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MSC-01855

SUPPLEMENT 4

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

APOLLO 12 MISSION REPORT

SUPPLEMENT 4

ASCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

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MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
SEPTEMBER 1972

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ASCENT PROPULSION SYSTEM
FINAL FLIGHT EVALUATION

PREPARED BY

TRW Systems

APPROVED BY

SEP 2[illegible] 1972

Owen G. Morris
Manager, Apollo Spacecraft Program

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
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17618-H001-R0-00

PROJECT TECHNICAL REPORT

APOLLO 12
LM-6
ASCENT PROPULSION SYSTEM
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NAS 9-8166

August 1970

Prepared for
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W. G. Griffin
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NASA/MSC

Concurred by: Z. D. Kirkland, Head
Systems Analysis Section

Concurred by: for R. E. Taylor, Manager
Ascent Propulsion Subsystem

Concurred by: C. W. Yodzis, Chief
Primary Propulsion Branch

TRW SYSTEMS

Approved by: R. J. Smith, Manager
Task E-19E

Approved by: P. H. Janak, Head
Propulsion Technology Section

Approved by: for J. W. Vernon, Manager
Power Systems Department

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1. PURPOSE AND SCOPE

The purpose of this report is to present the results of the postflight analysis of the Ascent Propulsion System (APS) performance during the Apollo 12 Mission. This report is a supplement to the Apollo 12 Mission Report. Determination of the APS steady-state performance under actual flight environmental conditions was the primary objective of the analysis.

This report includes such information as is required to provide a comprehensive description of APS performance during the Apollo 12 Mission.

Major additions and changes to results as presented in the mission report are listed below:

1) Calculated performance values for the APS burn.

2) Discussion of analysis techniques, problems and assumptions.

3) Comparison of postflight analysis and preflight prediction.

4) Reaction Control Systems (RCS) duty cycle included in APS performance analysis.

5) Revised estimates of propellant consumption.

2. SUMMARY

The duty cycle for the LM-6 APS consisted of one firing, a manned lift-off from the lunar surface. APS performance for this firing was evaluated and found to be satisfactory.

Engine ignition for the APS burn occurred at a ground elapsed time (GET) of 142:03:47.8 (hours:minutes:seconds). Total burn duration was 424.7 seconds with the engine being commanded off at 142:10:52.5. The burn time was approximately 9.8 seconds shorter than was predicted by the Real Time Computing Complex (RTCC) due to higher than expected engine performance, lower stage weight and an error in the predicted vehicle center of gravity location.

Average steady-state engine performance parameters for the burn are as follow:

Thrust - 3497. lbf

Isp - 310.8 sec

Mixture Ratio - 1.608

All performance parameters were well within their respective 3-sigma limits. Calculated engine throat erosion at engine cutoff for LM-6 APS was approximately 3% greater than predicted.

3. INTRODUCTION

The Apollo 12 Mission was the fifth flight, and fourth manned flight of the Lunar Module (LM). The mission accomplsihed the second lunar landing.

Launch from Kennedy Space Center (KSC) occurred at 11:22 a.m. Eastern Standard Time (EST) on 14 November 1969. Following earth orbit insertion, the S-IVB stage was restarted and performed the Translunar Injection (TLI) maneuver at approximately 2-3/4 hours Ground Elapsed Time (GET). CSM-LM docking occurred at approximately 3-1/2 GET. Separation of the docked vehicles from the S-IVB was accomplsihed one hour later. One midcourse correction burn was performed by the Service Propulsion System (SPS) during the translunar phase of the mission. Lunar Orbit Insertion (LOI-1) and Lunar Orbit Circularization (LOI-2) maneuvers were also performed using the SPS. The LOI-1 burn was conducted approximately 83-1/2 hours after launch and the LOI-2 burn occurred slightly more than 4 hours later. The Descent Propulsion System (DPS) duty cycle consisted of two firings: the Descent Orbit Insertion (DOI) burn and the Powered Descent Initiation (PDI) burn. Engine ignition time for the DOI and PDI burns were approximately 109-1/2 hours and 110-1/2 hours GET, respectively. Lunar landing occurred at 110:32:36 (hours:minutes:seconds) GET. Ascent Propulsion System (APS) ignition time for lunar liftoff was 142:03:47.8 GET with engine cutoff being commanded at 142:10:52.5 GET for an APS burn duration of 424.7 seconds. CSM-LM docking was accomplished at approximately 145-1/2 hours GET. After crew and equipment transfers had been effected, the LM was jettisoned. Exact data concerning ascent stage main engine ignition and cutoff times and the associated velocity change are shown in Table 1.

After a separation maneuver using the SM RCS, the LM was maneuvered with its RCS so as to impact on the lunar surface. Lunar impact occurred at approximately 150 hours GET, terminating APS telemetry data.

The Apollo 12 LM-6 APS was equipped with Rocketdyne Engine S/N 0001C. APS engine performance characterization equations used in preflight prediction and as a basis for the postflight analysis are found in Reference 2. Engine acceptance test data used in the determination of performance are from Reference 3. Physical characteristics of the engine and feed system are presented in Table 2.

There were no Apollo 12 Mission detailed test objectives specifically related to the APS.

4. STEADY-STATE PERFORMANCE ANALYSIS

Analysis Technique

Determination of steady-state performance during the manned lunar liftoff burn was the primary objective of LM-6 APS postflight analysis. The burn duration was 424.7 seconds, engine on to engine off command. The lunar liftoff burn was the only firing of the APS during the Apollo 12 Mission.

The APS postflight analysis was conducted using the Apollo Propulsion Analysis Program (PAP) as the primary computational tool. PAP utilizes a minimum variance technique to establish the best correlation between an engine characterization model, derived from ground test data, and selected flight measurements. The program embodies error models for the various flight and ground test data that are used as program inputs and combines these with the empirically derived engine characterization equations. Successive iterations through the program result in estimations of system performance history and weights which "best", in a minimum variance sense, reconcile the available data.

An initial estimate of the ascent stage damp weight at lunar liftoff of 10,750 lbm was obtained from Reference 4. This value was reduced to account for an additional 15 lbm of RCS propellant consumed during an RCS static firing on the lunar surface and further reduced by 21 lbm to account for a component weight change that was discovered after the issuance of Reference 4. Ascent Stage damp weight (total spacecraft weight less APS propellants) was considered to be constant throughout the run, except for a .03 lbm/sec[1] overboard flowrate to account for ablative material eroding from the nozzle.

[1] As furnished by Grumman Aerospace Corporation

RCS propellant usage and thrust histories were obtained from an analysis of the RCS bi-level measurements. All RCS consumption during the ascent burn was from the APS tanks. Table 3 presents a summary of propellant usage, including RCS consumption, from the APS tanks during the ascent burn. Propellant densities used in the program were based on equations from Reference 5, adjusted by measured density data for the LM-6 flight given in the Spacecraft Operational Data Book (SODB), Reference 6. Oxidizer and fuel temperatures were taken from measurement data and were 67.9°F and 68.3°F, respectively. These temperatures were considered to be constant throughout the segment of burn analyzed. The following flight measurement data were used in the analysis of the LM-6 APS burn: engine chamber pressure, engine interface pressures, vehicle thrust acceleration, propellant tank bulk temperatures, helium regulator outlet pressures, engine on-off commands, and RCS thruster solenoid bi-level measurements. Measurement numbers and other data pertinent to the above measurements, with the exception of RCS bi-levels, are given in Table 4. Plots of measurement data versus time are presented in the Appendix to this report.

Flight Data Analysis and Results

A 330-second segment of the APS burn was selected to be analyzed for the purpose of determining steady-state engine performance. APS ignition occurred at a GET of 142:03:47.8 and engine cutoff was commanded at 142:10:52.5 GET. The segment of the burn analyzed begins at 142:04:07.0 GET, 19.2 seconds after ignition, and ends at 142:09:37.0 GET, 75.5 seconds prior to cutoff. An unexplained shift in measured acceleration data at 350 seconds after ignition required that steady-state analysis be terminated at that point. The acceleration data shift did not appear to reflect changes in interface pressures or other system parameters so that satisfactory simulation of the change was not possible. Steady-state analysis of the APS burn revealed no anomalies. APS engine propellant consumption during the burn is presented in Table 3. Propellant consumption from engine on command to the start of the steady-state analysis segment and from the end of the steady-state analysis to the beginning of chamber pressure decay was extrapolated from steady-state analysis results. The primary engine performance determinations made during the LM-6 postflight analysis are as follow. All average values are over the 330-second period of steady-state analysis.

1) Average APS specific impulse was 310.9 seconds.
2) Average APS mixture ratio was determined to be 1.608.
3) Average APS thrust was 3493. lbf.
4) Engine throat erosion was 2% higher than predicted at 350 seconds from ignition (Figure 1).

An extrapolation of the APS steady-state analysis to include the entire burn, with the exception of ignition and shutdown transients, resulted in an average specific impulse, thrust and mixture ratio of 310.8 seconds, 3497. lbf and 1.608 units, respectively. LM-6 APS performance was greater than predicted with average engine specific impulse exceeding the predicted integrated average value by 1.3 seconds.

The general solution approach used in the LM-6 flight evaluation was to calculate a vehicle weight (including propellant loads) for the beginning of the segment of burn used to analyze steady-state performance and then allow the Apollo Propulsion Analysis Program to vary this weight and other selected performance parameters (state variables) in order to achieve an acceptable data match. The PAP simulations were made using the previously discussed APS engine characterization model driven by engine interface pressures. Raw flight interface pressure measurement data were first filtered with a sliding arc filter and then, because of excessive distortion, these data were further smoothed using a fifth degree curve fit. The initial estimates of the interface pressure biases[1] used as input to the program were based on ground test data and were -1.7 psi and -.9 psi for oxidizer and fuel, respectively. Program results determined the biases to be -.9 psi for oxidizer interface pressure and -.8 psi for the fuel interface pressure.

Simulation of RCS activity was accomplished by calculating individual thruster "on" time from the RCS accumulated "on" time data and using this to determine an impulse imparted to the vehicle in the direction of the APS engine thrust vector. This impulse was then converted to an effective thrust over a discrete time interval (10 seconds). RCS propellant flowrates for the same intervals were calculated as a percentage of a nominal consumption of .36 lbm/sec. The percentage of nominal consumption is equivalent to the value of the effective thrust as a percentage of a 100 lbm nominal thrust. RCS propellant consumption was verified by comparing the integrated value obtained from the method described above with the total consumption determined by multiplying total system "on" time by the nominal .36 lbm/sec flowrate. A small adjustment was made to propellant mass overboard to account for consumption of RCS engines in a plane perpendicular to the thrust vector of the APS engine. The resulting thrust and flowrate data were characterized with 5th degree curve fits, as functions of time, and input to PAP. It is apparent that these characterizations do not in general give the calculated instantaneous thrust and flowrates for the RCS thrusters due to the method of calculation and variations in thrust levels for varying engine pulse durations, but over the total time period evaluated they will satisfactorily approximate the total impulse and mass change. At discrete time points when the RCS residuals (curve fit minus calculated data) were excessive, minor adjustments

[1]As a convention in this report, a negative bias indicates that measured data was reading less than its true value.

were made to the RCS thrust and flowrates to reduce the residuals.

