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PROJECT TECHNICAL REPORT

APOLLO 10

LM-4

ASCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

NAS 9-8166

16 September 1969

Prepared for
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

Prepared by
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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 10 Mission. Determination of the APS steady-state performance under actual flight environmental conditions was the primary objective of the analysis. No formal analysis of APS transient performance was made.

Preliminary postflight analysis of APS performance is documented in Reference 1. This report supersedes Reference 1, and includes such additional information as is required to provide a comprehensive description of APS performance during the Apollo 10 Mission.

Major additions and/or changes to results as presented in Reference 1 are listed below:

1) Revised performance values for the APS BTD.
2) Discussion of analysis techniques, problems and assumptions.
3) Comparison of postflight analysis and preflight prediction.
4) Discussion of the LM-4 propellant settling anomaly.

2. SUMMARY

The duty cycle for the LM-4 Ascent Propulsion System (APS) consisted of two engine firings; a short manned burn and a longer unmanned burn to propellant depletion (BTD). APS performance for these two firings was evaluated and found to be satisfactory.

Engine ignition for the manned burn occurred at a ground elapsed time (GET) of 102:55:02.1 (hours:minutes:seconds) with the engine being commanded off at 102:55:17.7 GET for a total burn duration of 15.6 seconds. The unmanned BTD was initiated at a GET of 108:52:05.5 and was terminated, as planned, by fuel-first propellant depletion. Activation of the fuel low level sensor (LLS) occurred at 108:55:24 GET and the chamber pressure began to decay at 108:55:32.3 GET. Oxidizer LLS activation time was 108:55:37 GET. The engine off command was given at 108:56:14.4 GET. One anomaly was noted during the APS manned burn. The problem concerned the ascent engine quantity caution light which came on approximately one second after APS first burn ignition command; thus triggering a master caution and warning alarm. The engine quantity light is controlled by the oxidizer and/or fuel low level sensors. Since the low level sensors operated nominally during the remainder of the first burn and the entire second burn, it is concluded that the low level sensor did not malfunction but actually went dry. Insufficient propellant settling prior to the burn due to the large ullage volume that existed during the LM-4 flight is considered to be the most probable cause of this difficulty.

Measurement data from the APS BTD indicated two abnormalities. The first problem was observed in the oxidizer interface pressure measurement data which was approximately 11 psia higher than expected. It was concluded, after analysis, that the data was erroneously high. The second involved oscillations of the helium regulator outlet pressure measurements. These oscillations did not propogate to the interface or chamber pressures. Both problems will be more fully discussed in the Pressurization System section of this report. None of the above difficulties materially affected APS performance.

Steady-state engine performance parameters averaged over that portion of the BTD analyzed are as follows:

Thrust - 3432.1 lbf

Isp - 309 49 sec

Mixture Ratio - 1.597

All performance parameters were well within their respective 3-sigma limits. Calculated engine throat erosion for LM-4 APS was approximately one percent higher than the predicted at the end of the BTD.

The following recommendation is made based on the results of this analysis:

Review APS propellant settling requirements for vehicles with large ullage volumes.

3. INTRODUCTION

The Apollo 10 Mission was the tenth in a series of flights utilizing specification Apollo hardware. It was the first flight test of the Lunar Module (LM) in a lunar environment and the second manned LM flight. The mission encompassed the third and fourth manned flights of the Saturn V launch vehicle and Block II Command and Service Modules (CSM), respectively.

The overall mission objective was to duplicate, as closely as feasible, a G type mission with the exception of lunar landing and liftoff. Inherent in this objective was the performance of the Descent Orbit Injection (DOI) maneuver by the Descent Propulsion System (DPS) and a Lunar Orbit Insertion maneuver by the APS. Also included as mission objectives were verifications of both LM operation in a lunar environment and mission support of all spacecraft at lunar distances. In addition, the mission was expected to provide considerable information about the lunar gravitational potential.

Launch from Kennedy Space Center (KSC) occurred at 12:49:00 P.M., Eastern Standard Time, on May 18, 1969. A normal launch phase was followed by insertion of the spacecraft into a parking orbit,of 102.6 by 99.6 nautical miles, by the S-IVB stage of the AS-504 launch vehicle. The S-IVB stage was restarted and performed the Translunar Injection (TLI) maneuver approximately two-and-a-half hours after launch. Docking of the CSM with the LM and separation of the docked vehicles from the S-IVB occurred four hours after launch. Undocking of the LM from the CSM in lunar orbit occurred 98.5 hours after launch. Approximately two hours after completion of the DPS DOI and phasing maneuver burn (about 103 hours GET) the ascent and descent stages were separated and the APS engine was fired for 15.6 seconds (ignition to engine cutoff). Upon completion of this insertion maneuver, the ascent stage docked with the CSM and the crew and equipment transfer was effected. Approximately six hours later, the ascent stage was separated and the engine was ignited for the 248.9 second (ignition to cutoff command) burn to propellant depletion (BTD). Exact data concerning ignition and cutoff times and associated velocity changes are shown in Table 1.

The Apollo 10 LM-4 APS was equipped with Rocketdyne Engine S/N 0002B. APS engine performance characterization equations used in pre-flight prediction and post-flight 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.

Both firings of the APS engine were preceded by Reaction Control System (RCS) maneuvers to settle propellants. The RCS activity prior to the BTD also included a separation maneuver. Total duration of the ullage and separation maneuver prior to BTD was approximately 90 seconds.

Only one Apollo 10 Mission Detailed Test Objective (DTO), S13.13, dealt specifically with APS performance. The functional test objectives of that DTO are as follows:

1) Confirm APS performance characteristics
2) Confirm that APS propellant depletion shutdown in a space environment is not hazardous.

Specific requirements of this objective are described in Reference 4.

4. STEADY-STATE PERFORMANCE ANALYSIS

Analysis Technique

Postflight performance analysis for the LM-4 APS was primarily concerned with determining steady-state performance during the second ascent engine firing, an unmanned burn to fuel-first propellant depletion of 248.9 seconds (engine on to engine off command) duration. The first ascent engine burn was of insufficient length, 15.6 seconds, to properly determine steady-state performance; however, an examination of the measurement data (Appendix) was made and performance was found to be satisfactory.

The APS steady-state performance analysis was conducted using the Apollo Propulsion Analysis Program as the primary tool. This program utilizes a minimum variance technique to establish the best correlation between an engine characterization model derived from ground test data and selected flight test measurements. The minimum variance technique used in the program consists of a series of error models using various ground and flight data as inputs and a non-linear APS simulation model which is based on empriically derived engine characterization equations. Successive iterations through the program result in a "best" estimate, in a minimum variance sense, of system performance history and weights.

Initial vehicle damp and propellant weights were obtained from Reference 5, reaction control system (RCS) propellant usage was obtained from analysis of the bi-level measurements and APS propellant usage for the first burn was based on estimated steady-state usage. Vehicle damp weight is considered to be constant, with the exception of the RCS propellants consumed, throughout the run. Table 3 presents RCS consumption for the propellant settling and separation maneuver just prior to the BTD and during the BTD. All RCS consumption during the BTD was from RCS tanks. Propellant densities used in the program were based on equations found in Reference 6 adjusted by measured density data for the LM-4 flight given in the Spacecraft Operational Data Book, Reference 7. Oxidizer and fuel temperatures were taken from measurement data and were 69.8°F and 70.9°F, respectively. These temperatures were found to be constant throughout the segment of burn analyzed as steady-state.

The following flight measurement data were used in the analysis of the LM-4 APS BTD: engine chamber pressure, engine interface pressures, vehicle thrust acceleration, propellant tank bulk temperatures, helium regulator outlet pressures, propellant LLS activation times, engine on-off commands and RCS thruster solenoid bilevel 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

A 174 second segment of the APS BTD was selected to be analyzed for the purpose of determining steady-state engine performance. APS BTD ignition occurred at a GET of 108:52:05.5 and engine cutoff was commanded at 108:56:14.4 GET. The segment of the burn analyzed begins at 108:52:25.0 GET, 19.5 seconds after ignition, and ends at 108:55:19.0 GET. A longer analysis segment was not feasible since filtered acceleration data was distorted by the smoothing technique just prior to the start of chamber pressure decay. Steady-state analysis of the APS BTD revealed no significant anomalies. APS engine propellant consumption during the BTD is presented in Table 5. Consumption from the end of the steady-state analysis segment to the beginning of chamber pressure decay was extrapolated from steady-state analysis results.

The principal results determined by the APS steady-state analysis are as follows:

1) Engine throat erosion was slightly higher than predicted, approximately one per cent at the end of the BTD.

2) Average specific impulse over the 174 second period analyzed was 309.5 seconds.

3) The average mixture ratio based on LLS actuation time was determined to be 1.597.

4) Oscillations in helium regulator outlet pressure were found to have no adverse effects on engine performance.

LM-4 APS performance, was determined to be very close to predicted with actual engine average specific impulse being approximately 0.8 second higher than the predicted value. A portion of this increased performance is attributable to a slightly increased throat erosion rate, however, an increase in engine efficiency was also evidenced.

It should be noted that the number of APS flight measurements was substantially reduced for the LM-4 flight; thus automatically eliminating some of the checks and balances that had been available on previous flights. Of particular significance was the deletion of the tank bottom to engine interface differential pressure (ΔP) measurements, since they provided an excellent means of verifying throat erosion characteristics.

The general solution approach used in the LM-4 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. This technique led to excessive variations in both propellant loads and vehicle inert weight. In order to resolve these variations, the vehicle thrust and specific impulse were increased slightly from predicted values until weight differences become more realistic. A complicating factor in the analysis was the fact that the oxidizer interface pressure measurement, GP1503P, was erroneously high by approximately 11 pounds per square inch (psi). This bias and an approximately 2 psi positive bias on fuel interface pressure, determined from ground support equipment data, were applied to the flight interface pressure measurements input to the program. Program results indicated that the above biases were essentially correct. A further discussion of the oxidizer interface pressure bias may be found in the Pressurization System section of this report. It was also necessary to curve fit the interface pressure data since the smoothing technique applied to the raw data resulted in extensive distortion. The acceleration residual (measured data minus calculated) resulting from the program input data outlined above had a definite negative slope indicating that a slight increase in calculated acceleration as flight time increased was required to minimize the residual error. This affect is gained by increasing engine flowrates and/or increasing engine thrust on a time basis. Since experience on the LM-3 flight indicated that actual throat erosion rates could be considerably higher than predicted and since a slight increase in throat erosion would give the desired result, 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 inclusion of this calculated throat area curve in the analysis program resulted in an excellent acceleration match with essentially a zero mean and no significant slope. The derived throat erosion curve was approximately one per cent higher than predicted at the end of the BTD. This is considerably lower than the value of throat erosion determined during the LM-3 postflight evaluation. It should be noted that the segment of the burn analyzed as steady-state is somewhat shorter for LM-4 than it was for LM-3, however, the LM-4 throat erosion would not have been as large as that of LM-3 even if the burn times had been equivalent. Figure 1 shows the calculated throat area curve in comparison with the predicted curve for LM-4 and the maximum and minimum curves for which the throat erosion characterization is valid.

Simulation of RCS activity was not as crucial to an accurate postflight reconstruction of APS performance as it might have been since the thrusters were theoretically close coupled, i.e., the net translational thrust for the system is zero. In actuality, a small impulse, approximately equivalent to 5 lbf thrust in the negative X direction, existed throughout the BTD. The magnitude of this impulse was determined by analyzing accumulated "on" times for all up and down engines and multiplying by a nominal 100 lbf thrust. The RCS interconnect valves were closed during the BTD, therefore, RCS propellants came only from the RCS tanks. Weight overboard through the RCS engines was accounted for by multiplying the nominal one engine flowrate by total accumulated system "on" time and averaging this figure over the period of the BTD. Usage during a 90 second ullaging and separation maneuver prior to BTD ignition was accounted for separately but in a similar manner.

Figures 2 through 11 depict the principal performance parameters associated with the LM-4 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 (measured minus calculated values) are presented in Figures 2 and 3, respectively. Figures 4 and 5 contain oxidizer and fuel interface pressure measurement data as it appeared after smoothing of the raw data, the curve fit of this data that was ultimately input to the Apollo Propulsion Analysis Program, and the residual between the two. Data shown in Figures 4 and 5 were adjusted by the fixed biases previously discussed prior to being input to the program. Calculated steady-state values for the following parameters are shown in Figures 6-11; thrust, specific impulse, oxidizer flow rate, fuel rate, and oxidizer and fuel tank bottom to engine interface differential pressures.

