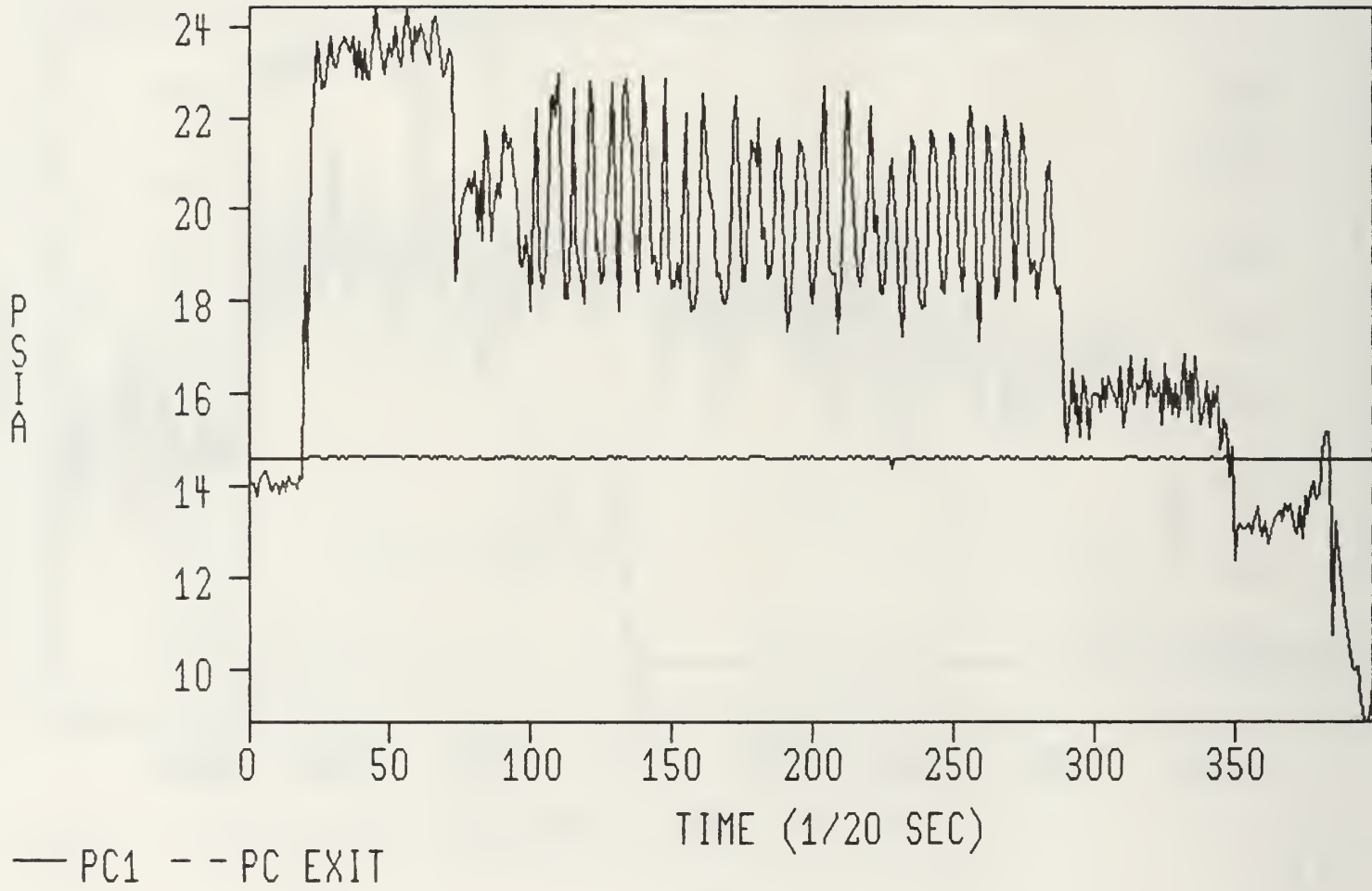


RUN #5 THRUST

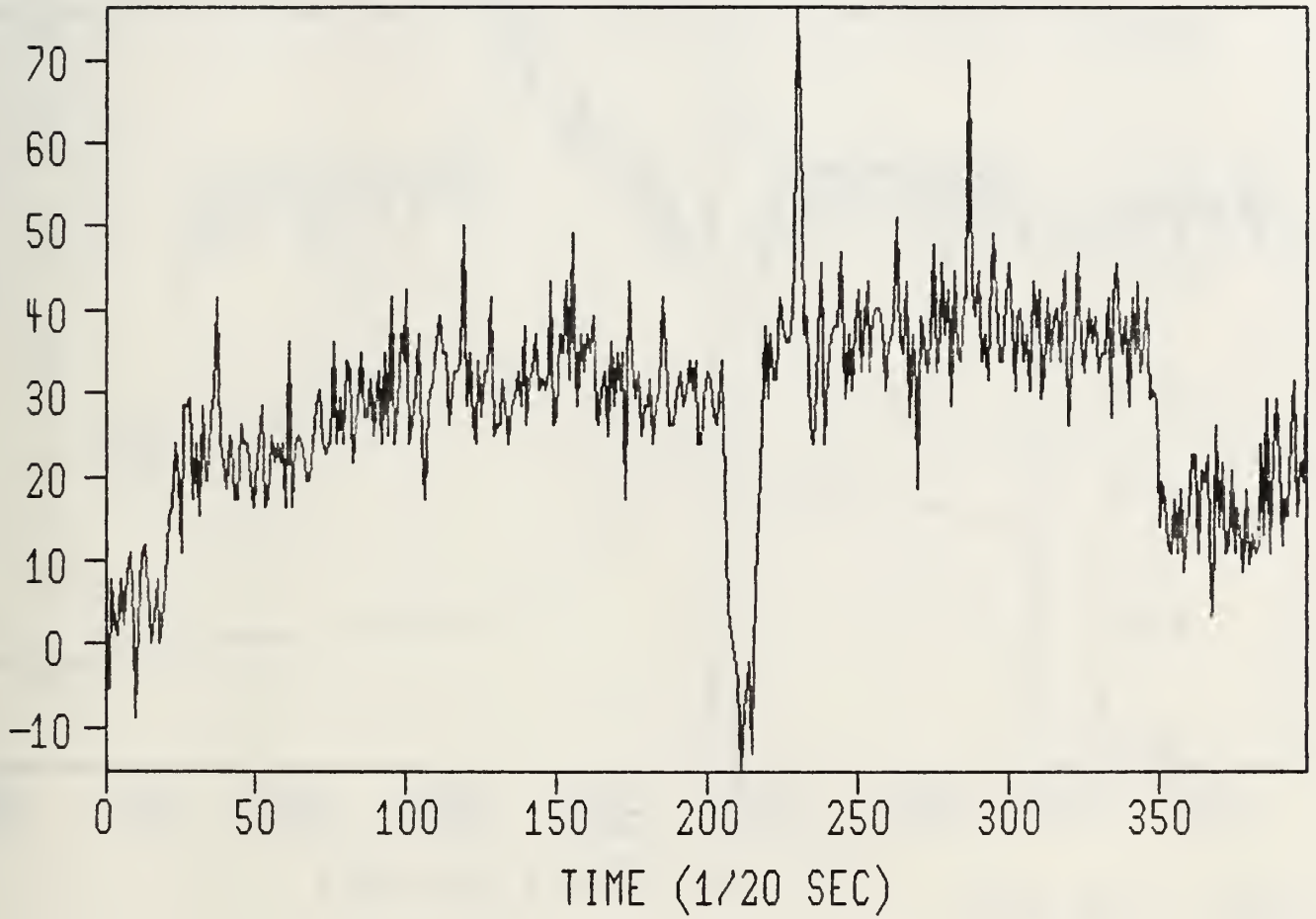