Initial PAP simulation results based on the input data outlined above were not acceptable in that the residuals (differences between the filtered flight data and the program calculated values) indicated time correlated errors. The acceleration residuals had a positive slope indicating that an increase in calculated acceleration with flight time was required to minimize the residual error. This effect may be gained by increasing engine flowrates and/or increasing engine thrust on a time basis. The chamber pressure residuals indicated that the measured chamber pressure was biased by -1. to -2. psi. In addition to the bias, the chamber pressure residuals had a negative slope which in combination with the need for an increase in calculated acceleration indicated that a greater than predicted throat erosion rate was necessary. A revised throat erosion curve was calculated using the partial derivatives of throat area with respect to acceleration at ten second intervals throughout the run. The revision of the throat area curve included raising the initial value to 16.48 in^2, about .7% larger than the preflight value. The inclusion of this calculated throat area curve in the analysis program resulted in an excellent acceleration match with a near zero mean and no significant slope. The derived throat erosion curve was 2% greater than predicted at approximately 350 seconds after ignition. Figure 1 shows the calculated throat area curve in comparison with the predicted curve for LM-6.

The chamber pressure match resulting from the inclusion of the calculated throat area curve was not as good as might have been expected. Referring to Figure 3, it can be seen that the chamber pressure residual curve slopes upward for approximately 180 seconds from the start of the steady-state analysis segment and then levels off for the remaining 150 seconds. It was not possible from the flight data to determine the reason for this residual shape; however, it is hypothesized that a time varying bias might be affecting the chamber pressure measurement. Chamber pressure residual data from past flights have exhibited much the same general shape as LM-6 data; however, the upward sloping section of the curve was over a much shorter time period, on the order of 40 to 60 seconds, and was attributed to a minor discrepancy in the engine characterization. It is extremely difficult to determine an exact measurement bias from PAP given a residual shape error of the type seen in the LM-6 data since the program attempts to minimize the residuals, thus distributing them about a zero mean. For this reason the chamber pressure bias for LM-6 could not be more accurately determined than the -1. to -2. psi range previously quoted. Ground test data indicated a chamber pressure measurement bias of -.7 psi. It should be noted from Figure 3 that the chamber pressure residuals are within a ±1 psi band. Residuals are calculated by subtracting biased program calculated data from flight measured data. Calculated chamber pressure for the LM-6 flight reconstruction was adjusted by -1.35 psi prior to calculation of the previously discussed residuals.

The principal indicator of the accuracy of the postflight reconstruction is the matching of calculated and measured acceleration data. A measure of the quality of the match is given by the residual slope and intercept data as shown in Figure 2. These data represent the intercept, on the ordinate, and slope of a linear fit to the residual data. The closer both these numbers are to zero, the more accurate is the match. The acceleration match achieved with the LM-6 postflight reconstruction is very good. A match of measured and calculated engine chamber pressure is given in Figure 3. The LM-6 flight reconstruction was by all indications an accurate simulation of actual flight performance. Residuals between calculated and measured parameters were all within measurement accuracies.

The total propellant residuals at engine cutoff signal from the results of the above data analysis were 215 lbm oxidizer and 148 lbm fuel. Based on these residual propellants, the remaining burn time capability of the ascent stage at APS engine cutoff was approximately 30 seconds, with the engine shutdown then resulting from oxidizer depletion. A shutdown as described would have resulted in 15 lbm of fuel remaining on board.

A vehicle damp weight reduction of 15 lbm was determined from the PAP reconstruction. The best estimate of total ascent stage weight at liftoff is 10,699 lbm.

Figures 2 through 9 show the principal performance parameters associated with the LM-6 postflight analysis. Four flight measurements were used as time varying input to the Propulsion Analysis Program. Two of these measurements, fuel and oxidizer interface pressure, were used as program drivers. The other two, acceleration and chamber pressure, were compared to calculated values by the program's minimum variance technique. The acceleration and chamber pressure measurements along with their residuals are presented in Figures 2 and 3, respectively. Figures 4 and 5 contain oxidizer and fuel interface pressure measurement data as they appeared after smoothing of the raw data, the curve fits of these data that were ultimately input to the Apollo Propulsion Analysis Program, and the residuals between the two data sets. Calculated steady-state values for the following parameters are shown in Figures 6-9: thrust, specific impulse, oxidizer flowrate and fuel flowrate.

Comparison with Preflight Performance Prediction

Predicted performance of the LM-6 APS is presented in References 8 and 9. The intention of the preflight performance prediction was to simulate APS performance under flight environmental conditions for the Mission H1 duty cycle. No attempt was made in the preflight prediction to simulate RCS operation.

Table 5 presents a summary of actual and predicted APS performance during the ascent burn. Measurement data compare quite closely with the reconstructed parameters. Engine specific impulse determined by the post-flight reconstruction is somewhat greater than had been predicted but is still well within the 3 sigma limits of ± 3.5 seconds presented in Reference 9. Comparisons of predicted and reconstructed values for specific impulse, thrust, and mixture ratio are presented in Figure 10 along with related three sigma dispersions. The variations in flight specific impulse, thrust and mixture ratio were within their respective three sigma dispersions.

Engine Performance at Standard Interface Conditions

Expected APS engine flight performance was based on an engine characterization which utilized data obtained during engine and injector acceptance tests. In order to allow actual engine performance variations to be separated from variations induced by feed system, pressurization system, and propellant temperature variations, the acceptance test data is adjusted to a set of standard interface conditions, thereby providing a common basis for comparison. Standard interface conditions are as follows:

Oxidizer interface pressure, psia	170.
Fuel interface pressure, psia	170.
Oxidizer interface temperature, ^{o}F	70.
Fuel interface temperature, ^{o}F	70.
Oxidizer density, lbm/ft^3	90.21
Fuel density, lbm/ft^3	56.39
Thrust acceleration, lbf/lbm	1.
Throat area, in^2	16.47

Analysis results (at 13 seconds from ignition) for the ascent burn corrected to standard interface conditions and compared to acceptance test values are shown below:

	Acceptance Test Data	Flight Analysis Results	% Difference
Thrust, lbf	3492.	3503	.3
Specific Impulse, $\frac{\text{lbf-sec}}{\text{lbm}}$	309.6	311.0	.4
Propellant Mixture Ratio	1.601	1.601	.0

Reduction of engine performance to standard interface conditions and comparison with acceptance test values shows good agreement with the largest difference being in the engine specific impulse. All differences are within two standard deviations of acceptance test values. This indicates that basic preflight prediction techniques are adequate, however, since greater than predicted performance has been noted in the LM-3, LM-4, LM-5 and LM-6 flight performance results, the present prediction techniques may be somewhat conservative.

It should be noted that due to the limited number of flight measurements available for use in determining APS propulsion system performance, it is not possible to independently determine engine and/or feed system resistance variations. As an example, given a system mixture ratio shift, it would not be possible to determine if the shift were attributable to the engine or feed system alone or was a result of the interaction of the two. It is apparent, therefore, that the adjustment of feed system data to standard engine inlet conditions could conceivably mask actual engine perturbations.

5. PRESSURIZATION SYSTEM

Helium Utilization

The helium storage tanks were loaded to a nominal 13.2 lbm. There was no indication of helium leakage during the mission and calculated usage agrees well with analytical predictions.

Ullage Pressure Decay During Coast

Decay of the propellant tank ullage pressures is observed indirectly through the fuel and oxidizer interface pressures which at launch were 151 and 127 psia, respectively. At approximately 90 hours GET these pressures had, as expected, decayed to 141 and 105 psia, respectively. This pressure drop is attributed to absorption of helium into the propellants. Pre-ignition pressurization of the propellant tank ullages was evidenced by the increase in both interface pressures to a value of approximately 185 psia at 141:44 hours GET.

Ullage Pressure Following the APS Burn

During the lunar orbit following APS cutoff, both interface pressures quickly increased from their respective flow pressures to lock-up pressures of approximately 181 psi and then continued to increase by a total of about 13 psi on the oxidizer side and about 6.5 psi on the fuel side. Approximately twenty minutes after shutdown with the interface pressures at 194 psi for oxidizer and 187.5 psi for fuel, loss of signal occurred as the vehicle went behind the moon. At re-acquisition of the telemetry signal some 50 minutes later, the oxidizer interface pressure had dropped to a level of 188 psia and the fuel interface pressure had risen to a level of 189.5 psia. Regulator outlet pressure during this period was essentially constant, so there was no indication of leakage. A similar pressure drop was observed during the LM-5 flight, however, during that flight both pressures showed a decrease. This phenomenon, while not explained, had no effect on APS performance or crew safety.

Helium Regulator Performance

Oscillations were noted in the data from both helium regulator outlet pressure measurements. The oscillations were approximately 6 psi, peak to peak, and 19 psi, peak to peak, in measurements GP0025P and GP0018P, respectively. A detailed study of the oscillations appearing in the LM-6 data was conducted by Grumman Aerospace Corporation (GAC) (Reference 10). It was concluded from that study, and available flight data, that LM-6 APS performance was not degraded due to the oscillations. The exact cause of the oscillations has not been determined, however, interaction between the check valves and the regulator is discussed in Reference 10 as a possibility. GAC has also hypothesized that the transducers amplify the oscillations but to varying degrees due to differences in the tap line geometries. Similar oscillations were noted in the LM-4 APS post-flight analysis (Reference 11) with no degradation in APS performance.

6. PROPELLANT LOADING AND USAGE

APS propellant loads for the LM-6 Mission were 3223.7 lbm of oxidizer and 2012.1 lbm of fuel. Of these amounts 34.2 lbm of oxidizer and 14.7 lbm of fuel are considered to be unusable or consumed during transient engine operation. The amounts of nominally deliverable propellants are, therefore, 3189.5 lbm and 1997.4 lbm for oxidizer and fuel, respectively. Propellant density samples taken at the time of loading showed an oxidizer density of 1.424 gm/cc at $4^{o}C$ and 14.7 psia and a fuel density of 0.8994 gm/cc at $25^{o}C$ and 14.7 psia.

Since all RCS propellant usage was from the RCS tanks prior to lunar liftoff, the APS propellant loads at APS ignition were 3223.7 lbm of oxidizer and 2012.1 lbm of fuel. All RCS consumption during the ascent burn was through the APS/RCS interconnect. Total propellant usage from the APS tanks is presented in Table 3. The APS consumption during the lunar liftoff burn was 2944 lbm, oxidizer and 1832 lbm, fuel. Total RCS consumption during the APS burn was 97.1 lbm. A total of 215 lbm of oxidizer and 148 lbm of fuel remained onboard at APS cutoff.

APS Overburn

An overburn of approximately 30 ft/sec was noted at the end of the lunar liftoff burn (References 12 and 13). The LM Guidance Computer (LGC) commanded engine cutoff at the proper time but, because of the failure of the LM commander to put the engine arm switch in the "Off" position prior to the LGC command, actual engine cutoff occurred about 1.7 seconds late. Measured acceleration just prior to cutoff was 19.3 ft/sec^2. Reference 13 discusses the possibility that part of the overburn might be attributed to a greater than predicted APS tailoff ΔV, however, a detailed examination of available APS flight data indicates that the shutdown impulse was near nominal. The overburn was easily nulled by an RCS maneuver.

Burn Time

APS burn time for the LM-6 mission was predicted to be 7 minutes 10 seconds and the burn time estimate by the RTCC just prior to liftoff was 7 minutes 12 seconds. The LGC computed burn time was 7 minutes 3 sec; nine seconds less than the burn time computed by the RTCC. It should be noted that the actual burn time was 7 minutes 4.7 seconds or 1.7 seconds longer than the LGC time due to the previously discussed overburn. The 9-second deviation is well within the 3 sigma dispersion but it represents a confirmation that the thrust-to-weight ratio was higher than expected. LM-6 weight, as determined from the PAP postflight analysis, was 91 pounds less than the real time estimate of 10,790 lbm. Average APS thrust for the lunar liftoff burn was 27 lbf greater than the predicted average value. Based on sensitivity coefficients from Reference 14, these variations would result in burn time being reduced by approximately 7.2 seconds. In addition, Reference 15 details a suspected deviation in the Z axis component of the vehicle center of gravity (ZCG) of approximately -.5 inches, which, again based on sensitivity coefficients from Reference 14, would further reduce burn time by some 3 seconds. The shorter than predicted burn time is, therefore, explained by deviations from predicted weight, thrust and ZCG location.