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. This data represents the intercept, on the ordinate, and slope of a linear fit to the residual data. It is readily seen that the closer both these numbers are to zero, the more accurate is the match. The acceleration match achieved with the LM-4 postflight reconstruction is excellent. A match of measured and calculated engine chamber pressure is given in Figure 3 and is also considered to be quite good. An additional indicator of the validity of the reconstruction is the matching of the actual amount of propellant in the tanks at LLS probe actuation times with the program calculated amount of propellant in the tanks at corresponding times. Based on the densities of the propellants, as determined from flight temperatures, and LLS probe heights, the weights of oxidizer and fuel in the tanks at their respective LLS actuation times were 48.2 lbm, oxidizer, and 38.4 lbm, fuel.

The weights of propellant in the oxidizer and fuel tanks at corresponding times, as calculated by the propulsion analysis program, were 58.9 lbm, oxidizer, and 45.0 lbm, fuel. The difference between actual and calculated in both cases is well within the acceptable variance due to propellant sloshing.

The LM-4 flight reconstruction is by all indications an accurate simulation of actual flight performance. Residuals between calculated and measured parameters are all within measurement accuracies.

Comparison with Preflight Performance Prediction

Predicted performance of the LM-4 APS is presented in Reference 8. The intention of the preflight performance prediction was to simulate APS performance under flight environmental conditions for the projected mission duty cycle. No attempt was made in the preflight prediction to simulate RCS operation.

Table 6 presents a summary of actual and predicted APS performance during the BTD. Measurement data in Table 6 have been adjusted by known biases as necessary and compare quite closely with the reconstructed parameters. The most significant difference between the actual and predicted APS performance is that predicted regulator outlet and interface pressures were higher than actual by about 3-4 psi throughout the BTD. This difference is within the Class I primary regulator operating band. The pre-flight prediction used a helium regulator outlet pressure based on the Class I primary regulator operating band mid-point value of 184 psia. Engine specific impulse determined by the postflight reconstruction is somewhat higher than had been predicted but is still well within specified 3 sigma limits. A comparison of predicted and reconstructed values for specific impulse, thrust, and mixture ratio is presented in Figure 12 along with related three sigma dispersions. The variation of flight specific impulse, thrust and mixture ratio were within their respective three sigma dispersions.

Engine Performance at Standard Interface Conditions

Expected flight performance of the APS engine was based on a model characterized with 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, °F	70.
Fuel interface temperature, °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	16.49

Analysis results (at 20 seconds from ignition) for the BTD corrected to standard inlet conditions and compared to acceptance test values are shown below:

	Acceptance Test Data	Flight Analysis Results	% Difference
Thrust, lbf	3508.	3513.	.1
Specific Impulse, $\frac{lbf\text{-}sec}{lbm}$	308.7	309.3	.2
Propellant Mixture Ratio	1.594	1.594	0.

These differences are well within the engine combined repeatability and acceptance test instrumentation uncertainties. Flight interface pressures for that part of the BTD analyzed as steady-state were approximately 3 psia below predicted levels. The engine thrust level for flight shows the effect of the reduced pressure but is not as low as it might have been due to a compensating increase in engine efficiency. Adjusting engine performance to standard interface conditions and comparing with acceptance test values shows good agreement with the small differences being attributable to the previously mentioned engine efficiency increase. All differences are within one standard deviation of acceptance test values. This indicates that basic preflight prediction techniques are adequate, with the possible exception of the throat erosion characterization. Based on the results of the LM-3 flight analysis, a question as to the adequacy of the APS throat erosion was raised. However, as previously discussed, throat erosion for the LM-4 flight was only slightly higher than predicted. The question could become somewhat academic in light of the fact that LM-5 and subsequent vehicles will utilize the SWIP chamber which may exhibit different throat erosion characteristics. It is noted that variations in throat area do not enter into the determination of standard interface condition performance since the acceptance test throat area becomes part of the standard interface conditions.

5. PRESSURIZATION SYSTEM

Helium Utilization

The helium load for the LM-4 flight was a nominal 13.2 lbm. Helium tank temperatures and pressures prior to launch were 3067 psia, 71.8°F and 3083 psia, 71.1°F respectively, for tanks 1 and 2. There was no indication of helium leakage during the mission. The calculated helium usage during the burn agrees with analytical predictions.

Ullage Pressure Decay During Coast

The fuel and oxidizer interface pre-launch pressures were 167 and 178 psia, respectively. Telemetry data received during the initial lunar module activation period indicated that the oxidizer interface pressure had increased to 187 psia. The oxidizer temperature just prior to launch was 74°F and was 71°F at initial lunar module activation. The predicted oxidizer interface pressure, based on temperature drop and helium solubility in propellant, corresponding to the 71°F temperature was 172.5 psia. Fuel interface pressure at initial lunar module activation was, as expected, 166 psia.

With temperatures holding near constant, pressure levels should either remain constant or decrease due to helium solubility in the propellants. In order for tank (or interface) pressures to rise, an increase in helium mass in the ullage would be required. The data available, plus the system configuration, indicated no mass could have been added to the ullage from the only possible source, the helium supply tanks. The conclusion reached from the above considerations is that the GP1503 measurement was erroneously high. This conclusion is further substantiated by the agreement of observed data with predictions on the LM-4 fuel side and both fuel and oxidizer measurements on LM-3.

Determination of the actual oxidizer interface pressure measurement bias to be used during the BTD steady-state analysis was made by comparing the level of the oxidizer interface pressure to that of the fuel interface pressure after biasing the fuel interface pressure on the basis of ground test data. It was assumed that the fuel and oxidizer interface pressures would be within one psia of each other and that the level of the fuel interface pressure, after biasing, was correct. This assumption was based primarily on analysis of the helium regulator outlet pressure data which gave no indication that the higher level of oxidizer interface pressure was valid. This technique indicated an oxidizer interface pressure bias of 10-11 psia, with the higher value being chosen. As previously mentioned, results from the propulsion analysis program substantiated this bias.

Regulator Performance

The regulator lockup pressure at initial APS pressurization was 184 psia (compared to 186 expected based on tests during checkout at KSC). Regulation during the insertion burn and lockup after the burn were nominal. At the start of the burn to depletion, the regulator outlet pressure dropped to the expected (from KSC checkout) value of 181 psia. At 118 seconds into the burn, pressure oscillations were observed in both helium regulator outlet pressure transducers (GP0025P and GP0018P). These oscillations were present for the remainder of the burn to depletion. GP0025P, located 4 inches downstream of the Class I primary regulator, indicated an amplitude of 5 psi (peak to peak). GP0018P, located upstream of the oxidizer check valves (3 feet downstream of GP0025P) indicated a maximum amplitude of 19 psi. Since both measurements are located in the helium manifold, the differences in the magnitude of the oscillations may be explained by (1) amplification of the pressure oscillations in the helium line or (2) varying sensitivity to oscillations of the two transducers. Pre-installation tests of the regulator module at Bethpage showed pressure oscillations of 3 psi peak-to-peak at the regulator outlet (GP0025P) and 6 psi peak-to-peak at the tank relief valve (downstream of the GP0018P location). The regulators have a specification limit of 15 psi peak-to-peak.

Pressure oscillations of amplitudes up to 10 psi have been observed at WSTF during PA-1 testing. The oscillations did not propogate to the interface or chamber pressures, due, probably, to the large propellant tank ullage volumes present during the LM-4 BTD. It is also considered likely that the pressure transducers amplified the oscillations to a certain extent. No harmful effects to the system due to the pressure oscillations were noted.

6. APS PROPELLANT LOADING AND USAGE PRIOR TO BURN TO DEPLETION

The oxidizer tank was fully loaded at a pressure of 67 psia and an oxidizer temperature of 72°F. The fuel tank was loaded at a pressure of 64 psia and a fuel temperature of 70°F. A density determination was made for both oxidizer (1.4818 gm/cc at 4°C and 14.7 psia) and fuel (.8992 gm/cc at 25°C and 14.7 psia) samples. Based on these density values, propellant tank pressures, and propellant temperatures; a determination was made of the quantity of propellant to off load. This off load (1072.0 lbm fuel and 1623.0 lbm oxidizer) was accomplished using the weigh tank five times. The actual propellant load was determined to be 981.4 lbm fuel and 1650.1 lbm oxidizer.

RCS consumption from the APS tanks during the Coelliptic Sequence Initiation (CSI) and Terminal Phase Initiation (TPI) RCS maneuvers was 28 lbm and 14 lbm, respectively. Therefore, the propellant available to the APS was 1622 lbm oxidizer and 967 lbm fuel.

APS consumption during the manned burn was approximately 106 lbm of oxidizer and 67 lbm of fuel. The RCS system interconnect valve was closed during APS activity so that all RCS consumption was from the RCS tanks.

7. PROPELLANT SETTLING ANOMALY

Shortly after the ignition signal for the first APS burn, the ascent engine quantity caution light came on triggering a master caution and warning alarm. The quantity caution light is controlled by the oxidizer and fuel LLS in each propellant tank. Approximately one second after the ascent engine "on" signal for the LM-4 APS manned burn, the oxidizer LLS was activated for about one second, while the master alarm was on for two seconds before being manually reset. Low level sensors in both oxidizer and fuel tanks operated as expected during the remainder of the first burn and the entire second burn. Based on this performance, it was concluded that the signal received during the first burn was valid.

The cause of this anomaly is believed to be insufficient settling of APS propellants prior to the burn. Based on Figure 4.8-10 in the Spacecraft Operational Data Book (SODB), Reference 9, an RCS propellant settling maneuver of 3-4 seconds duration would be adequate for a vehicle with the propellant load of LM-4. The actual RCS ullage propellant settling maneuver prior to APS first burn ignition lasted for a period of 4.1 seconds. This would seem to indicate that an adjustment to the SODB propellant settling figures is required for missions utilizing vehicles with large ullage volumes. Lunar landing missions do not fit into this category for two reasons; first the positive gravitational field on the lunar surface is sufficient to settle propellants and secondly the normal APS loading for lunar landing missions is very near tank capacity.

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5. TRW IOC 69.6522.2.1-30. "Apollo 10 Post Flight Mass Properties," 8 July 1969.

6. NASA Memorandum EP23-10-69, from EP/2 Head, Development Section to EP/2 Chief, Primary Propulsion Branch, "Propellant Densities (N_2O_4 and A-50)," 18 February 1969.

7. Spacecraft Operational Data Book, SNA-8-D-027 (III) Rev. 1, Amendment 58 Vol. III, 2 May 1969

8. TRW IOC 69.4354.2-47, "Mission F/LM-4/APS Propulsion Performance Prediction," from C. G. Johnston to D. W. Vernon, 3 June 1969.

9. Spacecraft Operational Data Book, SNA-8-D-027 (II) Rev 1, Amendment 2-180, 11 July 1969.

10. NASA Report MSC-00126, "Apollo 10 Mission Report," August 1969.

TABLE 1 - LM-4 APS DUTY CYCLES

BURN	Ignition FS-1 Hr:min:sec GET	Engine Cutoff FS-2 Hr:min:sec GET	Burn Duration Secs.	Velocity[1] Change Ft/sec
APS 1st Burn				
Lunar Orbit Insertion Maneuver	102:55:02.1	102:55:17.7	15.6	221.
APS 2nd Burn				
Burn to Depletion	108:52:05.5	108:56:14.4	248.9	3837.

(1) Reference 10

TABLE 3

RCS PROPELLANT USAGE DURING APS BTD

	RCS Storage Tanks (lbm)			APS Storage Tanks (lbm)		
	Oxidizer	Fuel	Total	Oxidizer	Fuel	Total
Propellant Settling and Separation Maneuver	46.7	23.3	70.	0	0	0
APS (BTD) Ignition* to Depletion	61.5	30.7	92.2	0	0	0
TOTALS	108.2	54.0	162.2	0	0	0

* Data presented in this table was derived from RCS thruster accumulated "on" time.