— PC1

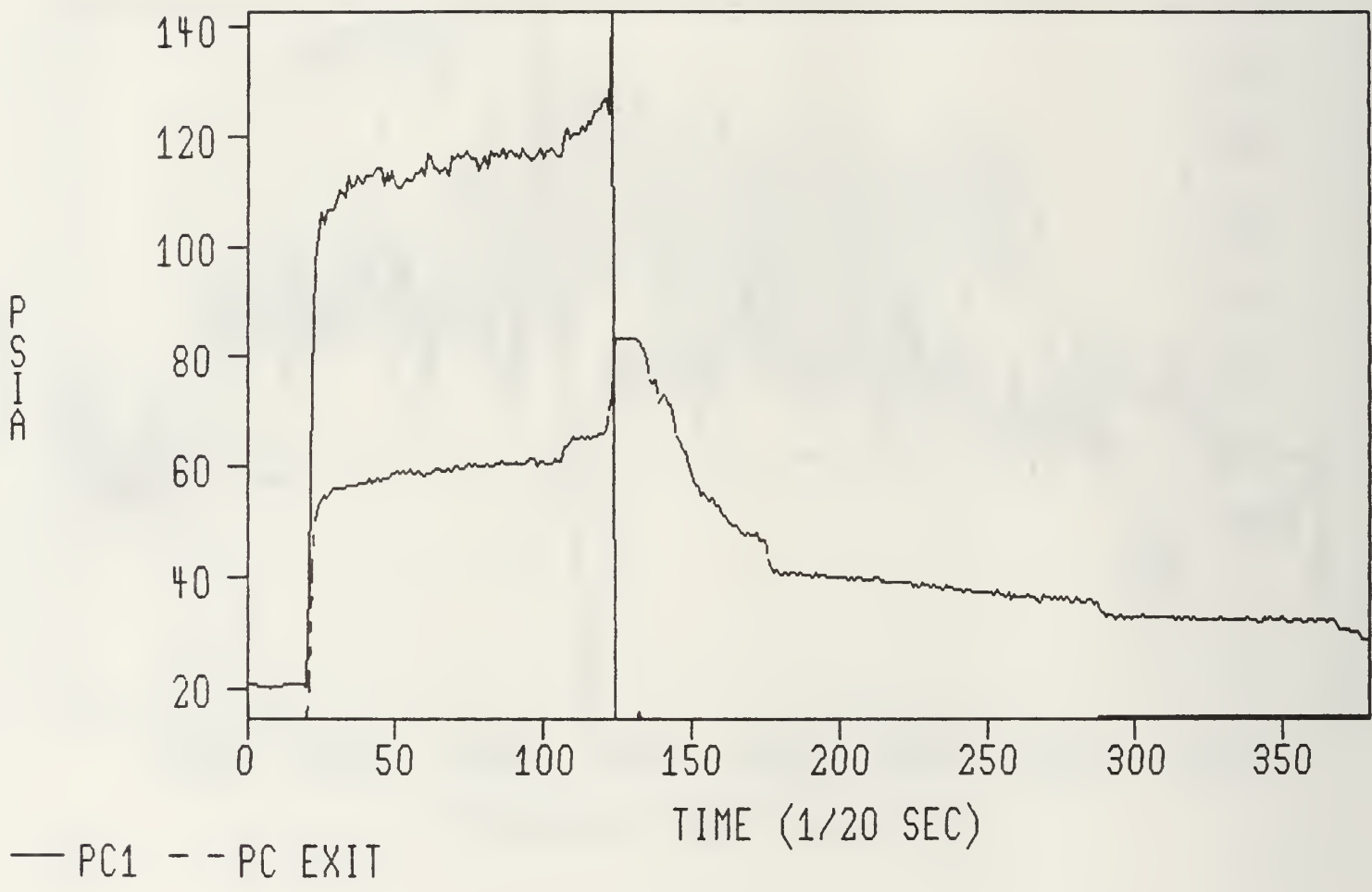
RUN #6 STATIC PRESSURE WITHIN MOTOR