8. CONCLUSIONS

APS flight performance reconstructions for LM-3, LM-4, LM-5 and LM-6 indicate that engine performance prediction techniques may be somewhat conservative. The cause of the difference between predicted and reconstructed performance values has not been determined but will be the subject of further investigation.

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5. NASA Memorandum EP-23-10-69, from EP/2 Head Development Section to EP/2 Chief, Primary Propulsion Branch, "Propellant Densities (N_2O_4 and A-50)", 18 February 1969.

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10. Grumman Aerospace Corporation LM Engineering Memorandum LMO-271-855, "LM-6 Regulator Oscillations," 12 January 1970.

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12. TRW IOC 70.7254.4-7, "Apollo 12 DAP Postflight Analysis", 20 January 1970.

13. NASA Memorandum 70-RM22-1, From FM/2 Landing Analysis Branch to FM/13 Mission Planning Support Office, "Apollo 12 LM Ascent Postflight Analysis, 7 January 1970.

14. NASA Internal Note 69-FM-166, "Apollo 11 (Mission G) Dispersion Analysis, Volume IV, Descent and Ascent Dispersion Analyses, Part 2, Lunar Ascent", 20 June 1969.

15. TRW Report 70.6521.2.1-21, "LM-5 Ascent Stage Center of Gravity," MSC/TRW Task 6H, prepared by R. S. West, 4 March 1970.

TABLE 1 - LM-6 APS DUTY CYCLE

BURN	Ignition FS-1 Hr:Min:Sec GET	Engine Cutoff FS-2 Hr:Min:Sec GET	Burn Duration Seconds	Velocity[1] Change Ft/Sec
APS Lunar Liftoff	142:03:47.8	142:10:52.5	424.7	6076.

[1] Calculated by integrating combined APS, RCS thrust acceleration.

TABLE 2

LM-6 APS ENGINE AND FEED SYSTEM PHYSICAL CHARACTERISTICS

Engine[(1)]	
Engine No.	Rocketdyne S/N 0001C
Injector No.	Rocketdyne S/N 4097716
Initial Chamber Throat Area (in.^{2})	16.358 [(4)]
Nozzle Exit Area (in.^{2})	748.959
Initial Expansion Ratio	45.785
Injector Resistance (lb_f-sec^2/lb_m-ft^5)@ time zero and 70°F	
Oxidizer	12832.
Fuel	20646.
Feed System	
Total Volume (pressurized, check valves to engine interface) (ft^3) [(2)]	
Oxidizer	36.94
Fuel	37.02
Resistance, Tank Bottom to Engine Interface (lb_f-sec^2/lb_m-ft^5) at 70°F [(3)]	
Oxidizer	2396.
Fuel	4008.

(1) Rocketdyne Log Book, "Acceptance Test Data Package for Rocket Engine Assembly - Ascent LM - Part No. RS000580-001-00, Serial No. 0001", 30 August 1968.

(2) Per Telecon P. E. Cota, MSC Propulsion, 1 August 1969.

(3) Per telecon L. Rothenberg, GAEC Propulsion, 24 July 1969.

(4) The initial throat area determined from postflight reconstruction was 16.48 in^2.

TABLE 3 - PROPELLANT CONSUMPTION FROM ASCENT PROPULSION SYSTEM TANKS

Event/Usage	Time (GET) hr:min:sec	Consumed (LBM)		Remaining (LBM)	
		Oxidizer	Fuel	Oxidizer	Fuel
Liftoff	00:00:00.7	--	--	3223.7	2012.1
Ignition for Ascent Lunar Liftoff	142:03:47.8	--	--	3223.7	2012.1
RCS Usage During Ascent Burn		64.7	32.4	3159.0	1979.7
APS Consumption		2944.0	1831.5	215.0	148.2
APS Engine Cutoff	142:10:52.5	--	--	215.0	148.2

TABLE 4

FLIGHT DATA USED IN STEADY-STATE ANALYSIS

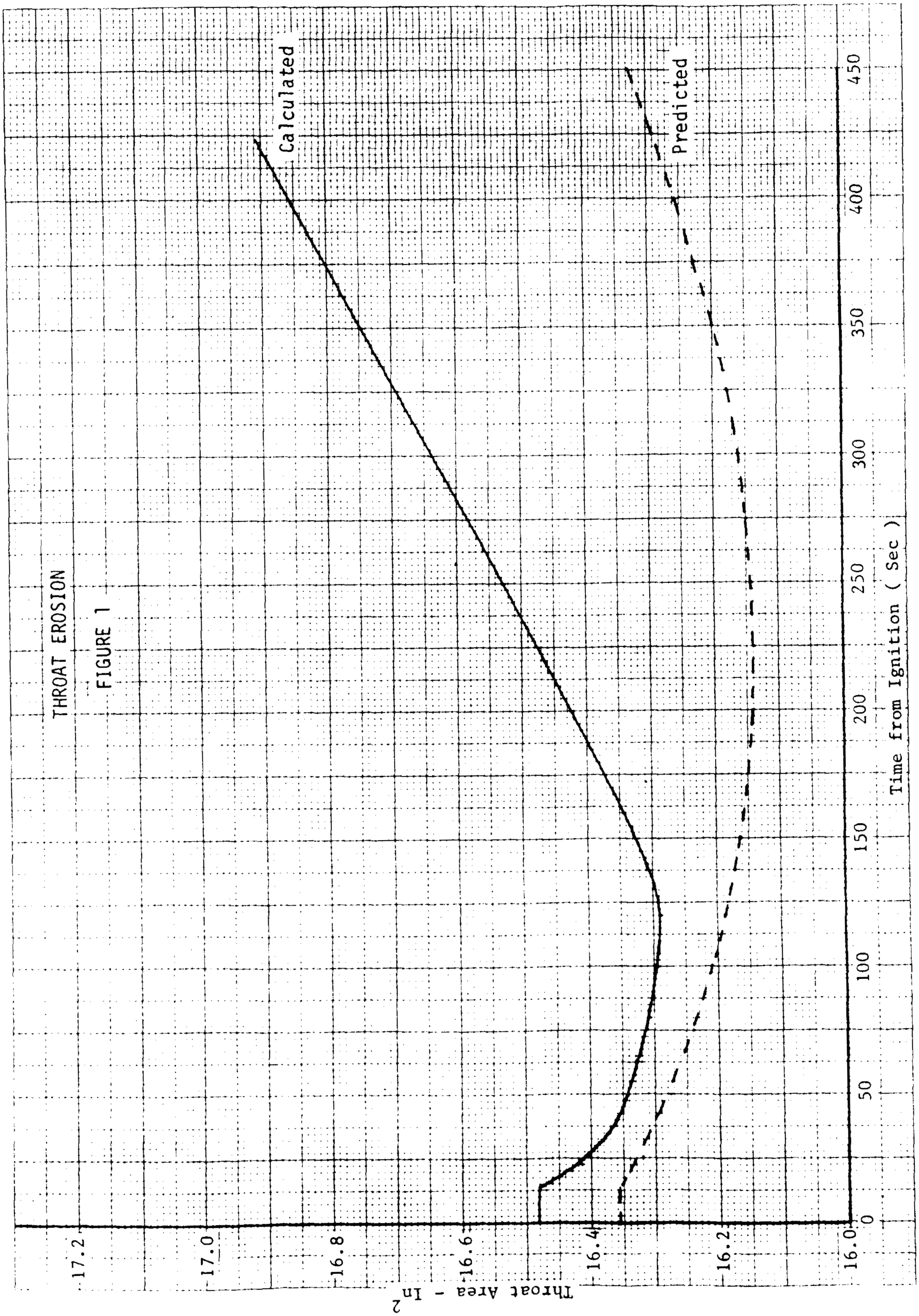
Measurement Number	Description	Range	Sample Rate Sample/sec
GP2010P	Pressure, Thrust Chamber	0-150 psia	200.
GP1503P	Pressure, Engine Oxidizer Interface	0-250 psia	1
GP1501P	Pressure, Engine Fuel Interface	0-250 psia	1
GP0025P	Pressure, Regulator Outlet Manifold	0-300 psia	1
GP0018P	Pressure, Regulator Outlet Manifold	0-300 psia	1
GP1218T	Temperature, Oxidizer Tank Bulk	20-130°F	1
GP0718T	Temperature, Fuel Tank Bulk	20-130°F	1
GH1260X	Ascent Engine On/Off	Off-On	50
CG0001X*	PGNS Downlink Data	Digital Code	50

*Acceleration determined from PIPA data

TABLE 5 - LM-6 APS STEADY-STATE PERFORMANCE

PARAMETER	30 Sec after Ignition			220 Sec after Ignition			300 Sec after Ignition		
	Pred. (a)	Reconstructed (b)	Measured (c)	Pred. (a)	Reconstructed (b)	Measured (c)	Pred. (a)	Reconstructed (b)	Measured (c)
Regulator Outlet Pressure (psia)	184.	-----	184.8 (d)	184.	-----	183.8 (d)	184.	-----	183.8 (d)
Oxidizer Bulk Temperature °F	69.9	-----	67.9	69.6	-----	67.9	69.4	-----	67.9
Fuel Bulk Temperature °F	69.7	-----	68.3	69.6	-----	68.3	69.6	-----	68.3
Oxidizer Interface Pressure, psia	171.2	170.1	169.4	170.9	169.3	168.4	170.6	169.0	168.2
Fuel Interface Pressure, psia	170.7	169.6	169.0	170.4	168.8	168.0	170.2	168.6	168.0
Engine Chamber Pressure, psia	123.2	122.3	120.3	123.5	121.7	120.7	123.3	120.7	120.0
Mixture Ratio	1.610	1.612	-----	1.605	1.608	-----	1.604	1.605	-----
Thrust, lbf	3492.	3504.	-----	3465.	3489.	-----	3460.	3504.	-----
Specific Impulse, sec.	309.6	311.2	-----	309.7	311.0	-----	309.4	310.2	-----

(a) Preflight prediction based on acceptance test data and assuming nominal system performance.
(b) Reconstruction minimum variance technique.
(c) Smoothed flight data without biases determined by postflight analysis.
(d) Regulator outlet data estimated from oscillating measurement.