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
GP1408X	Oxidizer Tank Low Level Sensor	Off-On	1
GP0025P	Pressure, Regulator Outlet Manifold	0-300 psia	1
GP1218T	Temperature, Oxidizer Tank Bulk	20-120°F	1
GP0718T	Temperature, Fuel Tank Bulk	20-120°F	1
GH1260X	Ascent, Engine On/Off	Off-On	50
GG0001X*	PGNS, Down Link Data	Digital Code	50

* Acceleration determined from PGNS, Down Link Data

TABLE 5 - PROPELLANT CONSUMPTION FROM ASCENT PROPULSION SYSTEM TANKS

Event	GET Time, hr:min:sec	APS Propellant on Board	
		Oxidizer	Fuel
Launch	0:00:00	1650.1	981.4
Ignition for Lunar Orbit Insertion Maneuver	102:55:02.1	1650.1	981.4
Ignition APS Burn to Depletion	108:52:05.0	1516.3	900.3
Start of Chamber Pressure Decay for BTD	108:55:32.3	107.0	17.7

TABLE 6 - STEADY-STATE PERFORMANCE DURING BURN TO DEPLETION

PARAMETER	15 sec after Ignition			100 Sec after Ignition			200 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)	185		181.	185		181	185		181
Oxidizer Bulk Temperature °F	70	70	70	70	70	70	69	70	70
Fuel Bulk Temperature °F	70	71	71	71	71	71	70	71	71
Oxidizer Interface Pressure, psia	171	168	167	171	167	167	168	166	166
Fuel Inter-face Pressure psia	171	167	167	171	166	166	168	166	166
Engine Chamber Pressure, psia	123	121	121	123	121	121	123	120	120
Mixture Ratio	1.592	1.600	----	1.589	1.597	----	1.585	1.592	----
Thrust, lb	3492	3449	----	3482	3430	----	3459	3427	------
Specific Impulse, sec	308.3	309.3	----	308.6	309.6	-----	308.4	309.3	----

(a) Preflight Prediction based on acceptance test data and assuming nominal system performance.
(b) Reconstruction, minimum variance technique.
(c) Actual flight data with known biases removed.