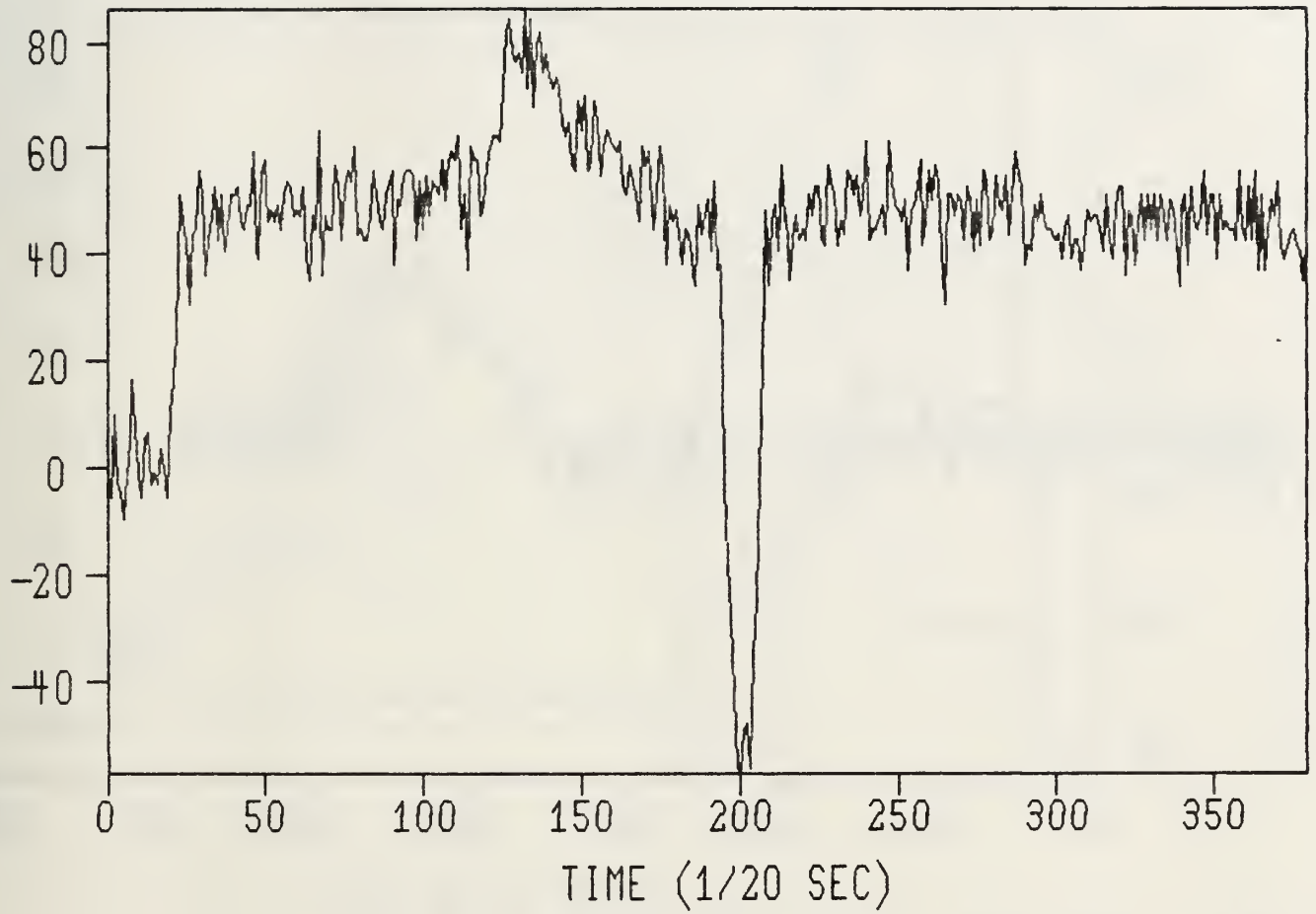
RUN #6 THRUST



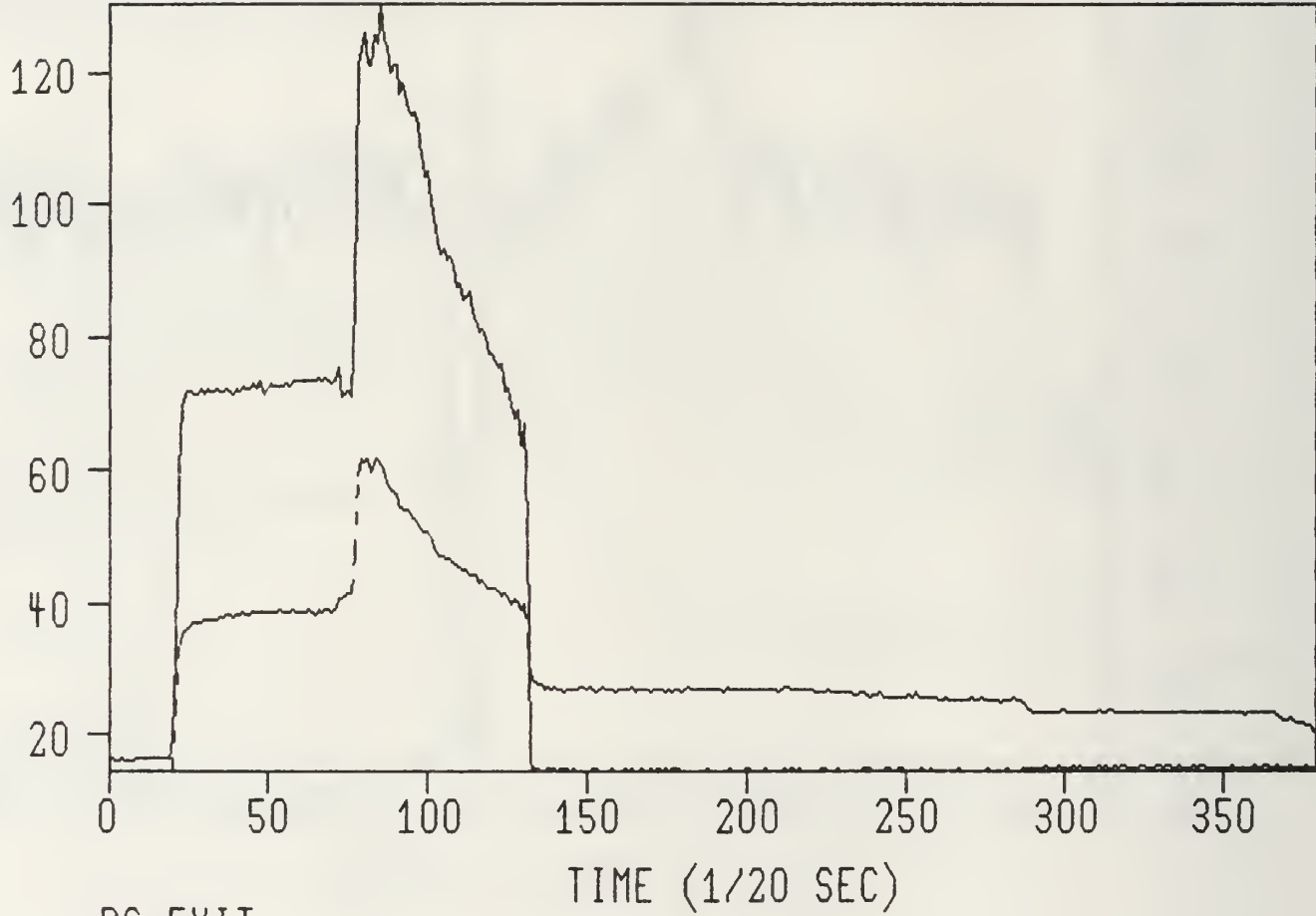