THROAT EROSION

FIGURE 1

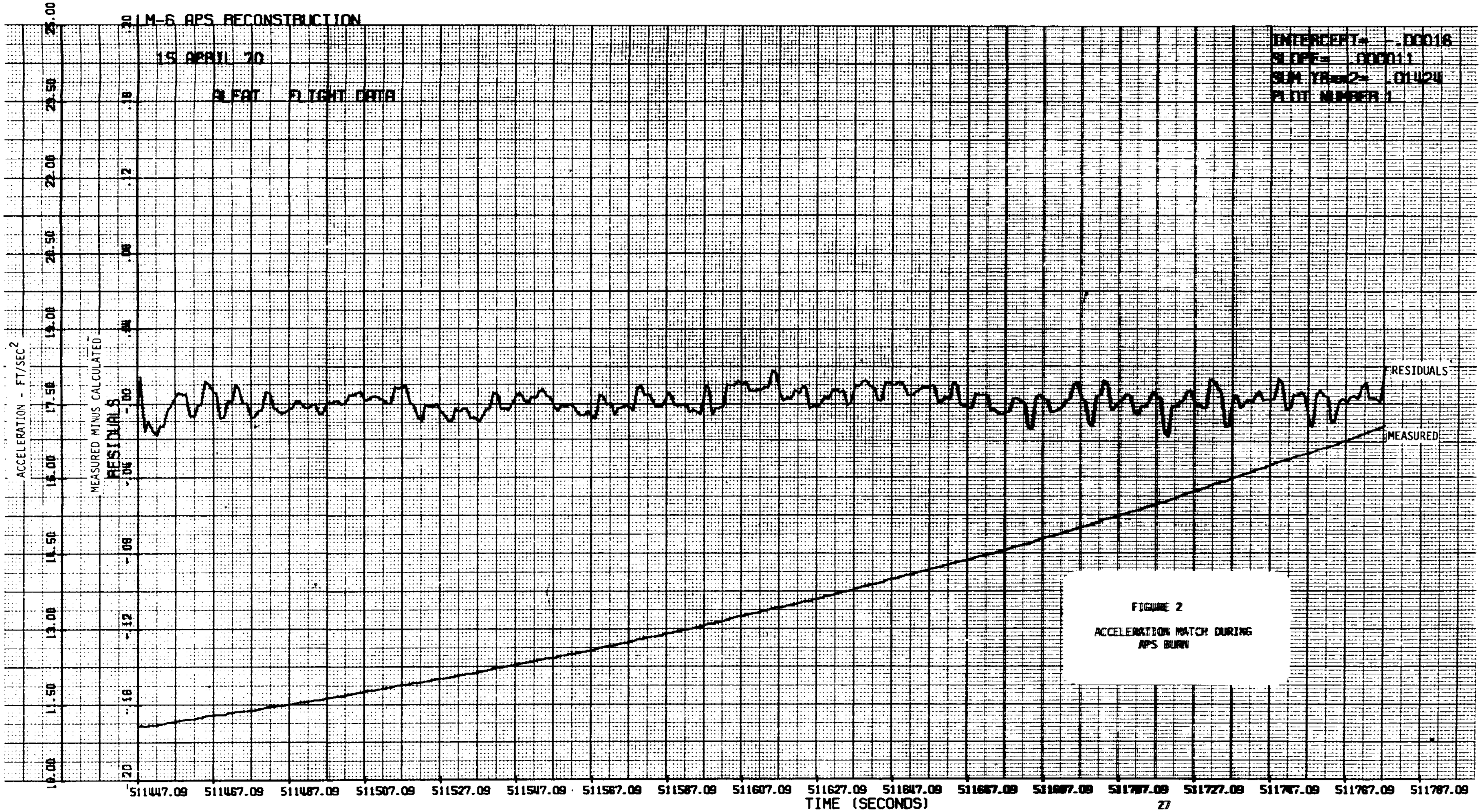


FIGURE 2

ACCELERATION MATCH DURING APS BURN

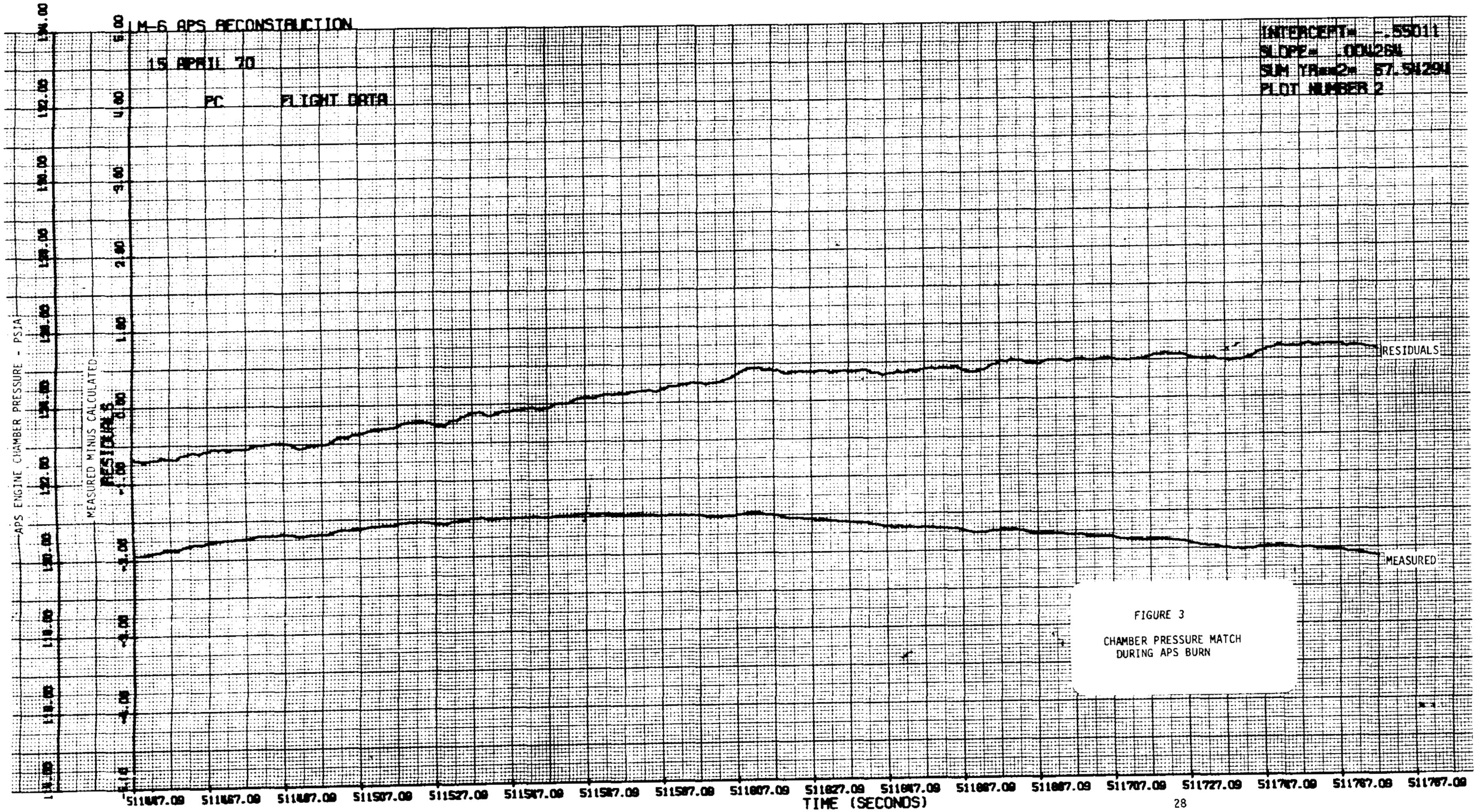


FIGURE 3

CHAMBER PRESSURE MATCH DURING APS BURN

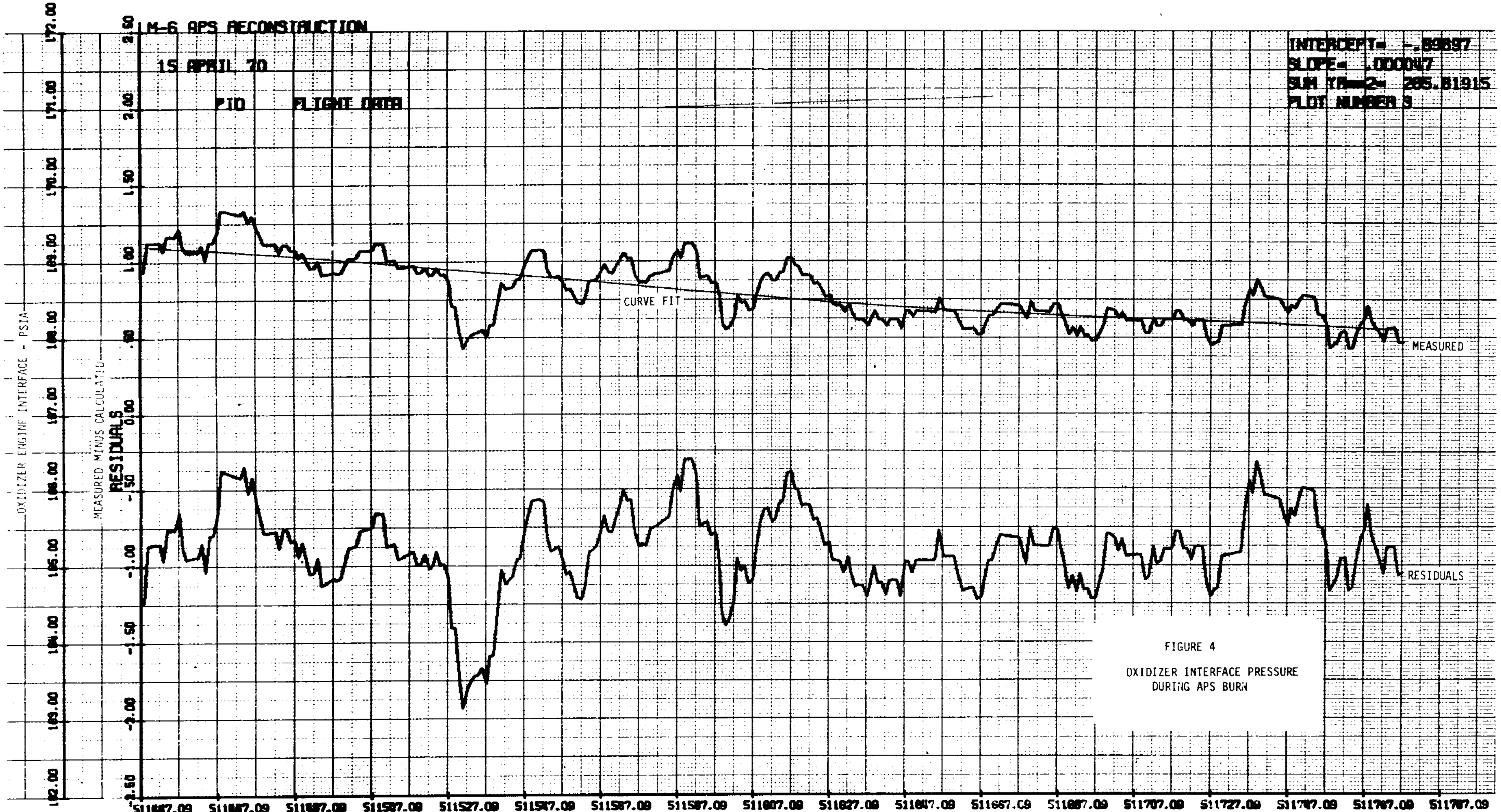


FIGURE 4

OXIDIZER INTERFACE PRESSURE DURING APS BURN

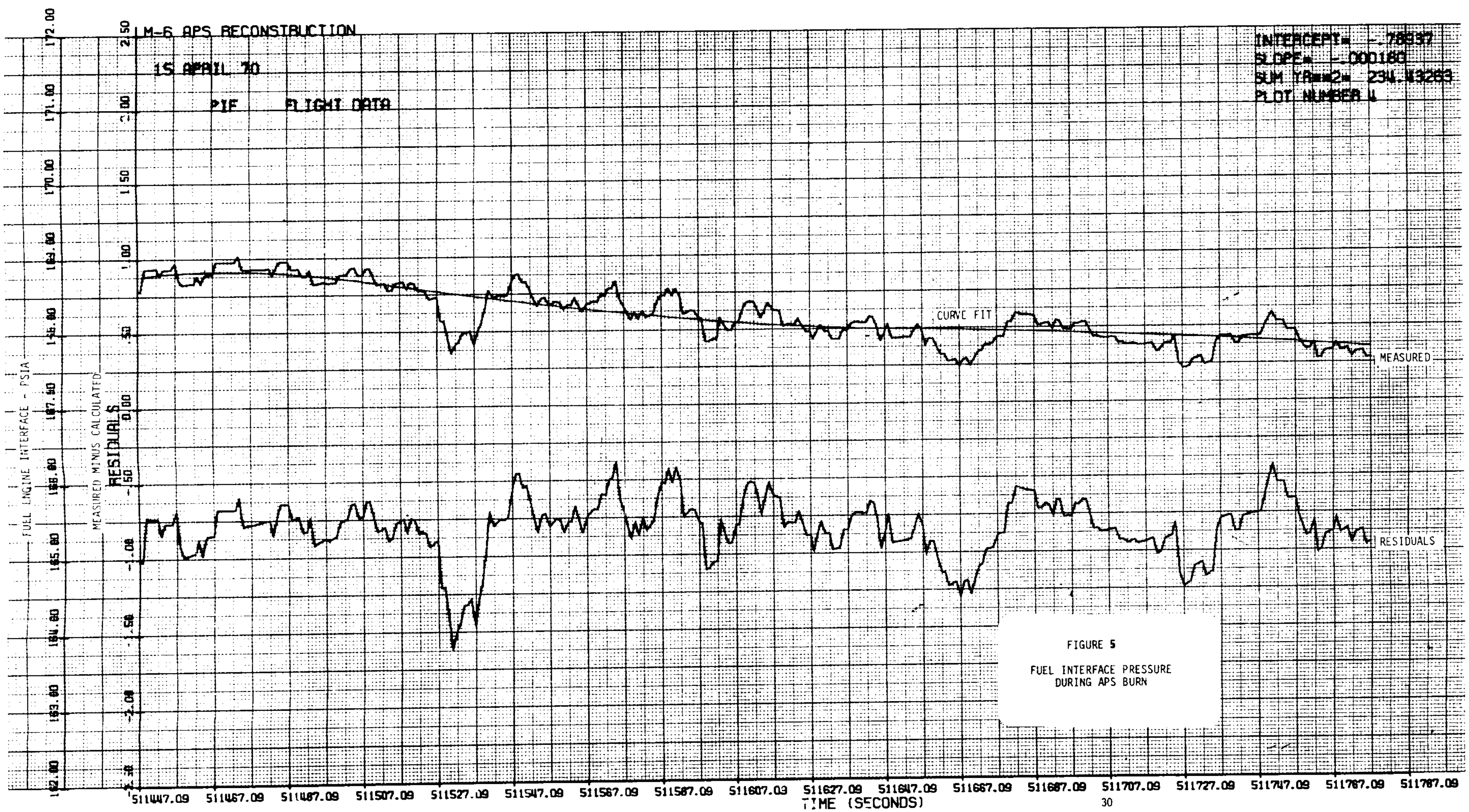


FIGURE 5

FUEL INTERFACE PRESSURE DURING APS BURN

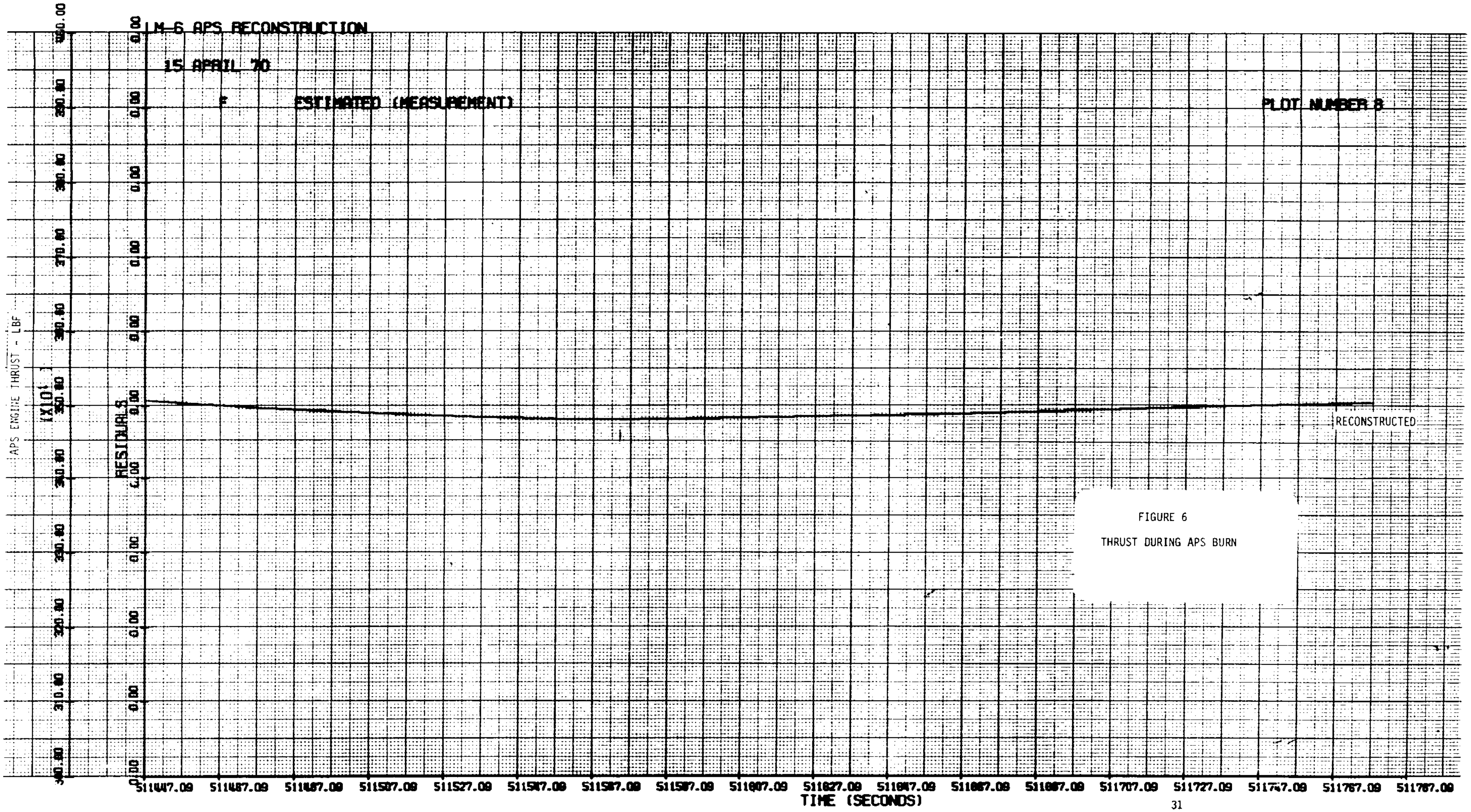