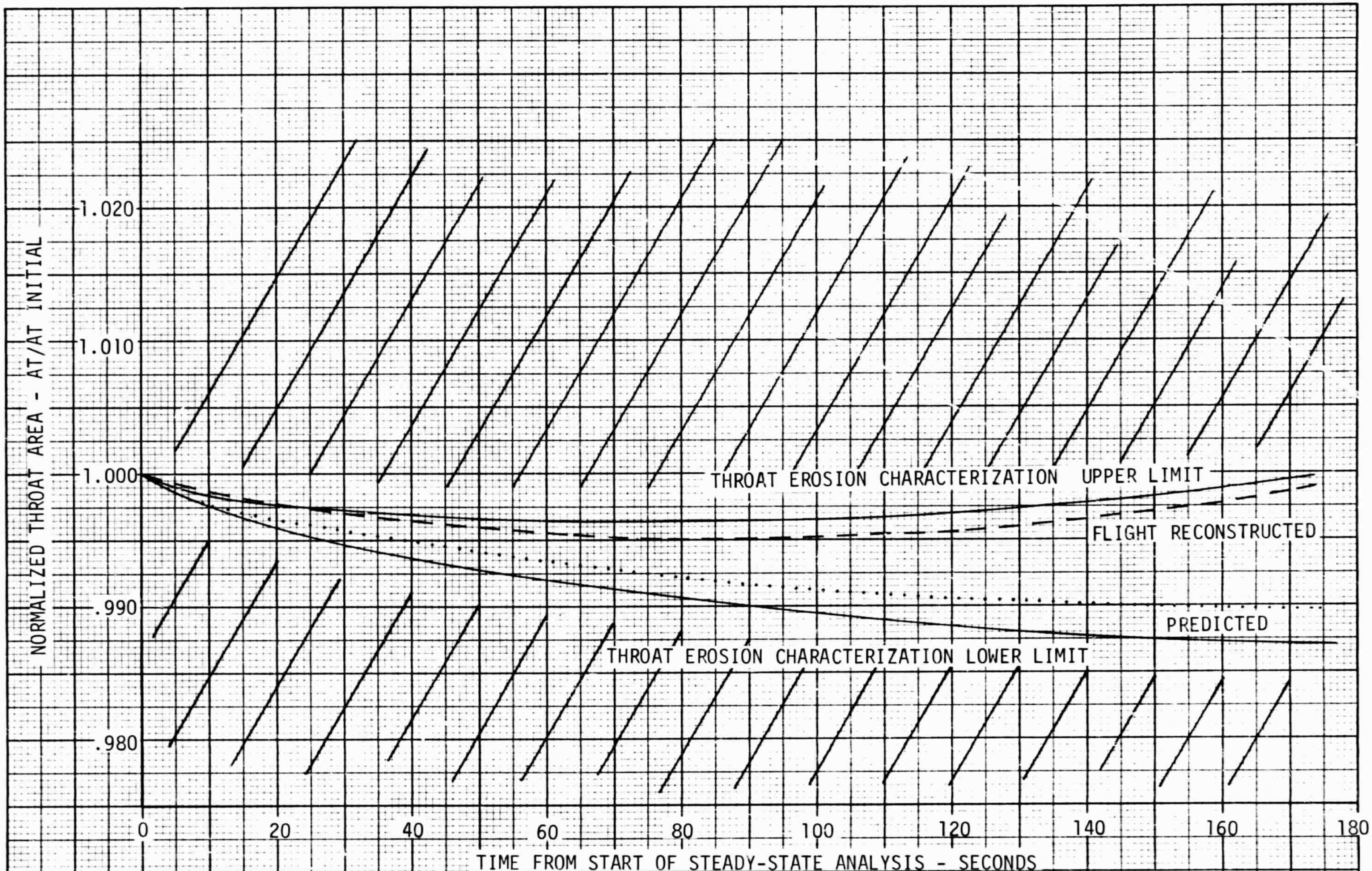


FIGURE 1 COMPARISON OF THROAT EROSION, PREDICTED AND FLIGHT

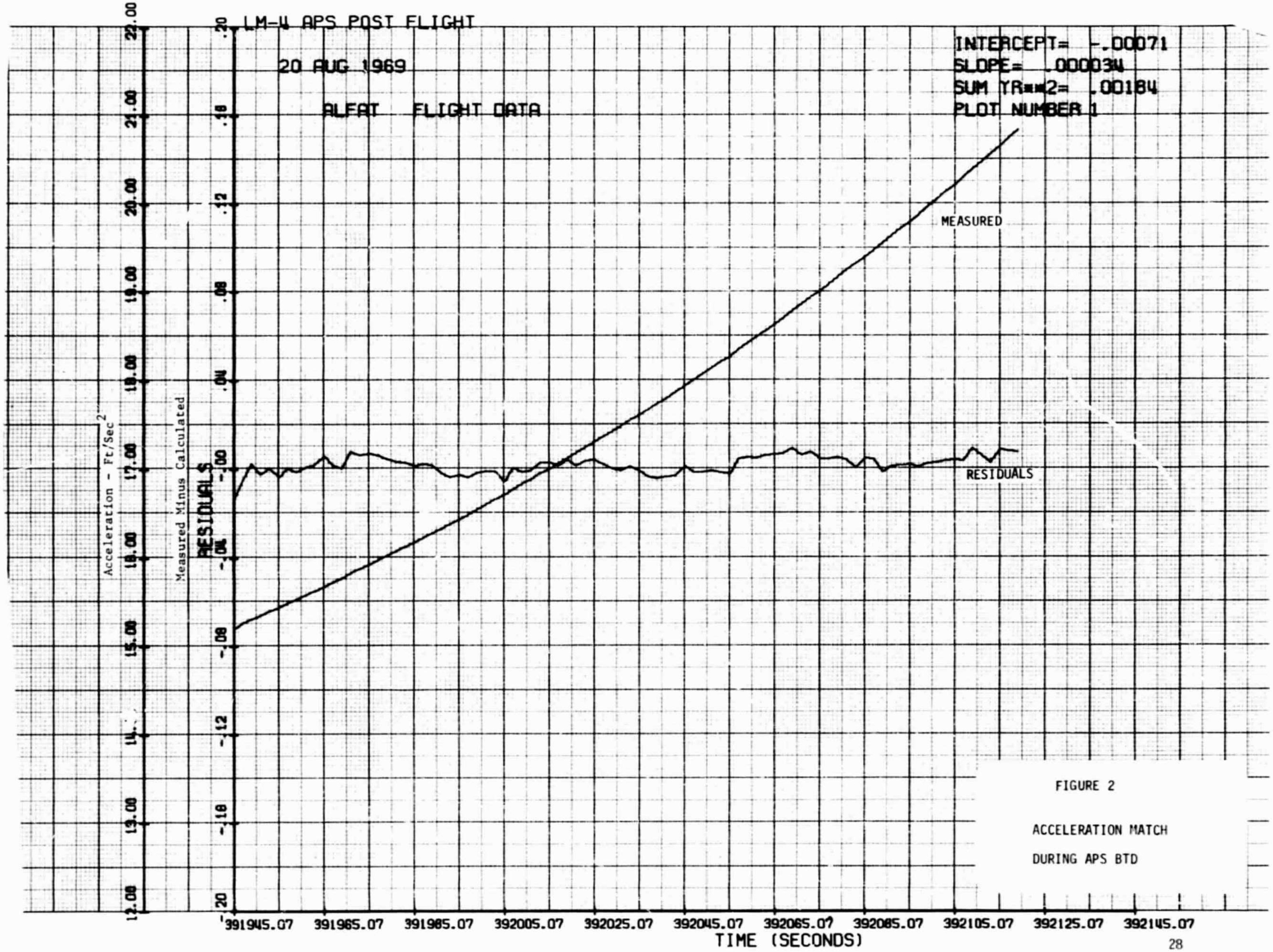


FIGURE 2

ACCELERATION MATCH

DURING APS BTD

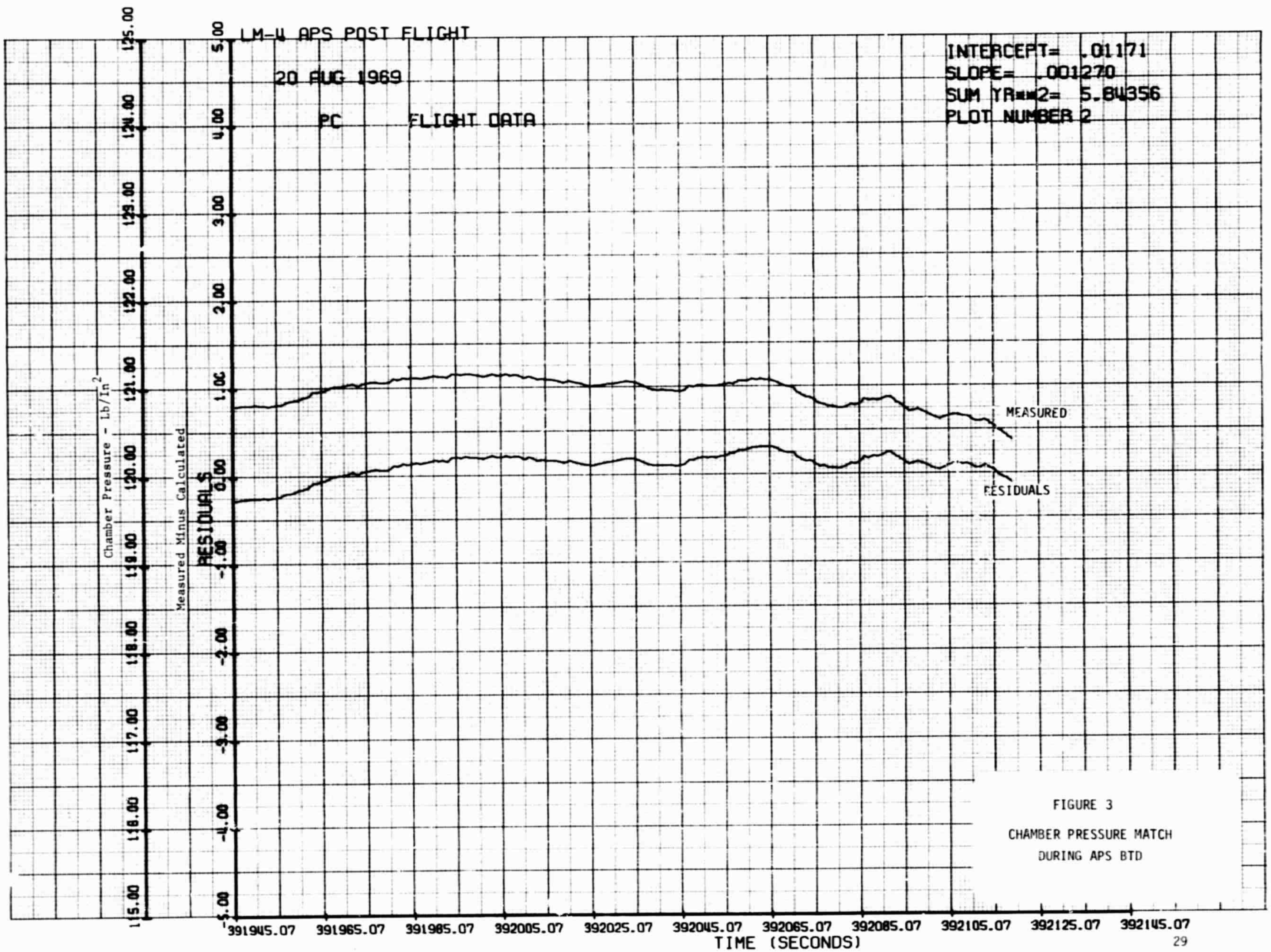


FIGURE 3

CHAMBER PRESSURE MATCH

DURING APS BTD

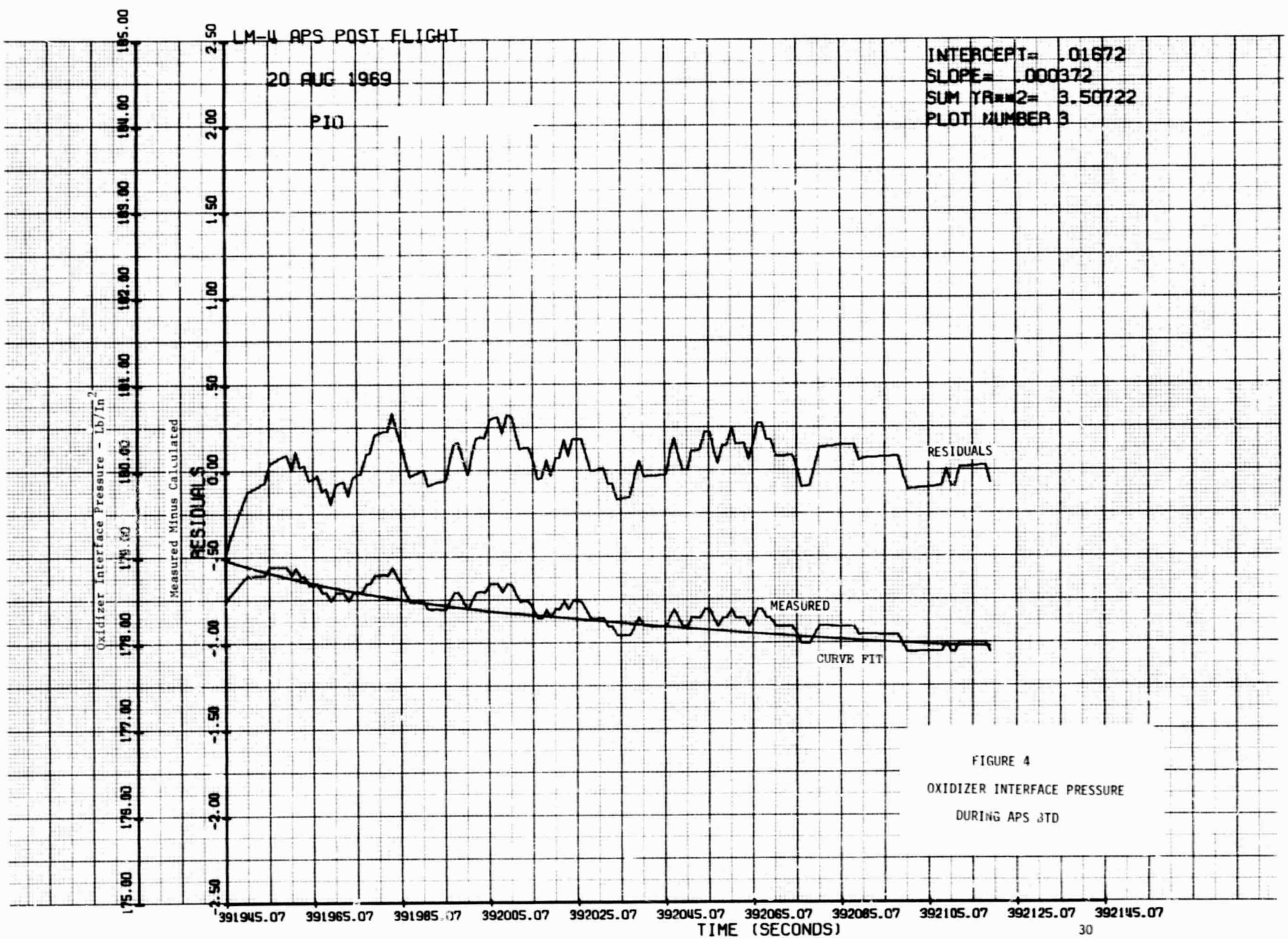


FIGURE 4

OXIDIZER INTERFACE PRESSURE

DURING APS 3TD

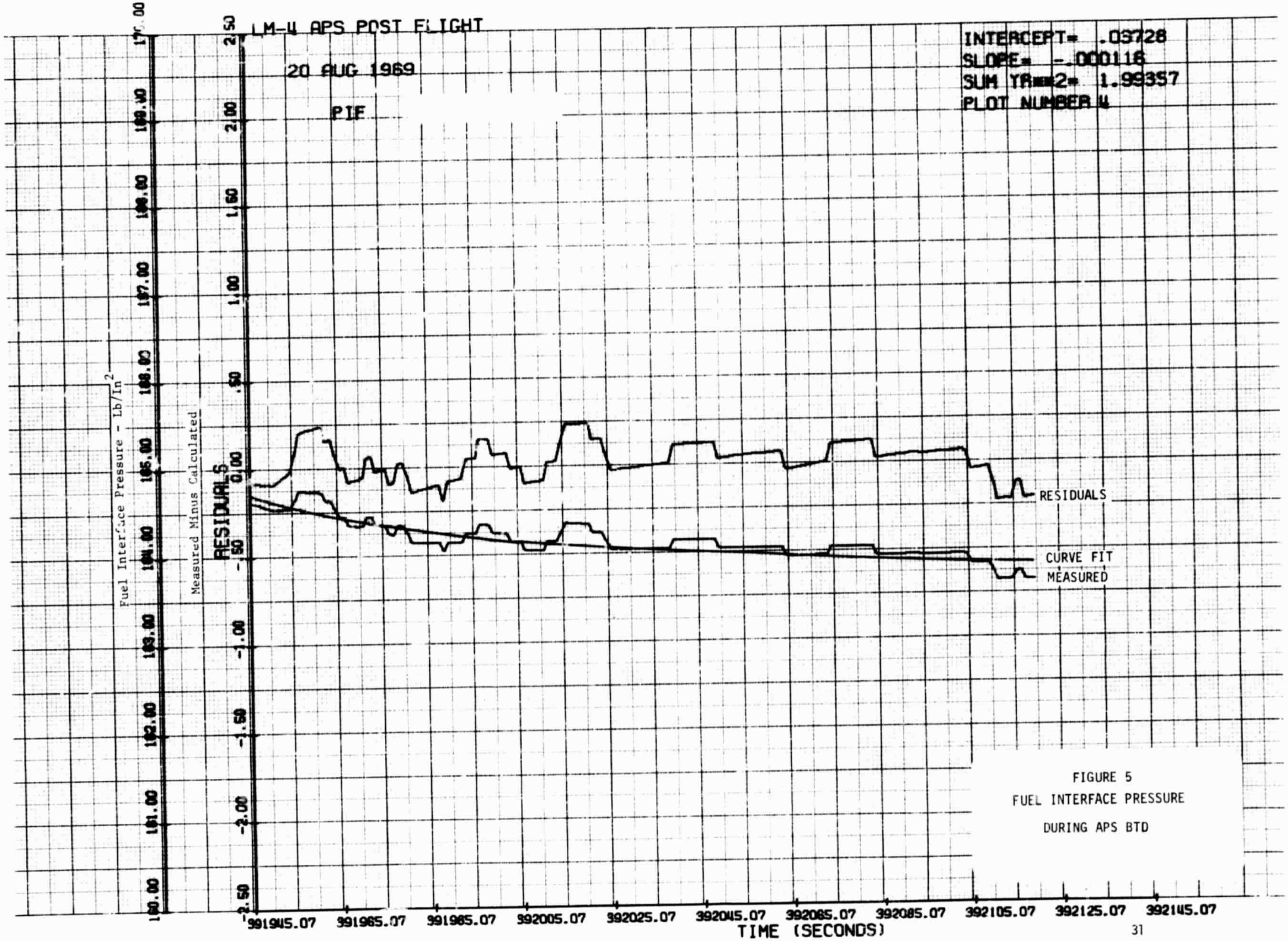