RUN #7 STATIC PRESSURE WITHIN MOTOR



RUN #7 THRUST

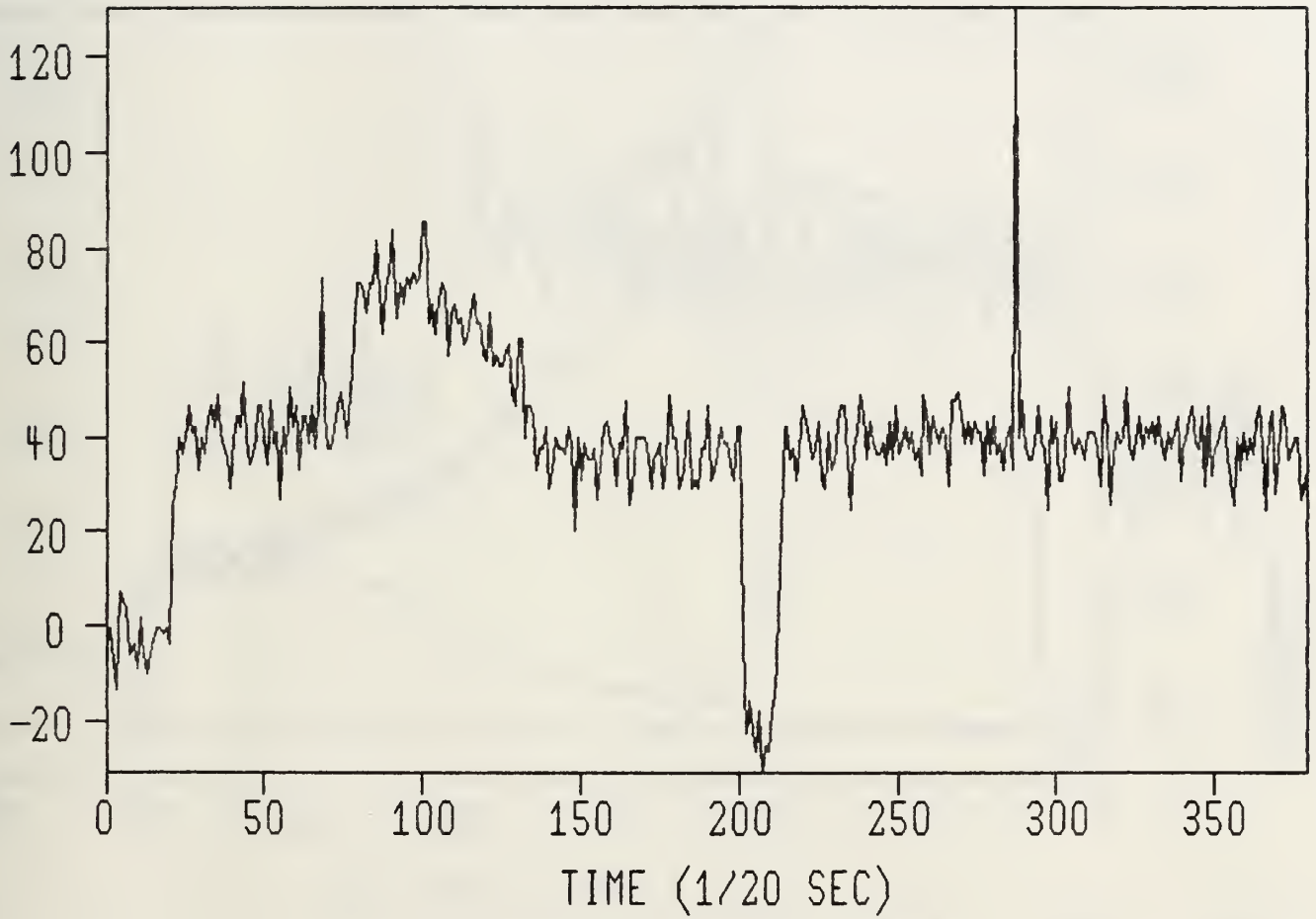