FIGURE 6

THRUST DURING APS BURN

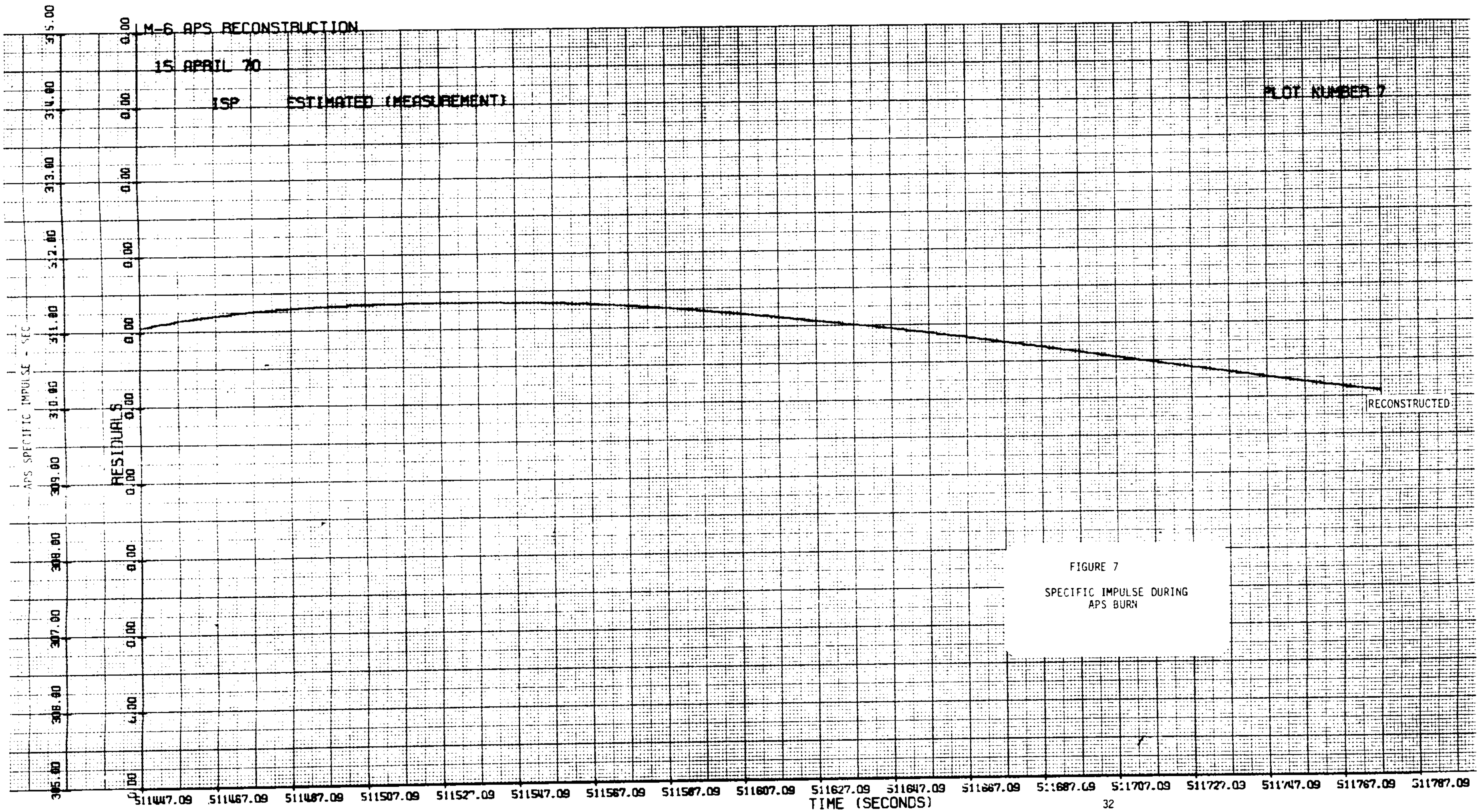


FIGURE 7

SPECIFIC IMPULSE DURING APS BURN

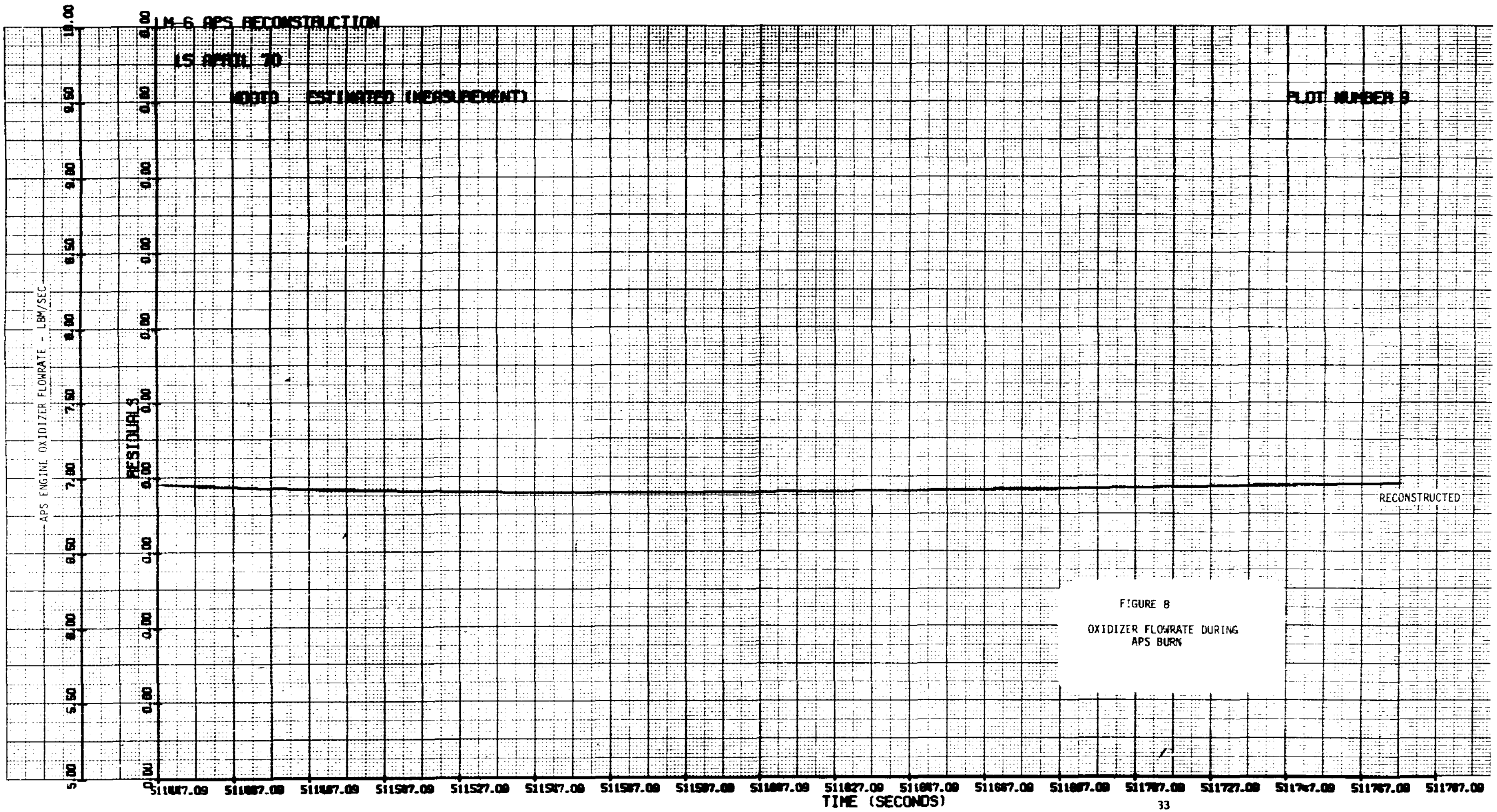


FIGURE 8

OXIDIZER FLOWRATE DURING APS BURN

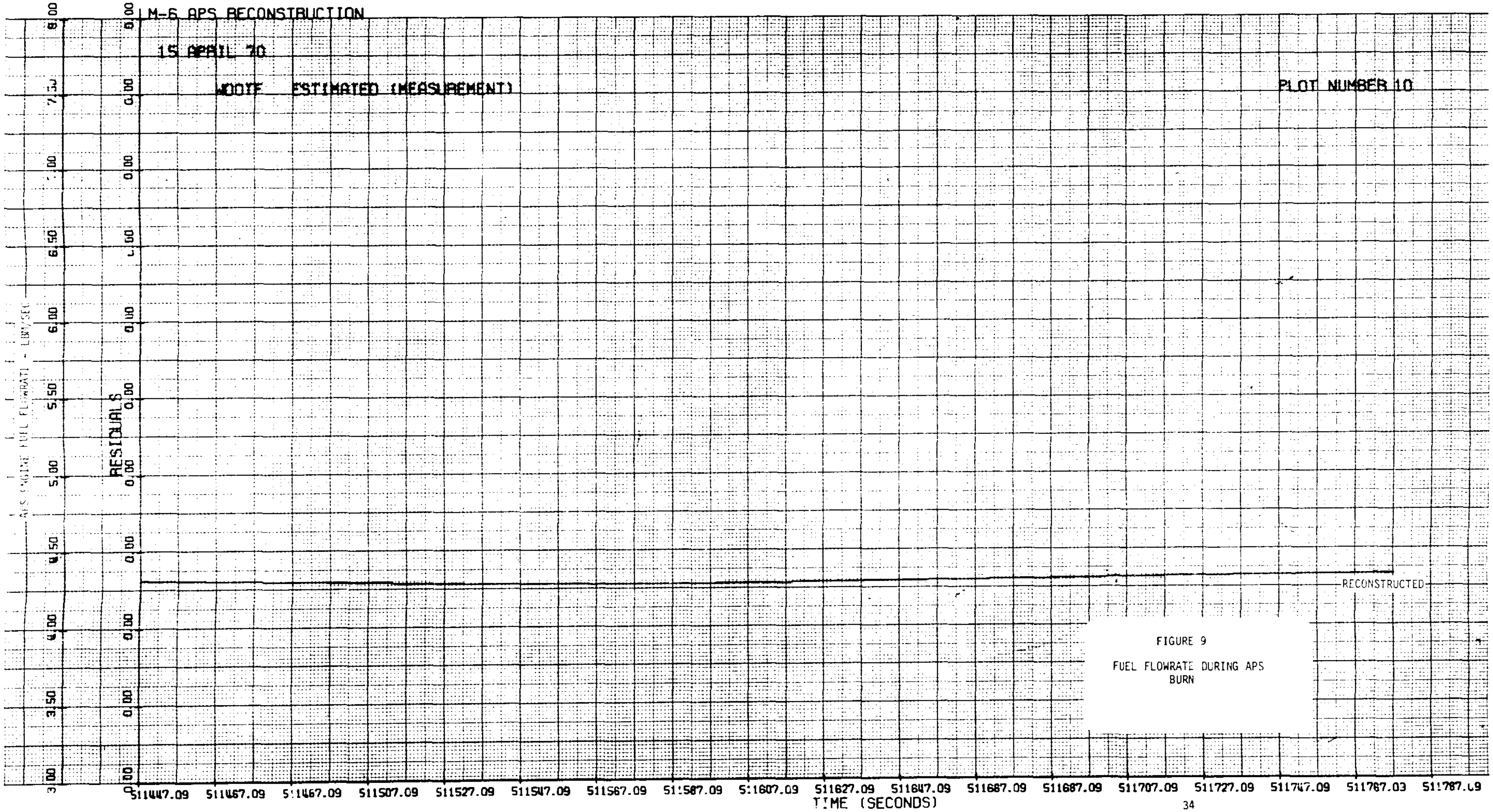
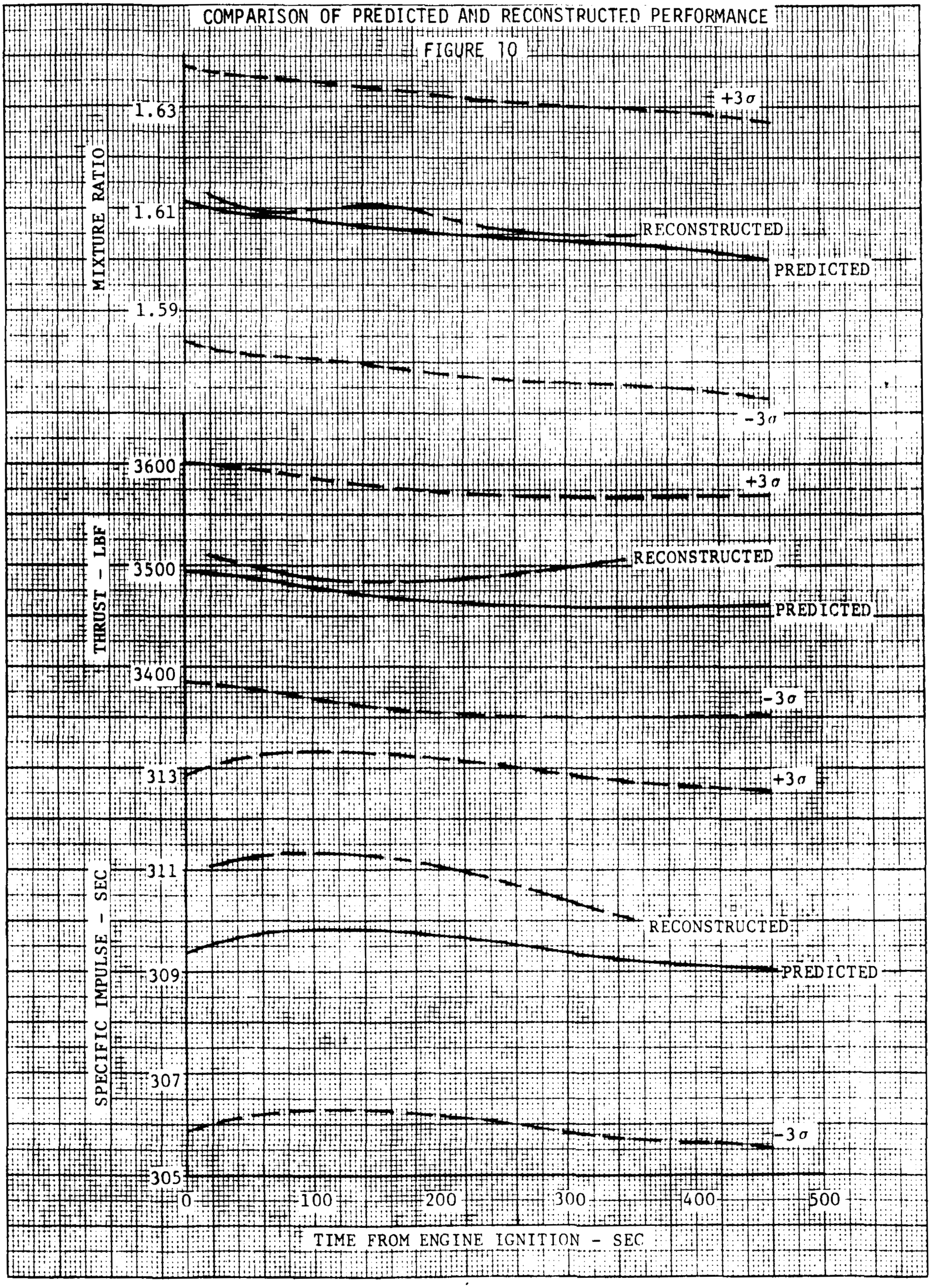


FIGURE 9

FUEL FLOWRATE DURING APS BURN


COMPARISON OF PREDICTED AND RECONSTRUCTED PERFORMANCE
FIGURE 10