FIGURE 5
FUEL INTERFACE PRESSURE
DURING APS BTD

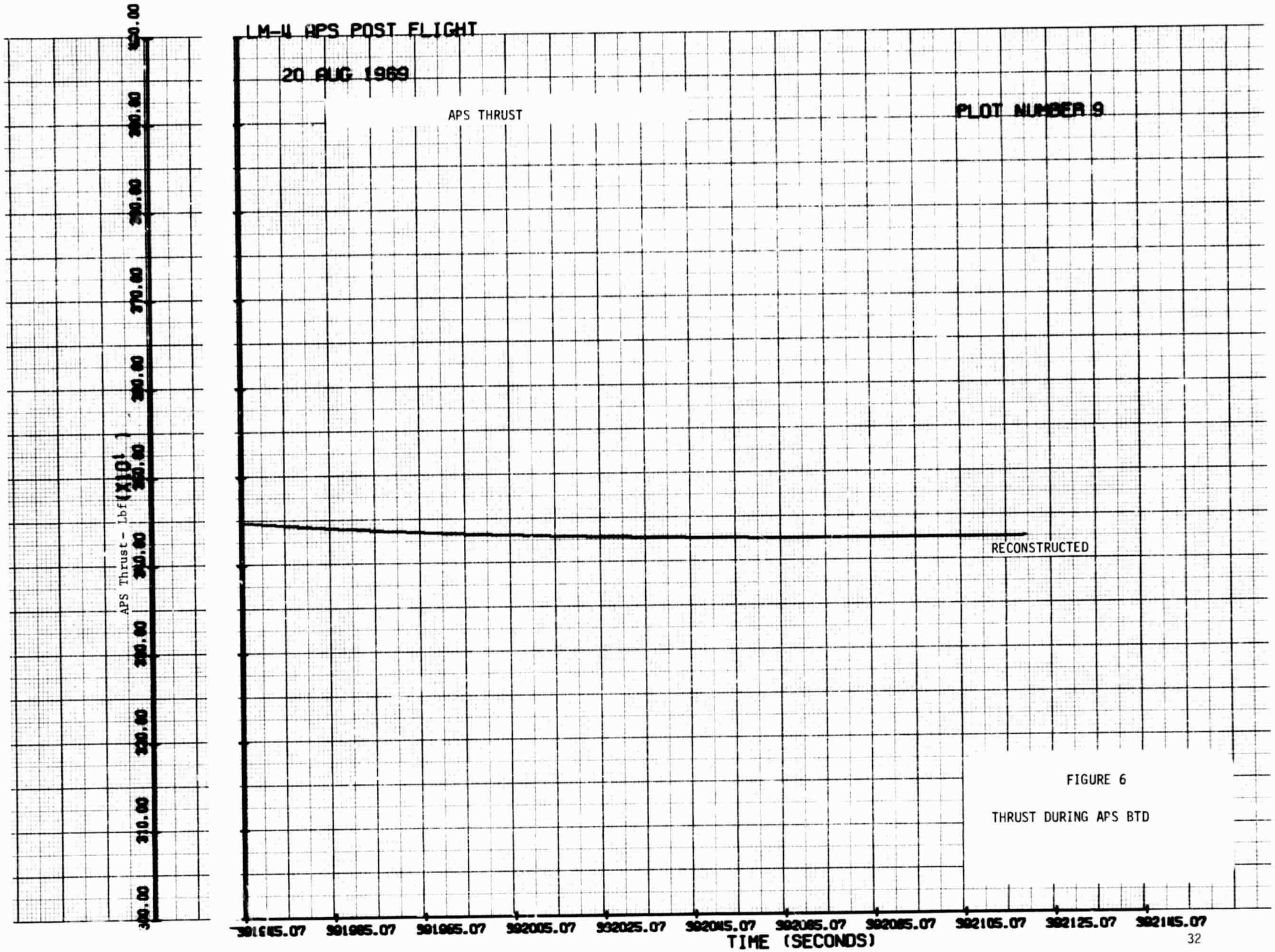


FIGURE 6

THRUST DURING APS BTD

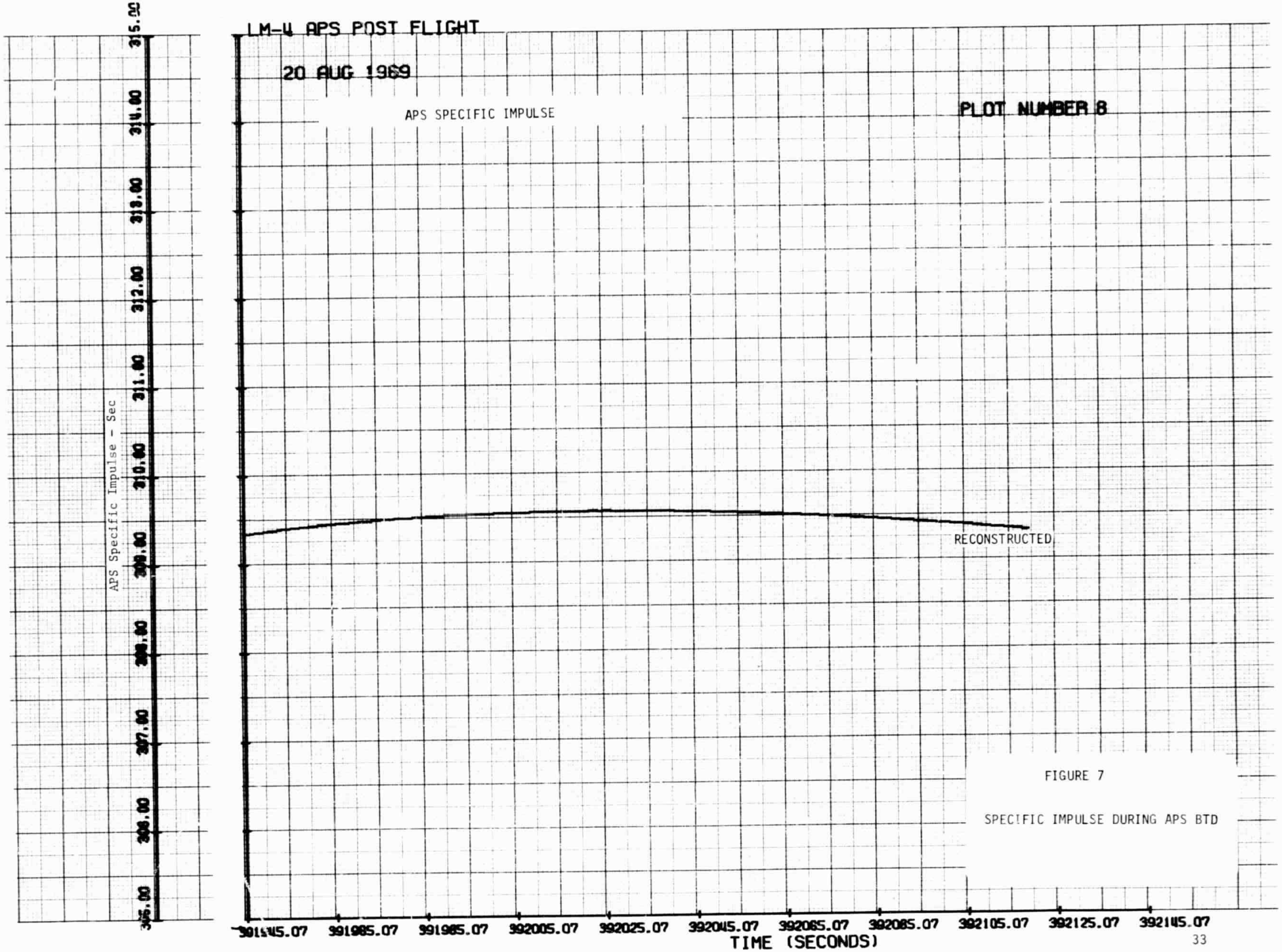


FIGURE 7

SPECIFIC IMPULSE DURING APS BTD

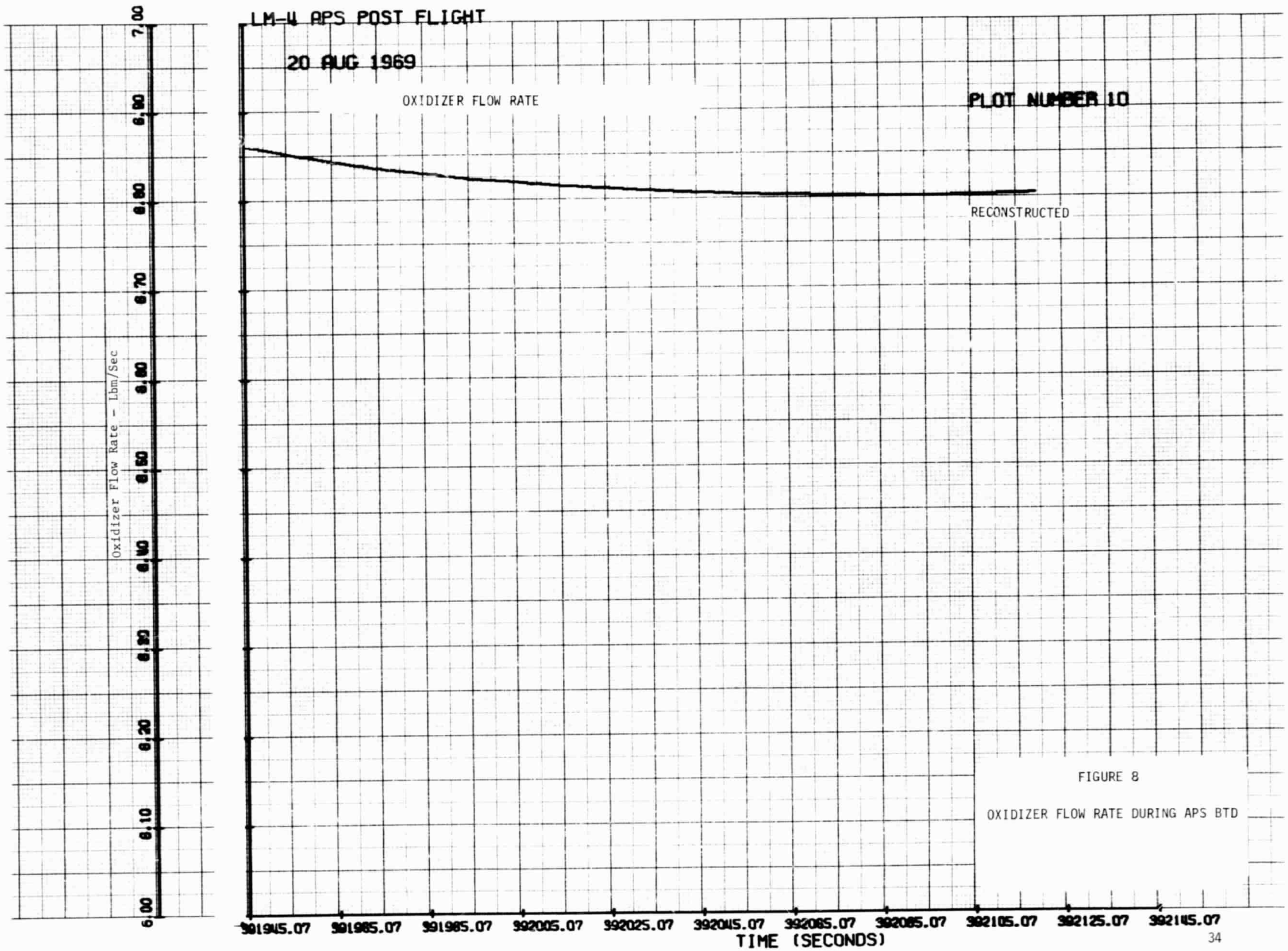


FIGURE 8

OXIDIZER FLOW RATE DURING APS BTD

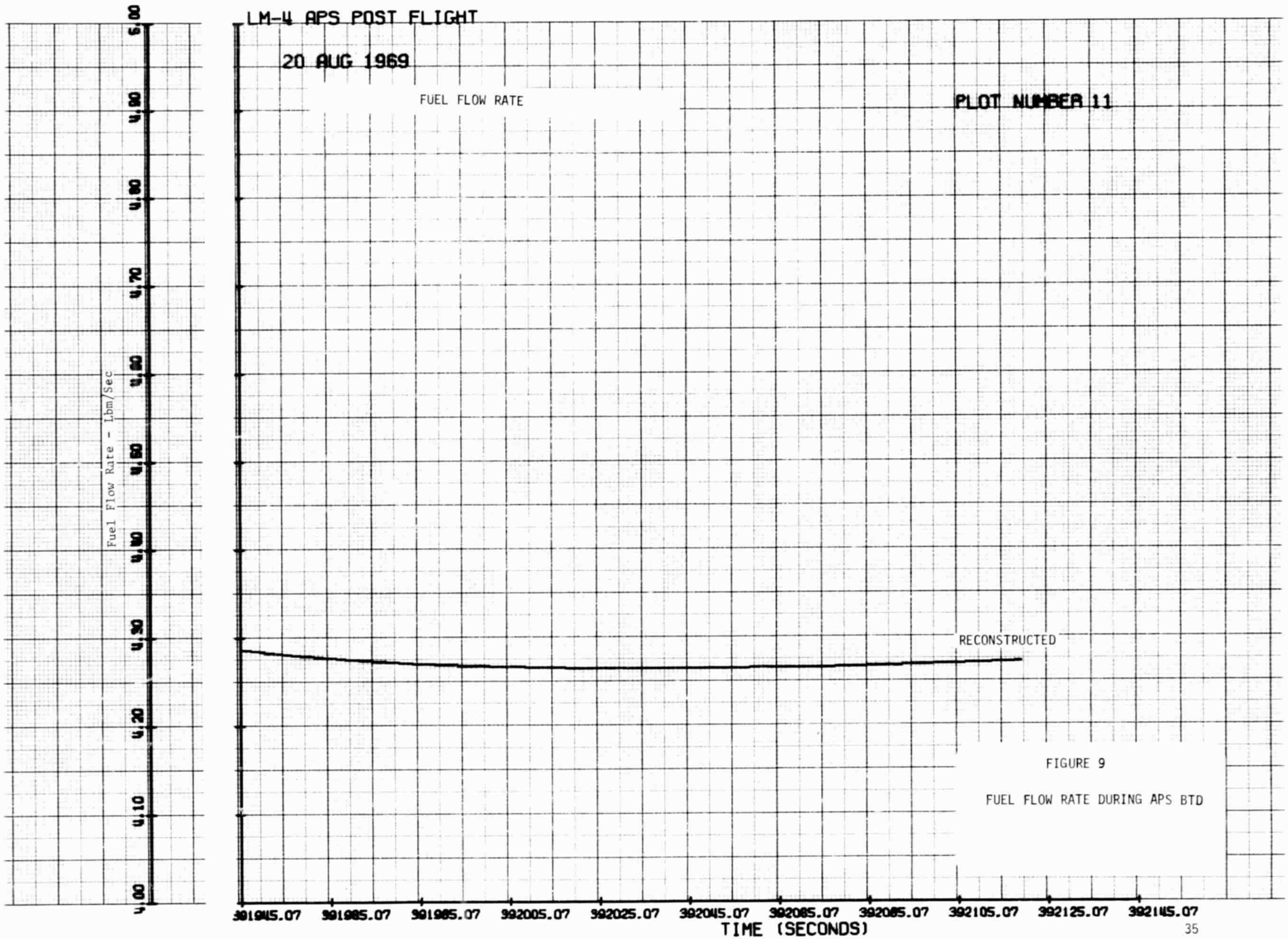


FIGURE 9

FUEL FLOW RATE DURING APS BTD

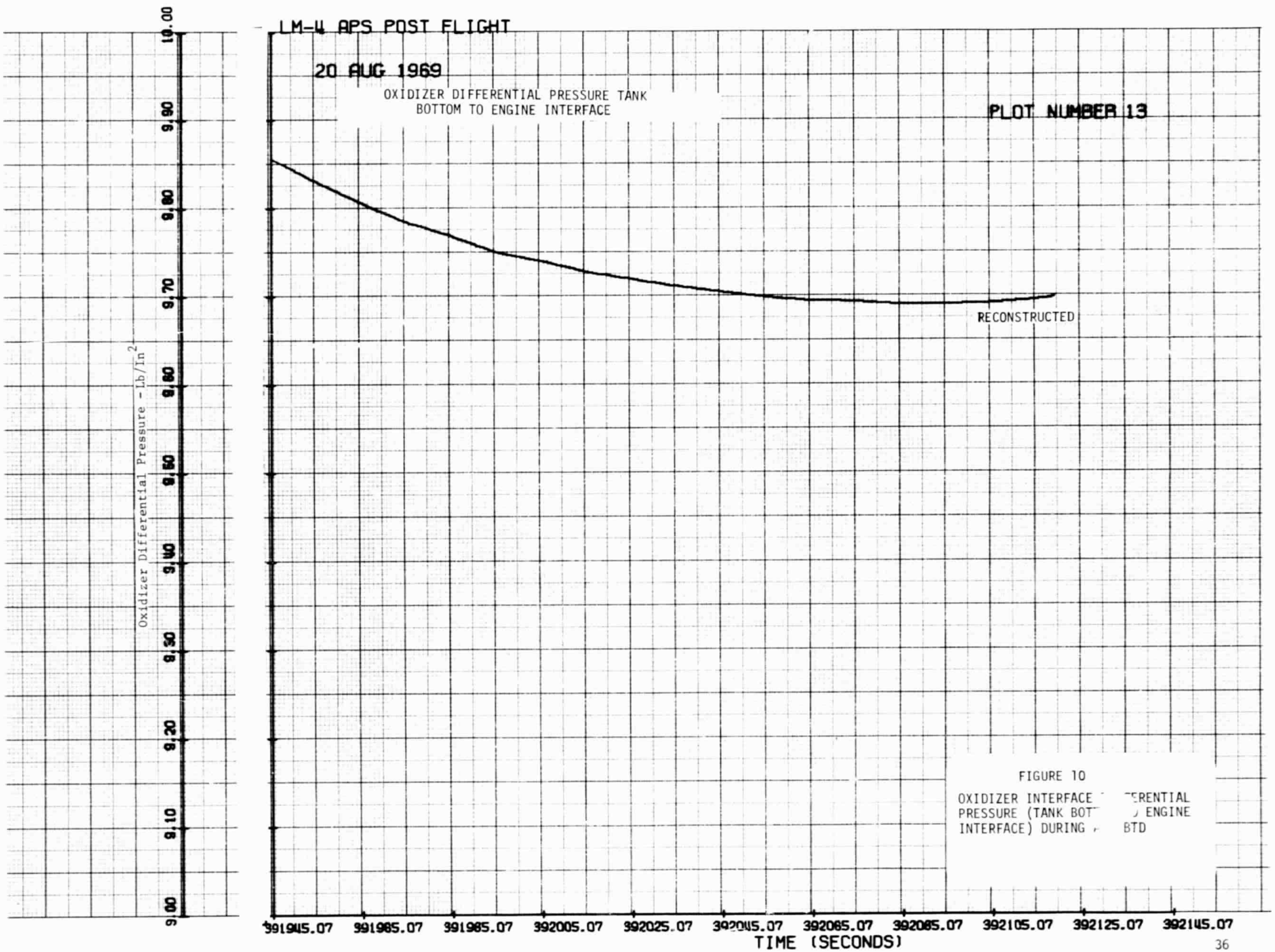