RUN #8 STATIC PRESSURE WITHIN MOTOR

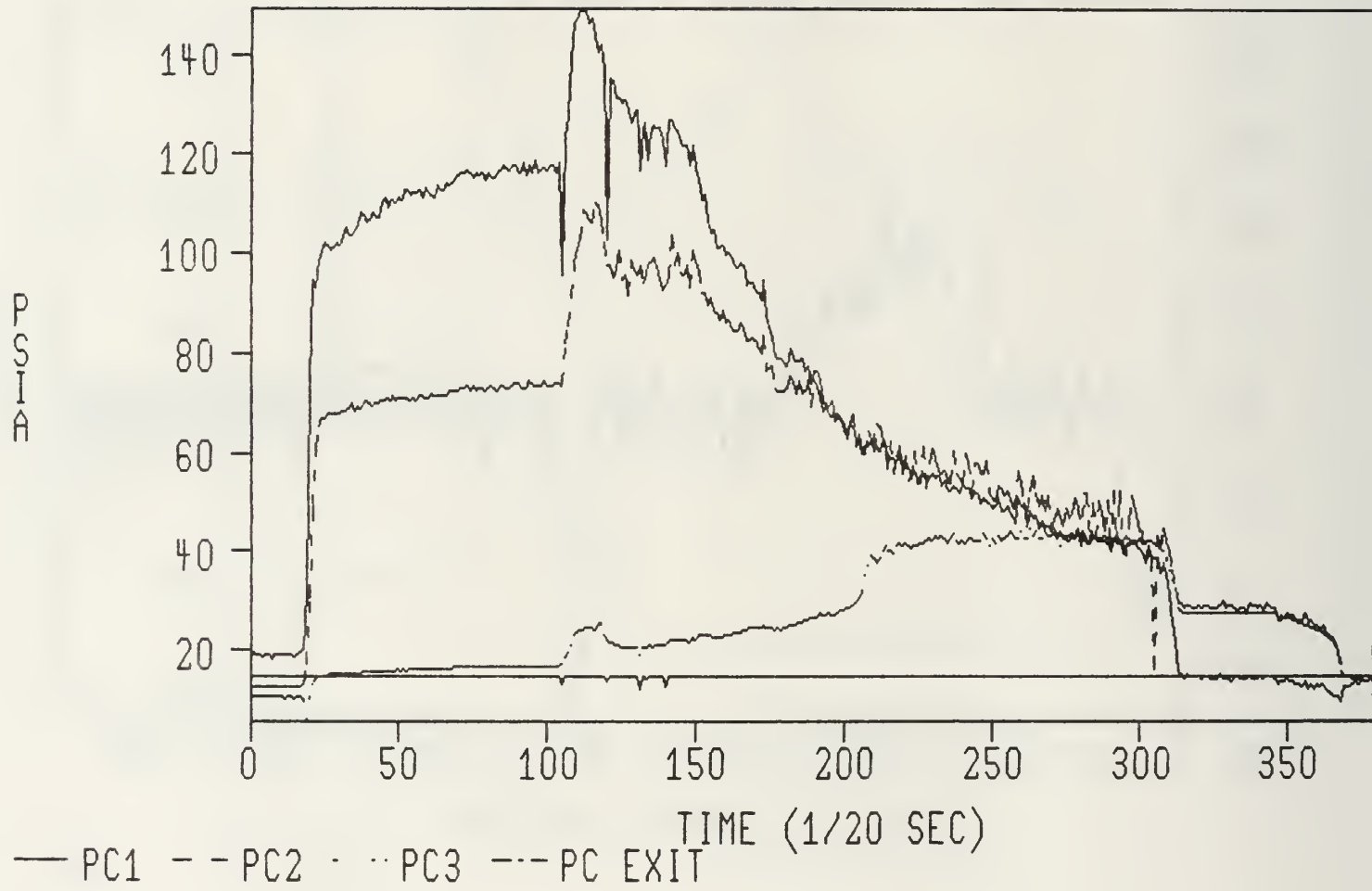


— PC1 - - PC EXIT .