MIXTURE RATIO
1.63
1.61
1.59
+3σ
RECONSTRUCTED
PREDICTED
-3σ
THRUST - LBF
3600
3500
3400
+3σ
RECONSTRUCTED
PREDICTED
-3σ
SPECIFIC IMPULSE - SEC
313
311
309
307
305
+3σ
RECONSTRUCTED
PREDICTED
-3σ
0
100
200
300
400
500
TIME FROM ENGINE IGNITION - SEC

APPENDIX

FLIGHT DATA

APOLLO 12 SC108/LM6-APS-(RAW DATA),BURN1,INSERTION

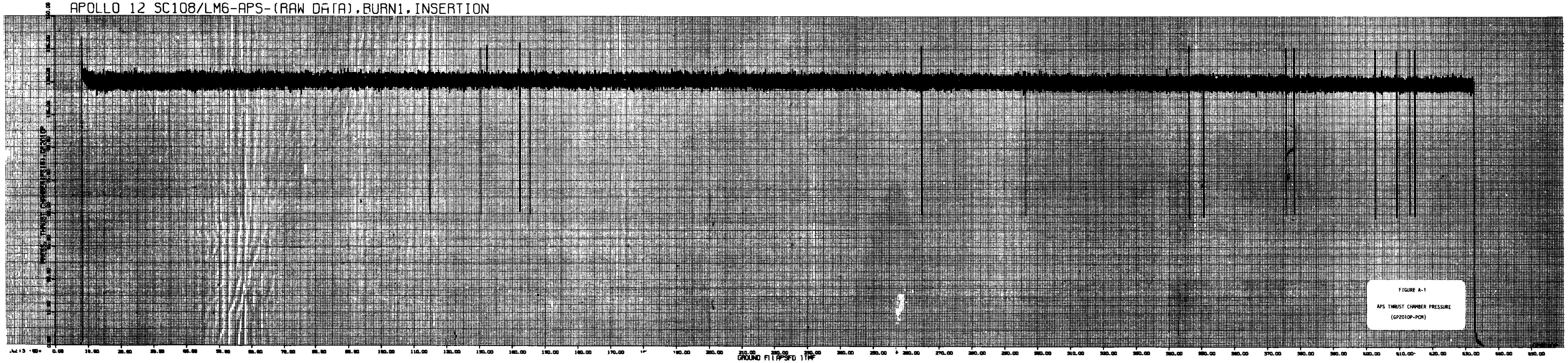


FIGURE A-1

APS THRUST CHAMBER PRESSURE

(GP2010P-PCM)