FIGURE 10

OXIDIZER INTERFACE ERENTIAL PRESSURE (TANK BOT ENGINE INTERFACE) DURING BTD

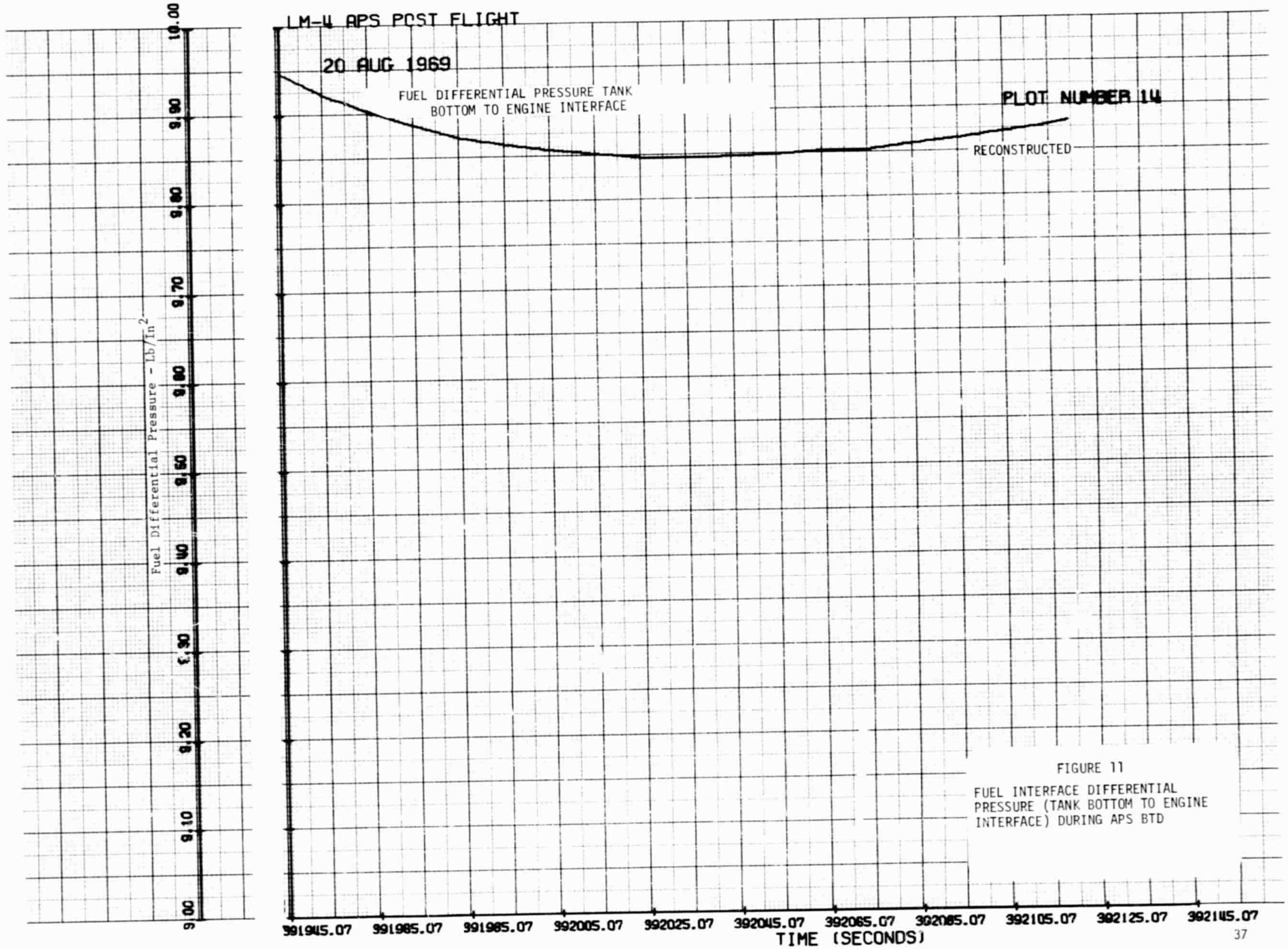


FIGURE 11

FUEL INTERFACE DIFFERENTIAL PRESSURE (TANK BOTTOM TO ENGINE INTERFACE) DURING APS BTD

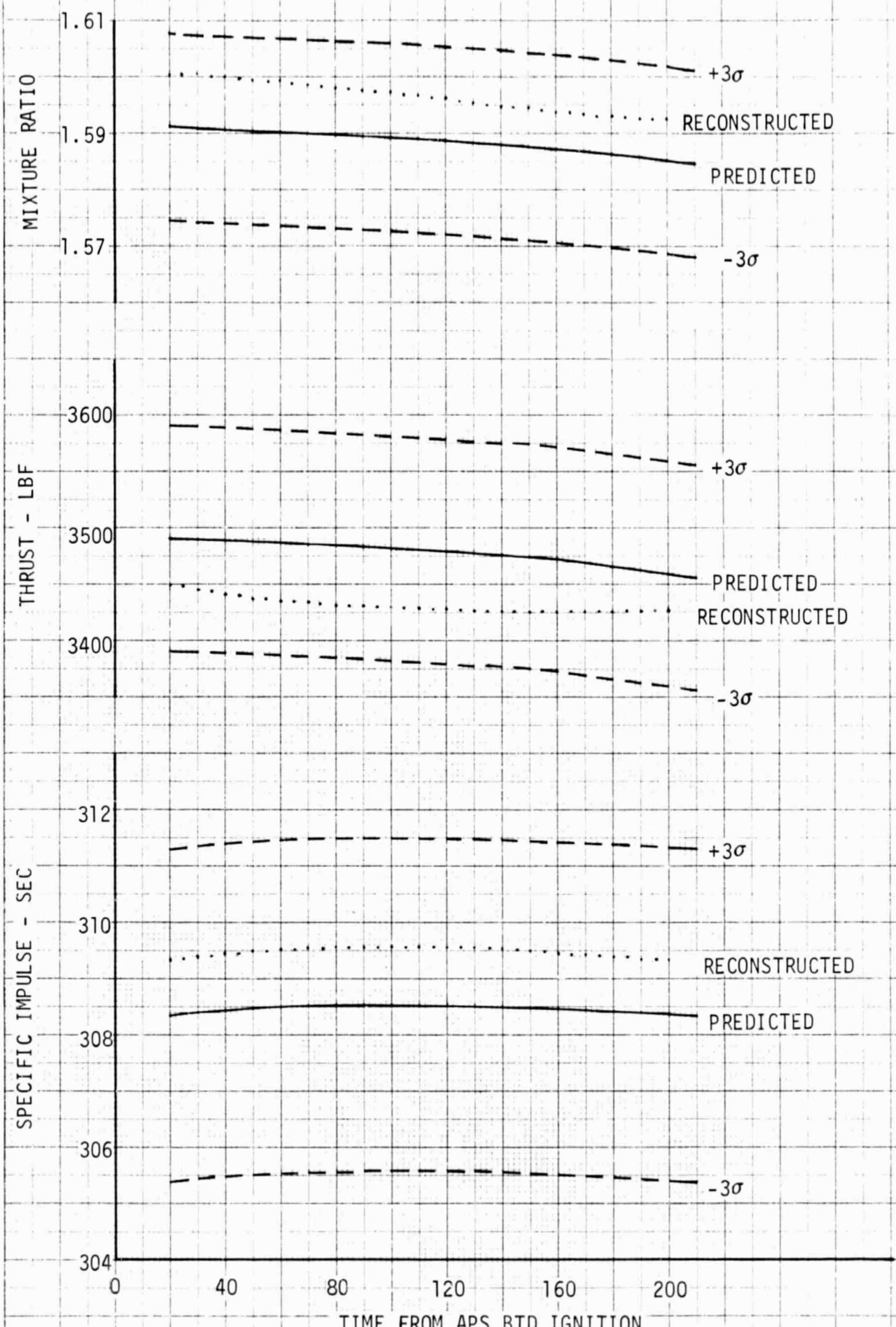


FIGURE 12 COMPARISON OF PREDICTED AND FLIGHT RECONSTRUCTED PERFORMANCE

Appendix

Flight Data

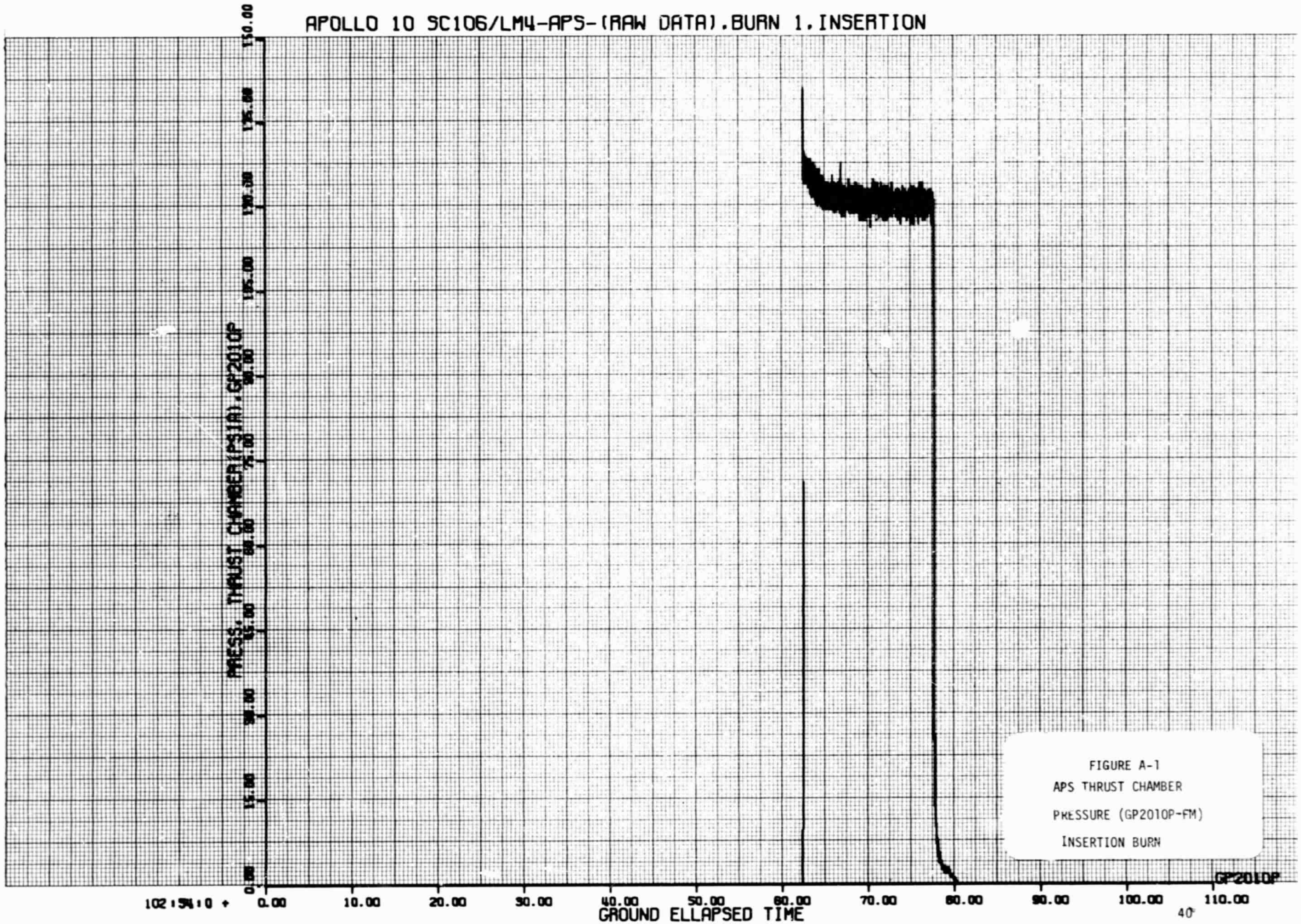


FIGURE A-1