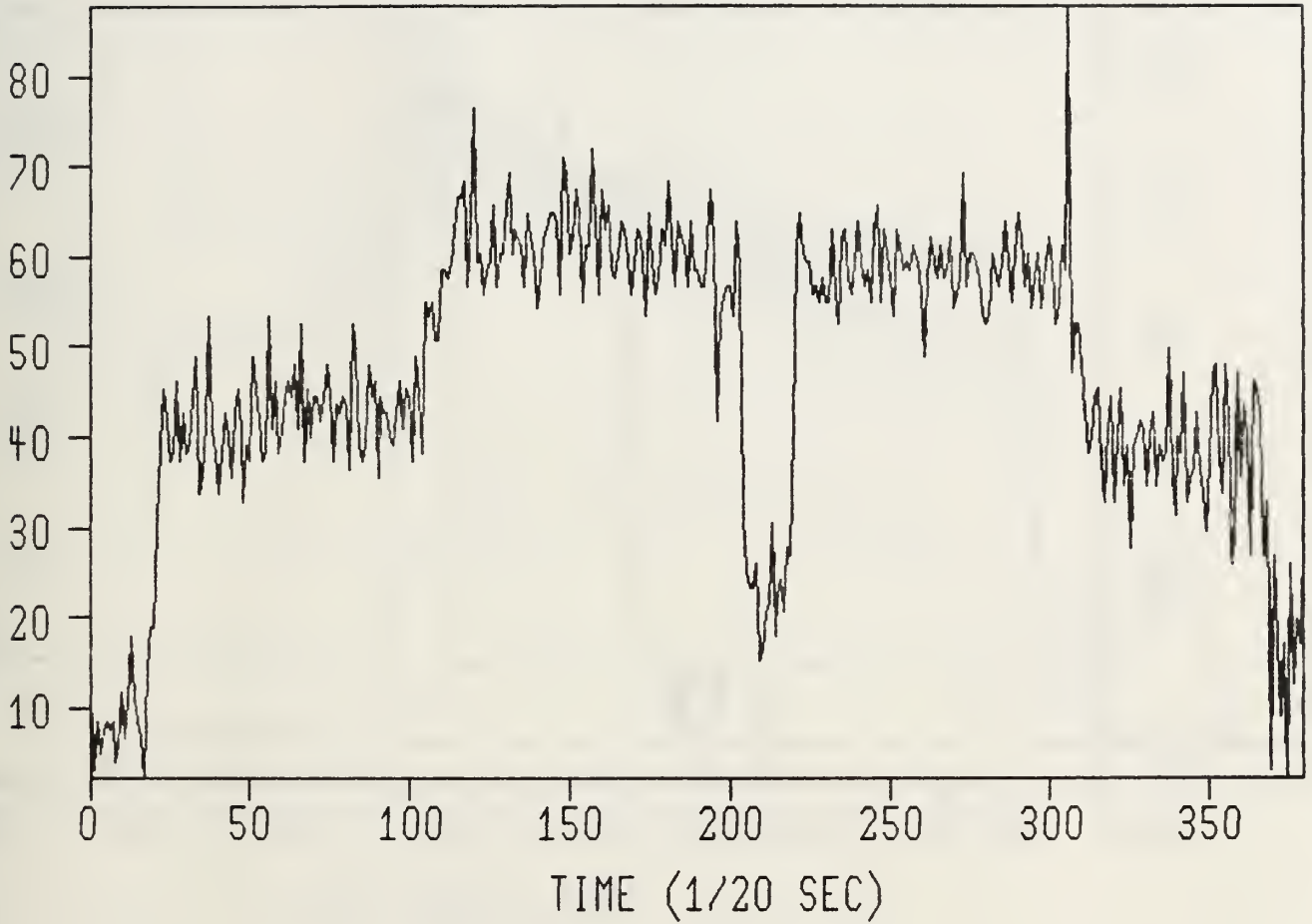
RUN #8 THRUST



RUN #9 STATIC PRESSURE WITHIN MOTOR

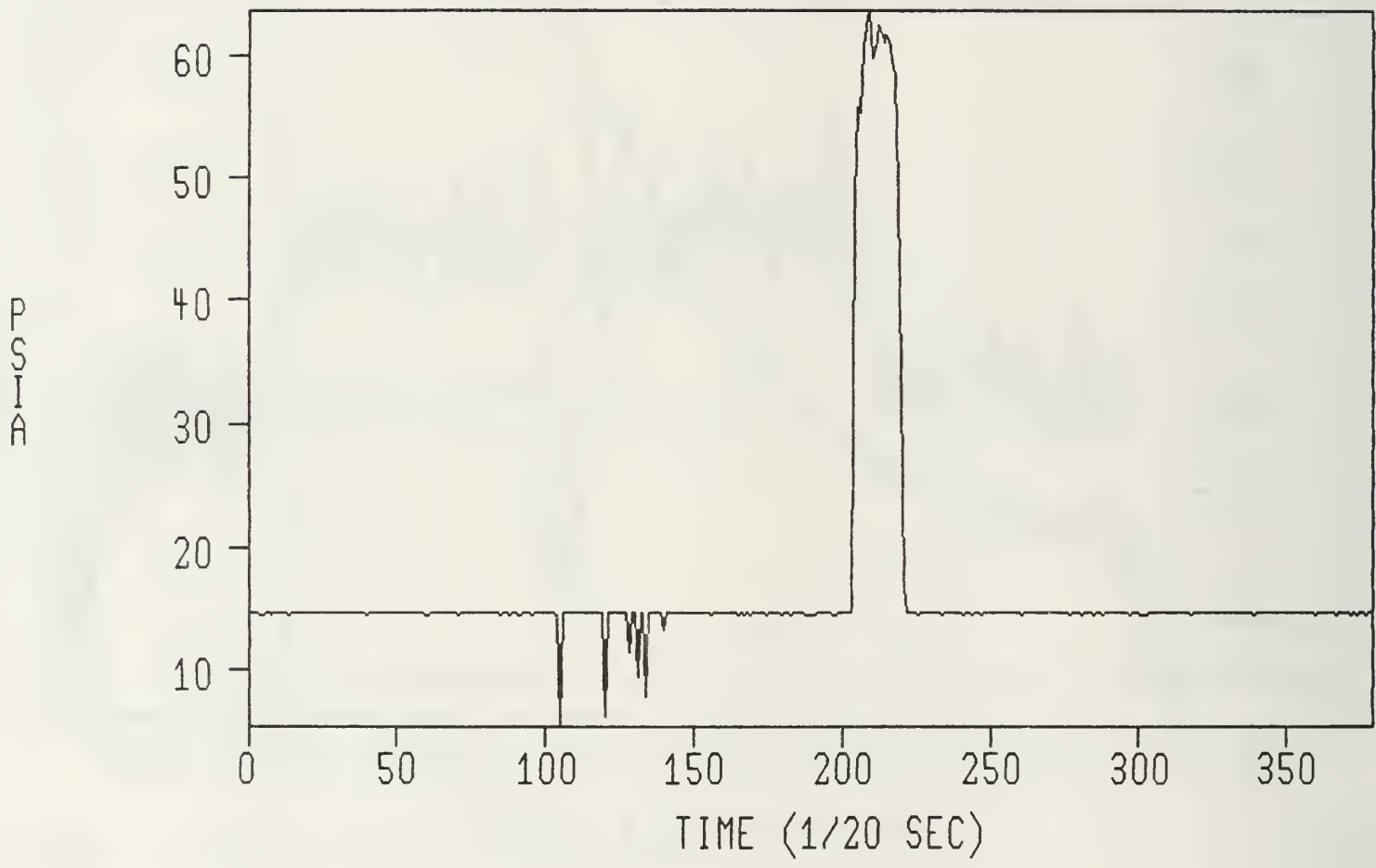


RUN #9 THRUST

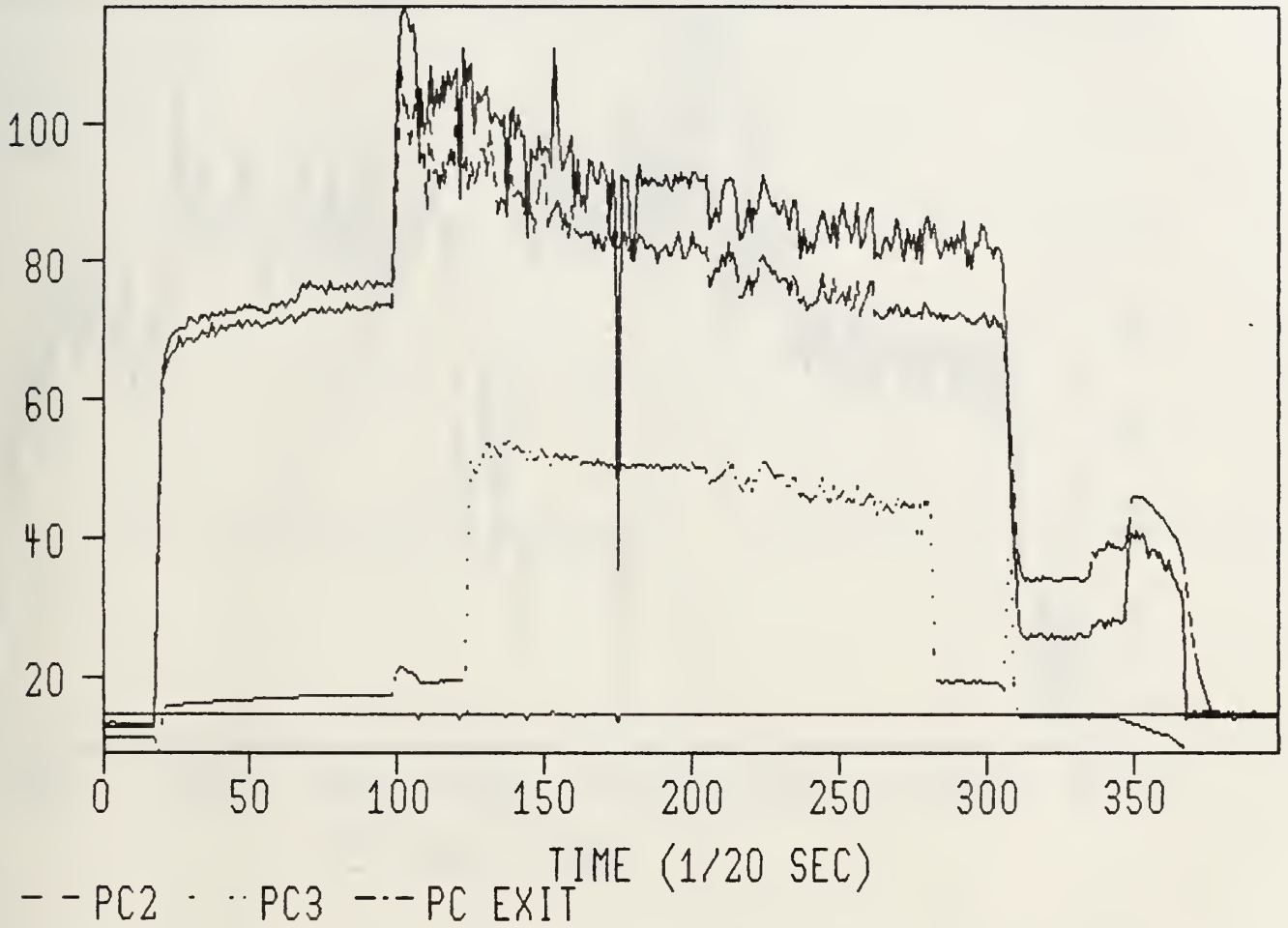


PC1

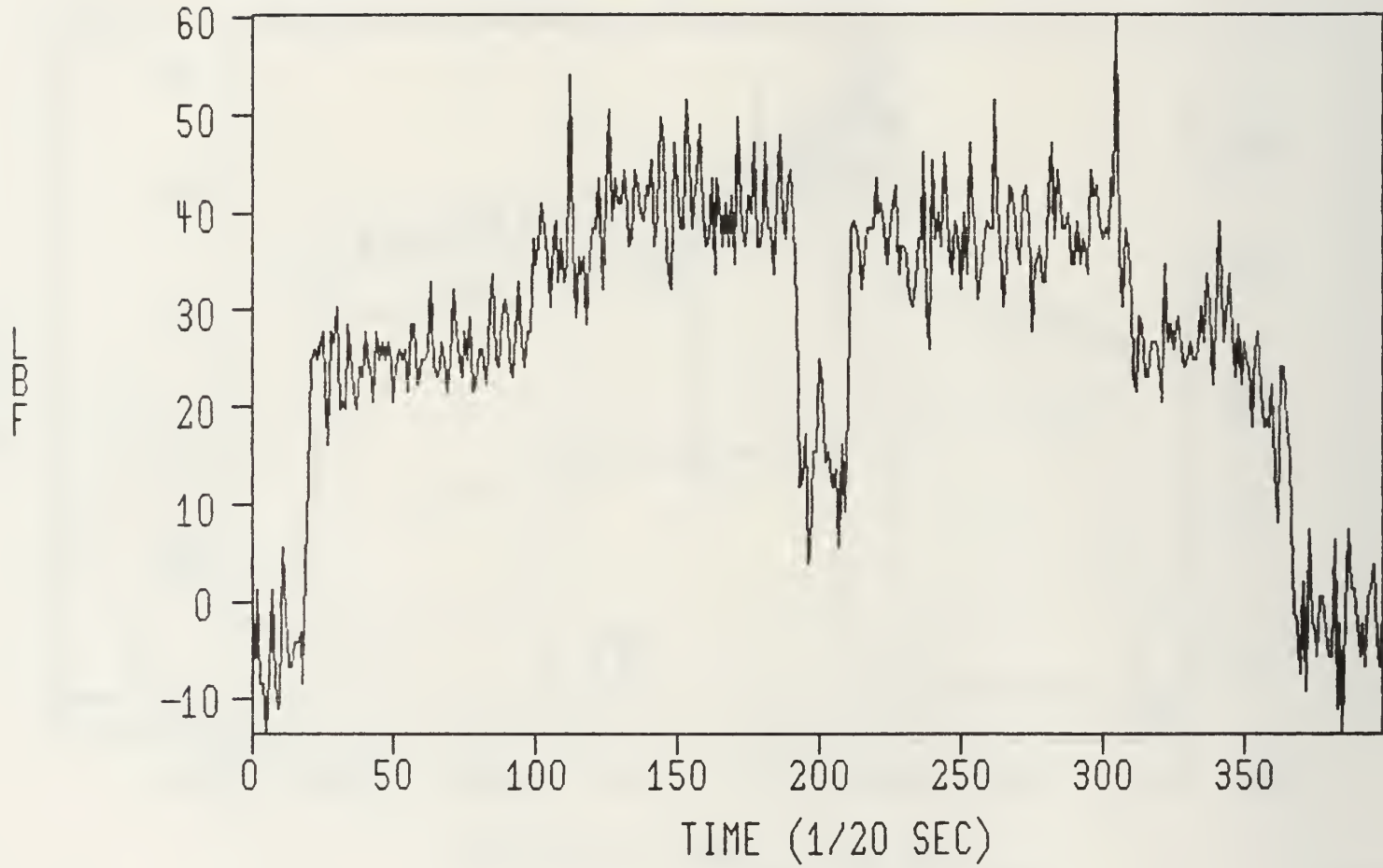
RUN #9 EXIT STAGNATION PRESSURE



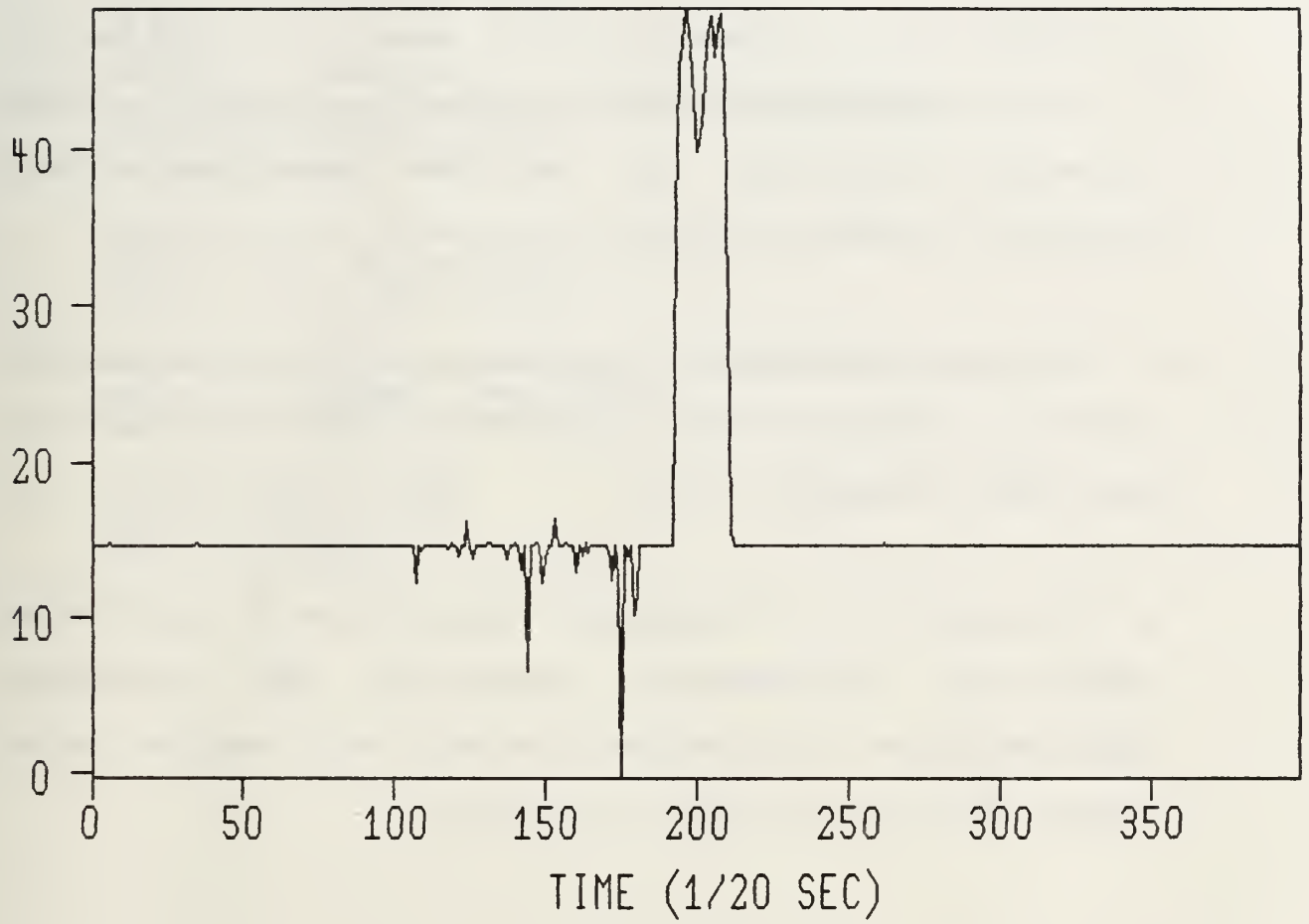
RUN #10 STATIC PRESSURE WITHIN MOTOR



RUN #10 THRUST



RUN #10 EXIT STAGNATION PRESSURE



LIST OF REFERENCES

1. Witt, M. W., *An Investigation into the Feasibility of Solid Fuel Dual Mode Ramjet and Scramjet Propulsion*, pp. 8-17, Naval Postgraduate School, 1989.
2. Snyder, T. S., Jarymowycz, T. A., Pace, K. K., and Kuo, K. K., *Solid Fuel Ignition and Combustion Characteristics Under High-Speed Crossflows*, pp. 4-7, The Pennsylvania State University, 1990.
3. Jarymowycz, T., Yang, V., and Kuo, K., *A Numerical Study of Solid Fuel Combustion Under Supersonic Crossflows*, pp. 2-16, Penn State University, 1990.
4. Cruise, D. R., *Theoretical Computations of Equilibrium Composition, Thermodynamic Properties, and Performance Characteristics of Propellant Systems*, Naval Weapons Center Report, NWC TP 6037, Rev. 1, Nov 1991.

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