APOLLO 12 SC108/LM6-APS-(RAW DATA),BURN1,INSERTION

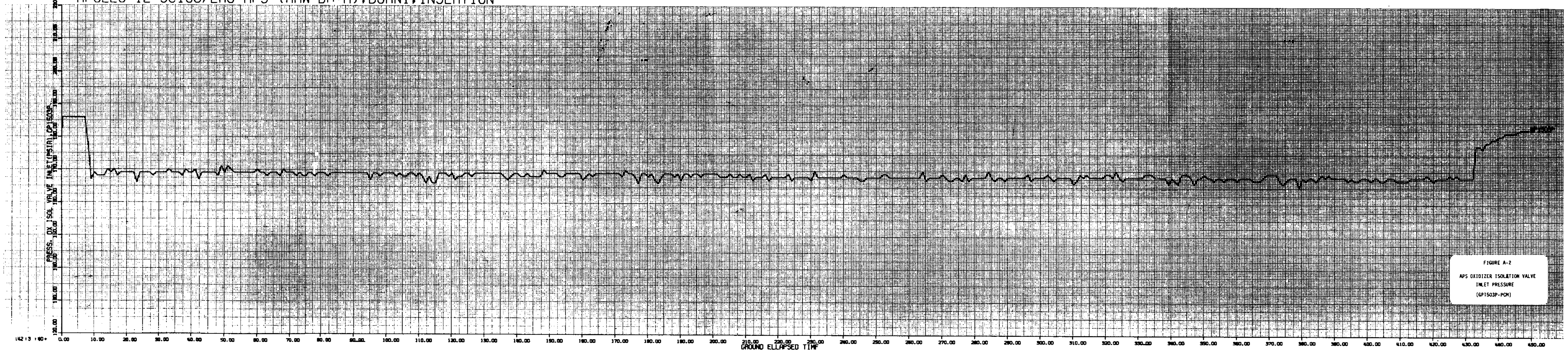


FIGURE A-2

APS OXIDIZER ISOLATION VALVE
INLET PRESSURE
(GP1503P-PCM)

APOLLO 12 SC108/LM6-APS-(RAW DATA),BURN1,INSERTION

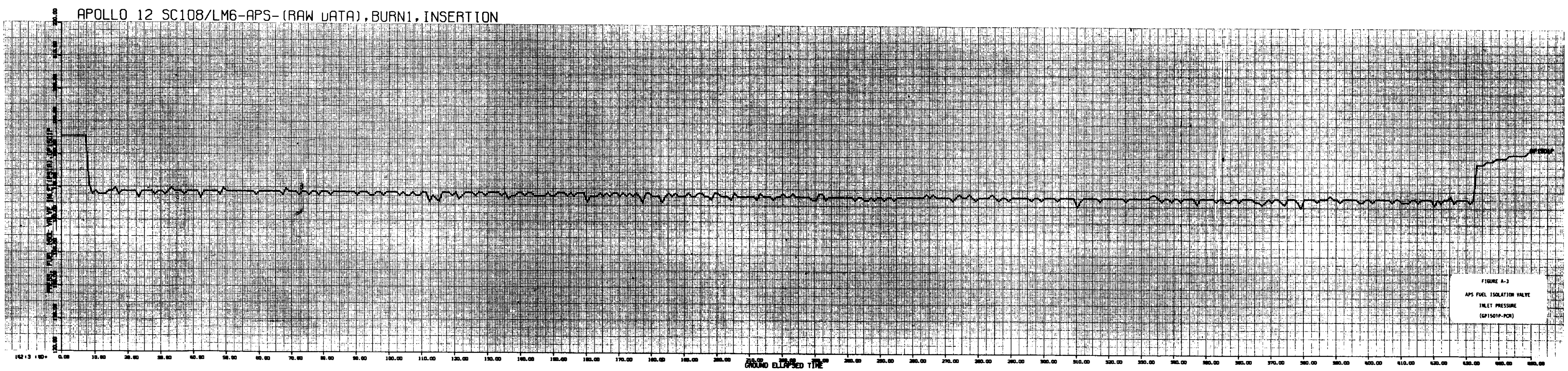


FIGURE A-3
APS FUEL ISOLATION VALVE
INLET PRESSURE
(GP1501P-PCM)

APOLLO 12 SC108/LM6-APS-(RAW DATA),BURN1,INSERTION

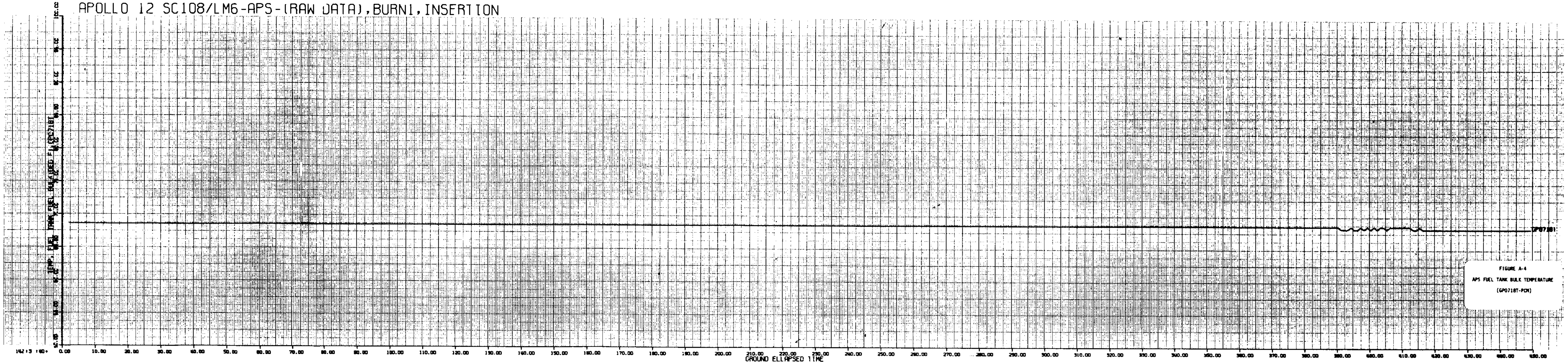


FIGURE A-4

APS FUEL TANK BULK TEMPERATURE

(GP0718T-PCM)

APOLLO 12 SC108/LM6-APS-(RAW DATA),BURN1,INSERTION

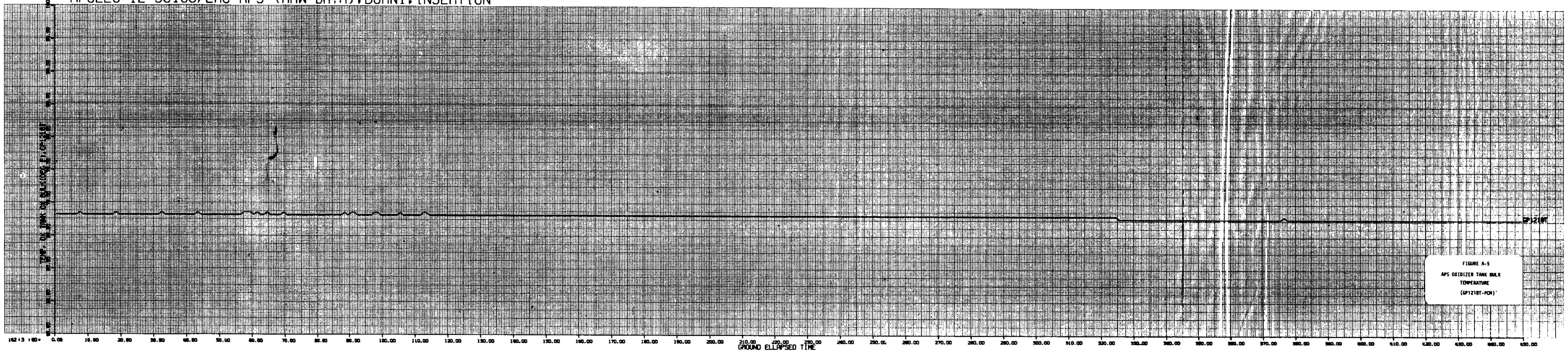


FIGURE A-5

APS OXIDIZER TANK BULK TEMPERATURE

(GP1218T-PCM)

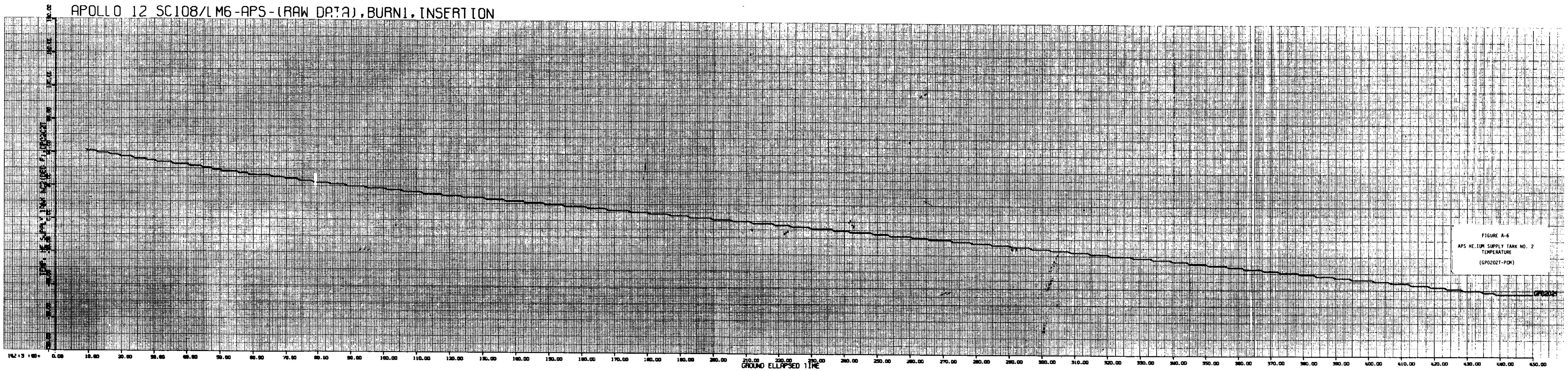


FIGURE A-6

APS HELIUM SUPPLY TANK NO. 2
TEMPERATURE

(GP0202T-PCM)