APS THRUST CHAMBER
PRESSURE (GP2010P-FM)
INSERTION BURN

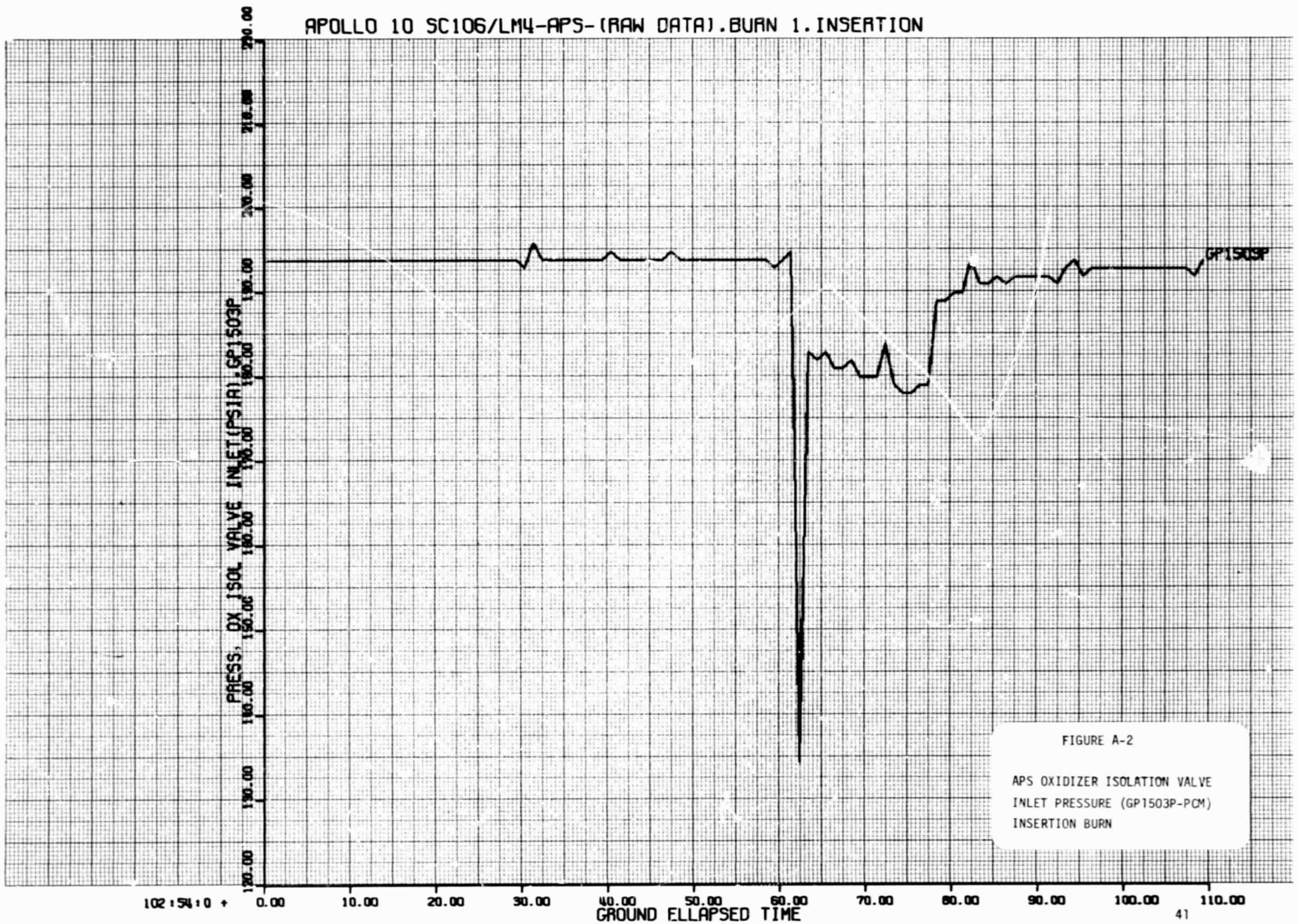


FIGURE A-2

APS OXIDIZER ISOLATION VALVE INLET PRESSURE (GP1503P-PCM) INSERTION BURN

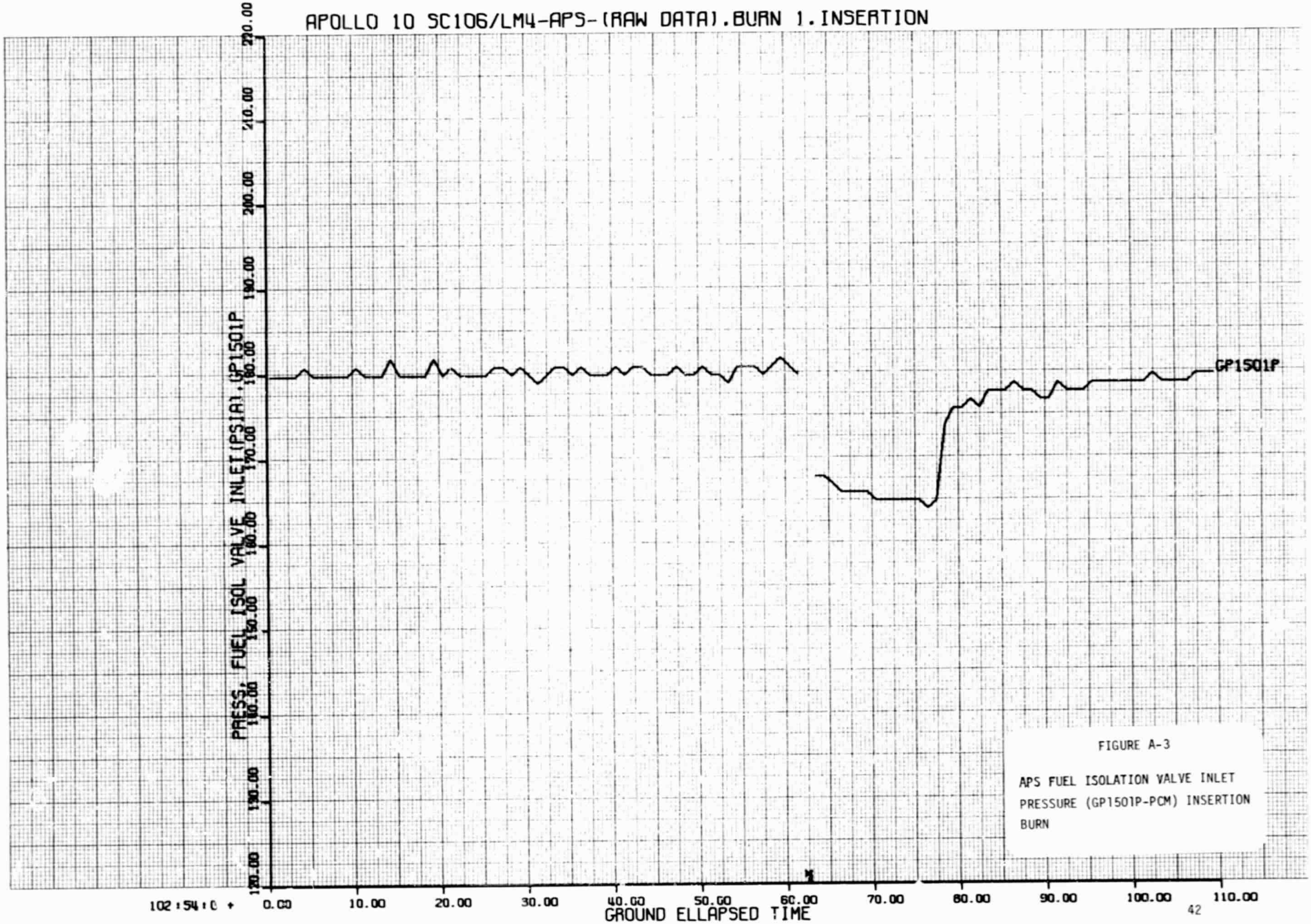


FIGURE A-3

APS FUEL ISOLATION VALVE INLET PRESSURE (GP1501P-PCM) INSERTION BURN

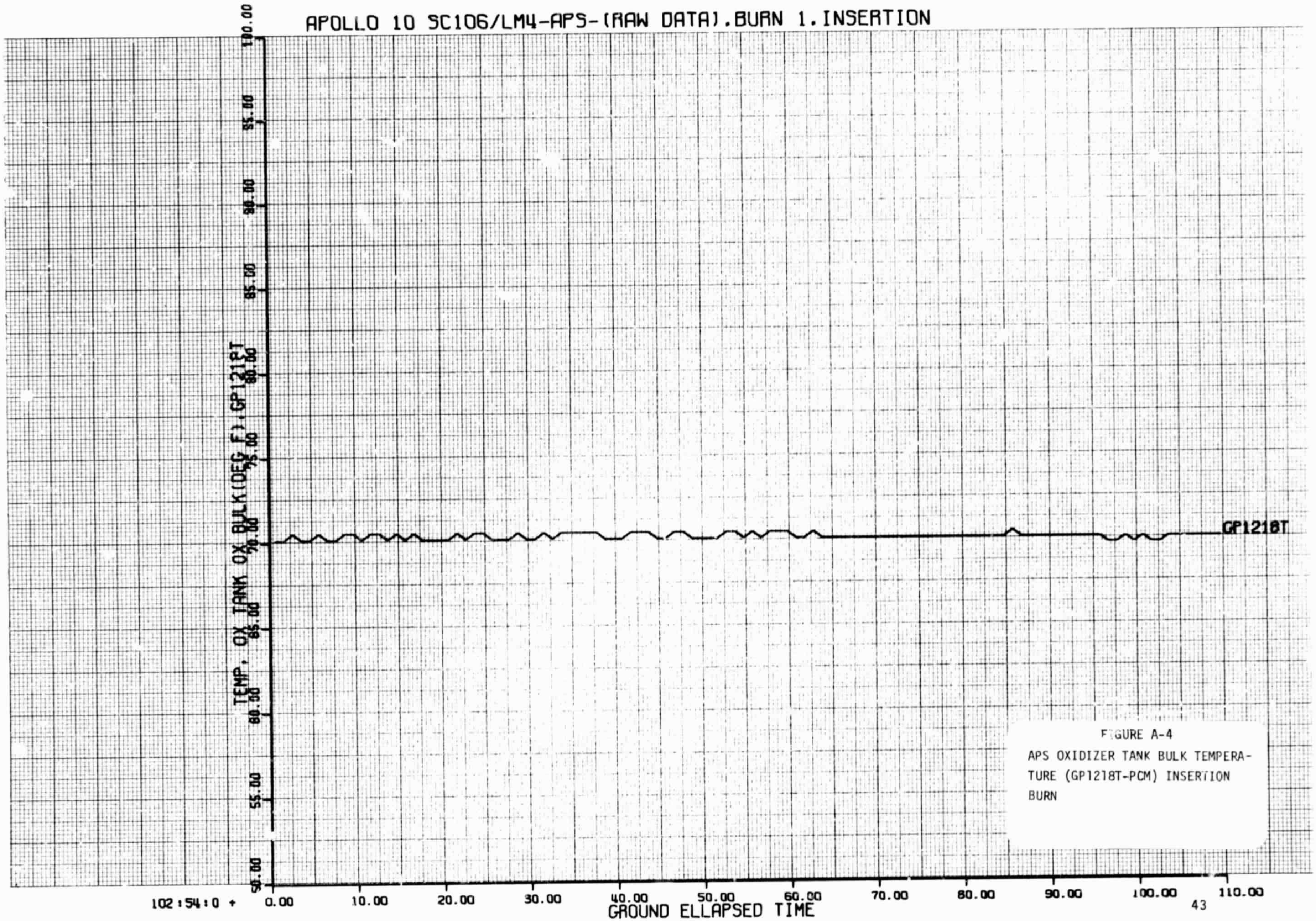


FIGURE A-4
APS OXIDIZER TANK BULK TEMPERATURE (GP1218T-PCM) INSERTION BURN