APOLLO 12 SC108/LM6-APS-(RAW DATA),BURN1,INSERTION

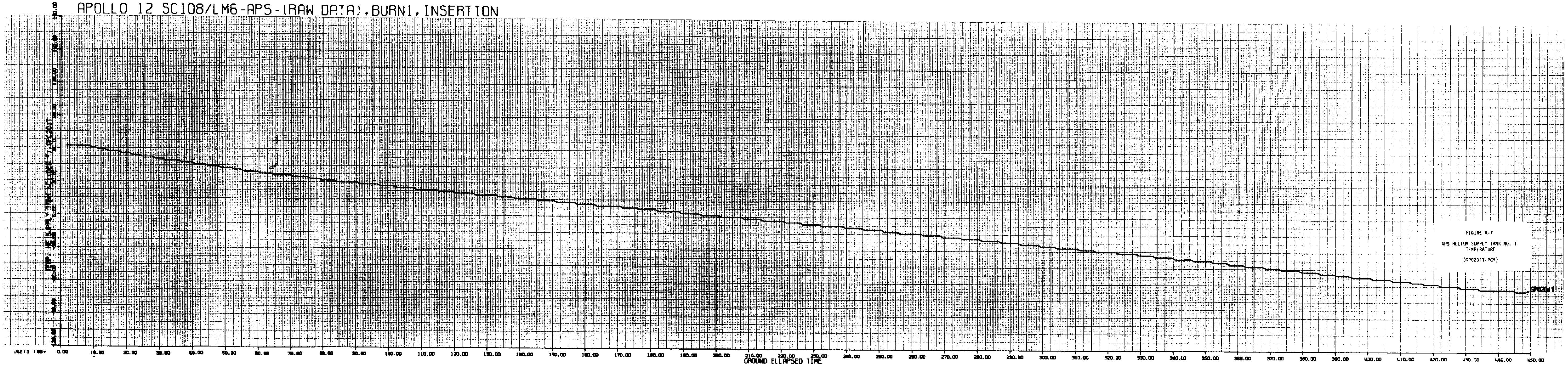


FIGURE A-7

APS HELIUM SUPPLY TANK NO. 1
TEMPERATURE

(GP0201T-PCM)

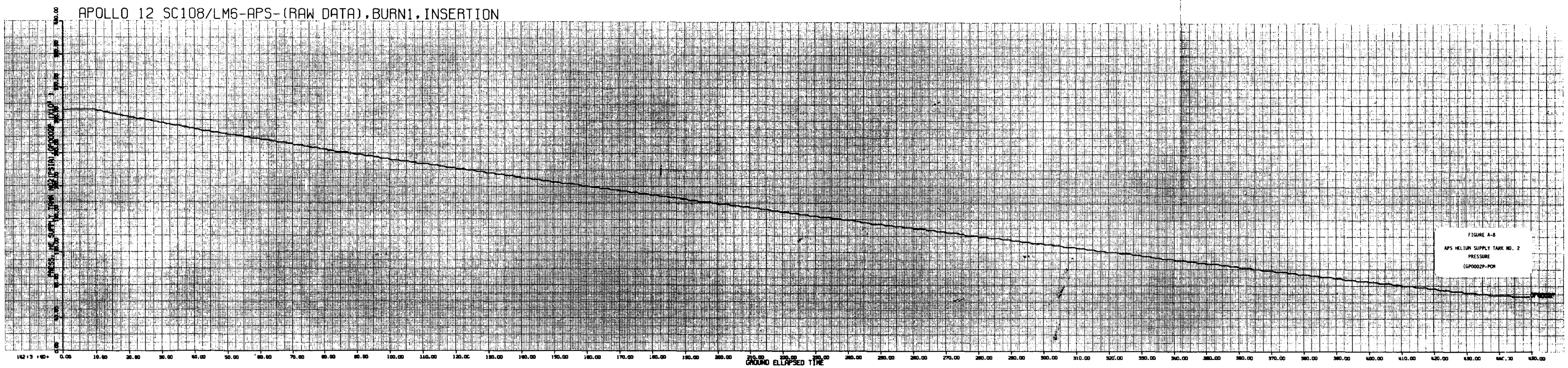


FIGURE A-8
APS HELIUM SUPPLY TANK NO. 2
PRESSURE
(GP0002P-PCM

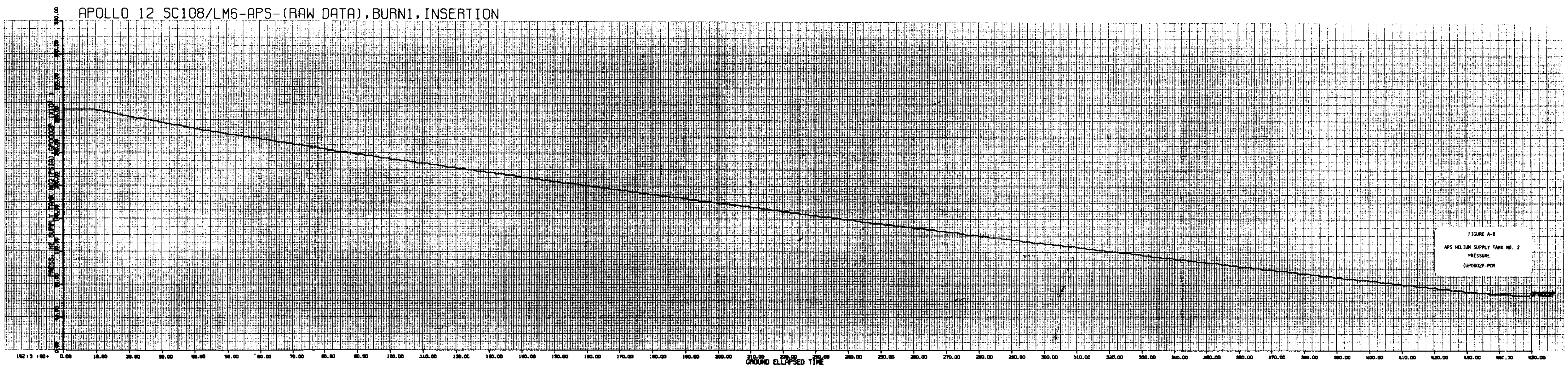


FIGURE A-8

APS HELIUM SUPPLY TANK NO. 2
PRESSURE
(GP0002P-PCM

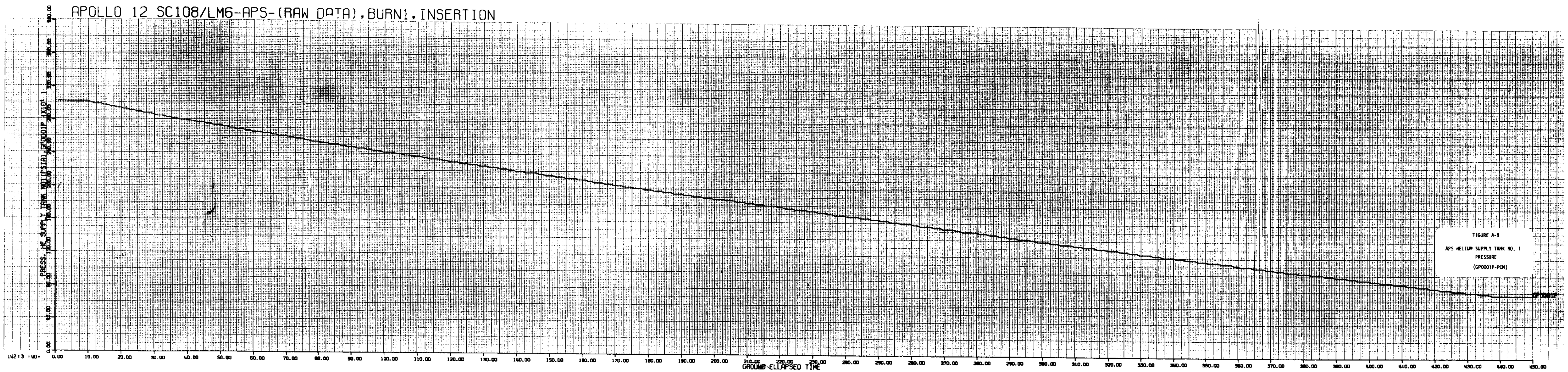


FIGURE A-9

APS HELIUM SUPPLY TANK NO. 1
PRESSURE
(GP0001P-PCM)

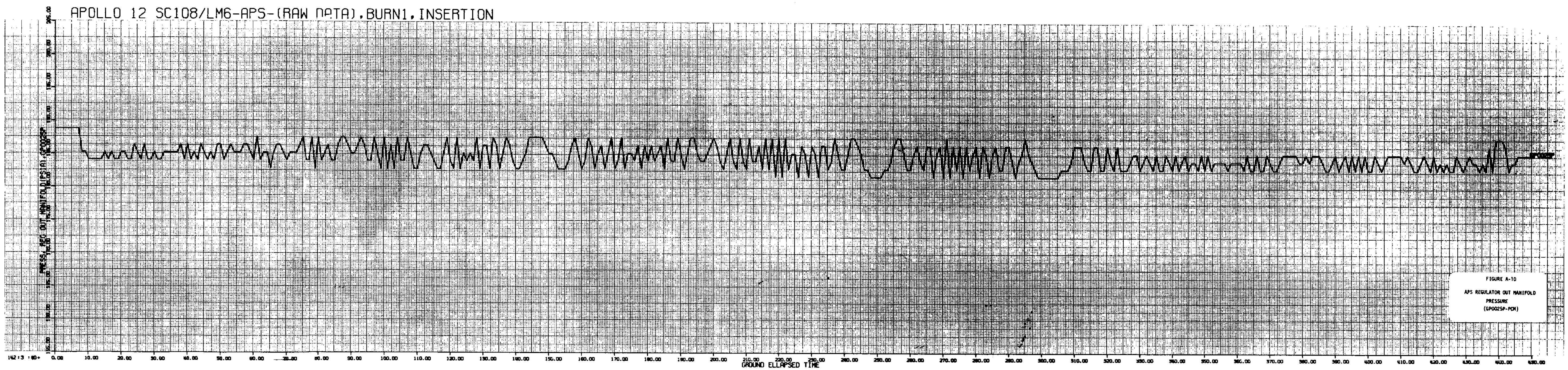


FIGURE A-10
APS REGULATOR OUT MANIFOLD PRESSURE (GP0025P-PCM)

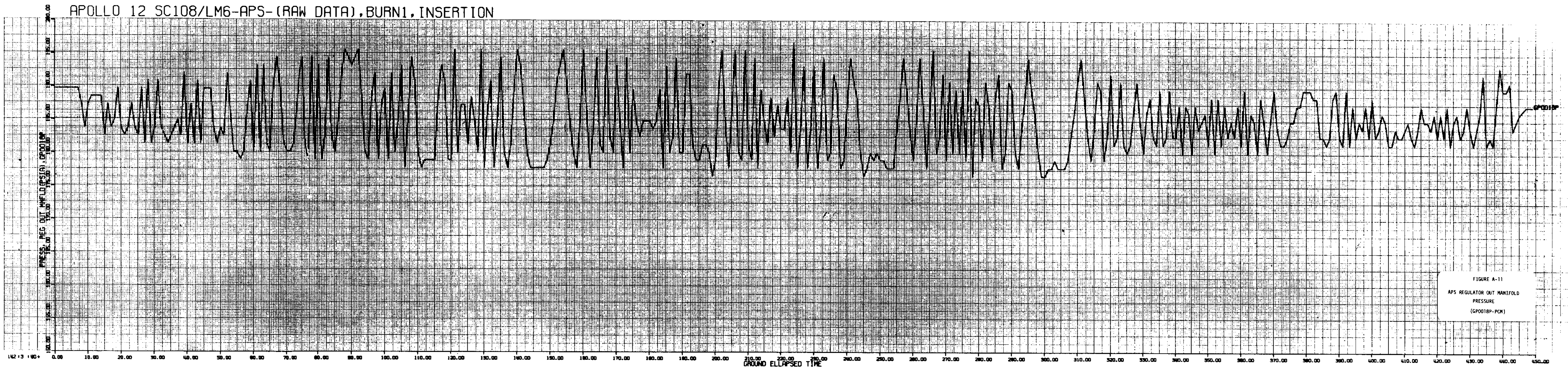


FIGURE A-11

APS REGULATOR OUT MANIFOLD PRESSURE

(GP0018P-PCM)