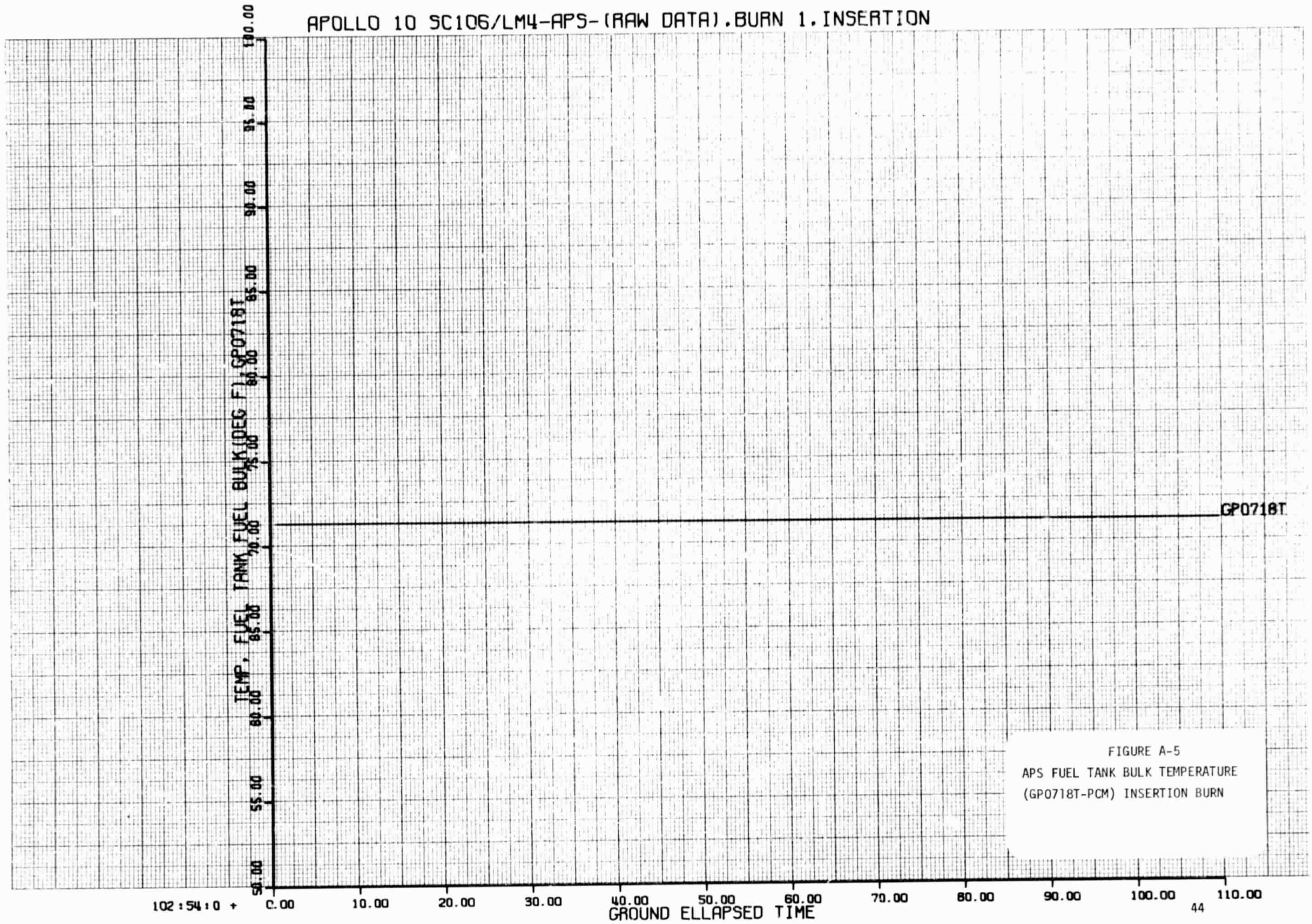


FIGURE A-5
APS FUEL TANK BULK TEMPERATURE
(GP0718T-PCM) INSERTION BURN

APOLLO 10 SC106/LM4-APS-(RAW DATA).BURN 1.INSERTION

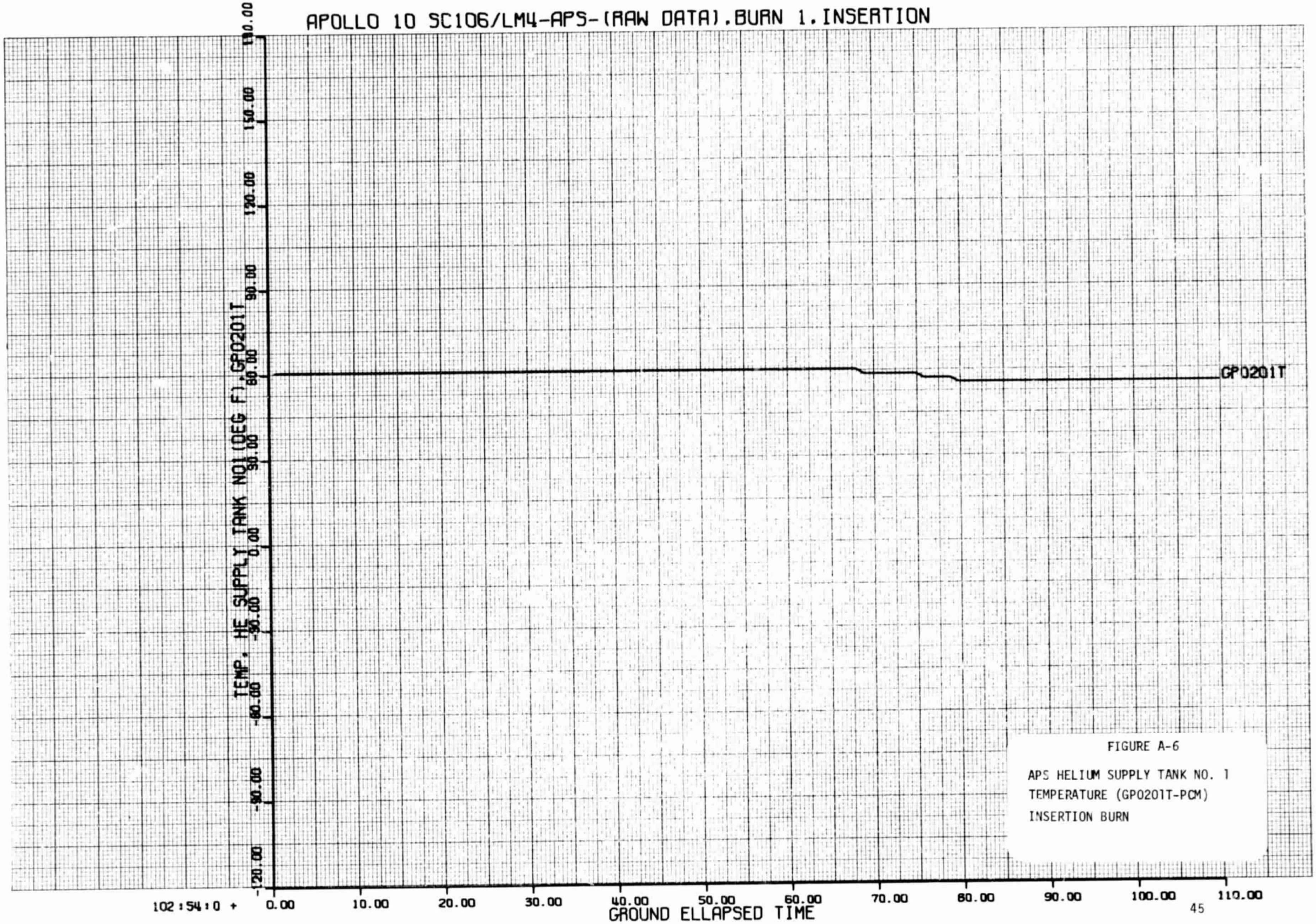


FIGURE A-6

APS HELIUM SUPPLY TANK NO. 1
TEMPERATURE (GP0201T-PCM)
INSERTION BURN

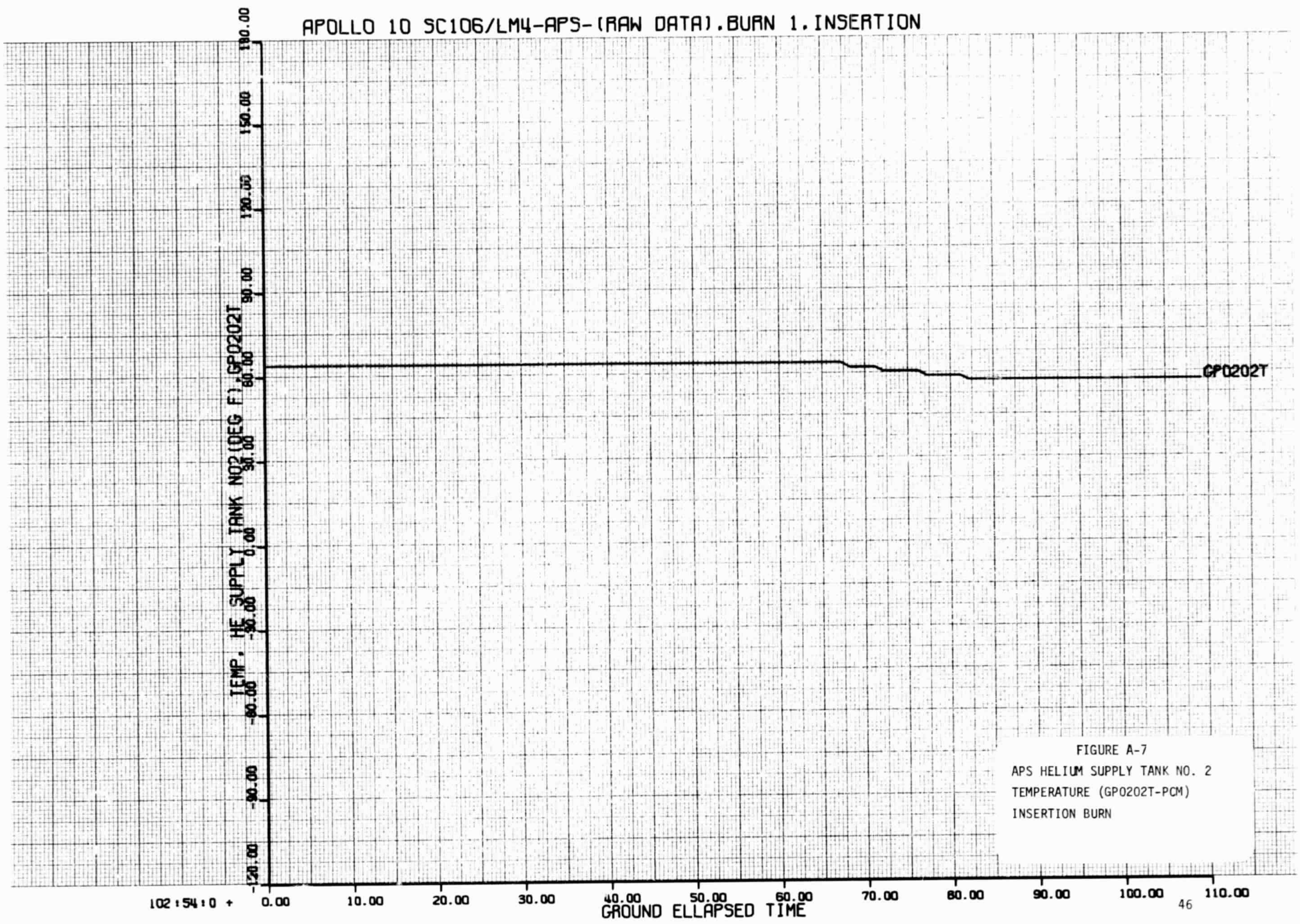


FIGURE A-7
APS HELIUM SUPPLY TANK NO. 2
TEMPERATURE (GP0202T-PCM)
INSERTION BURN

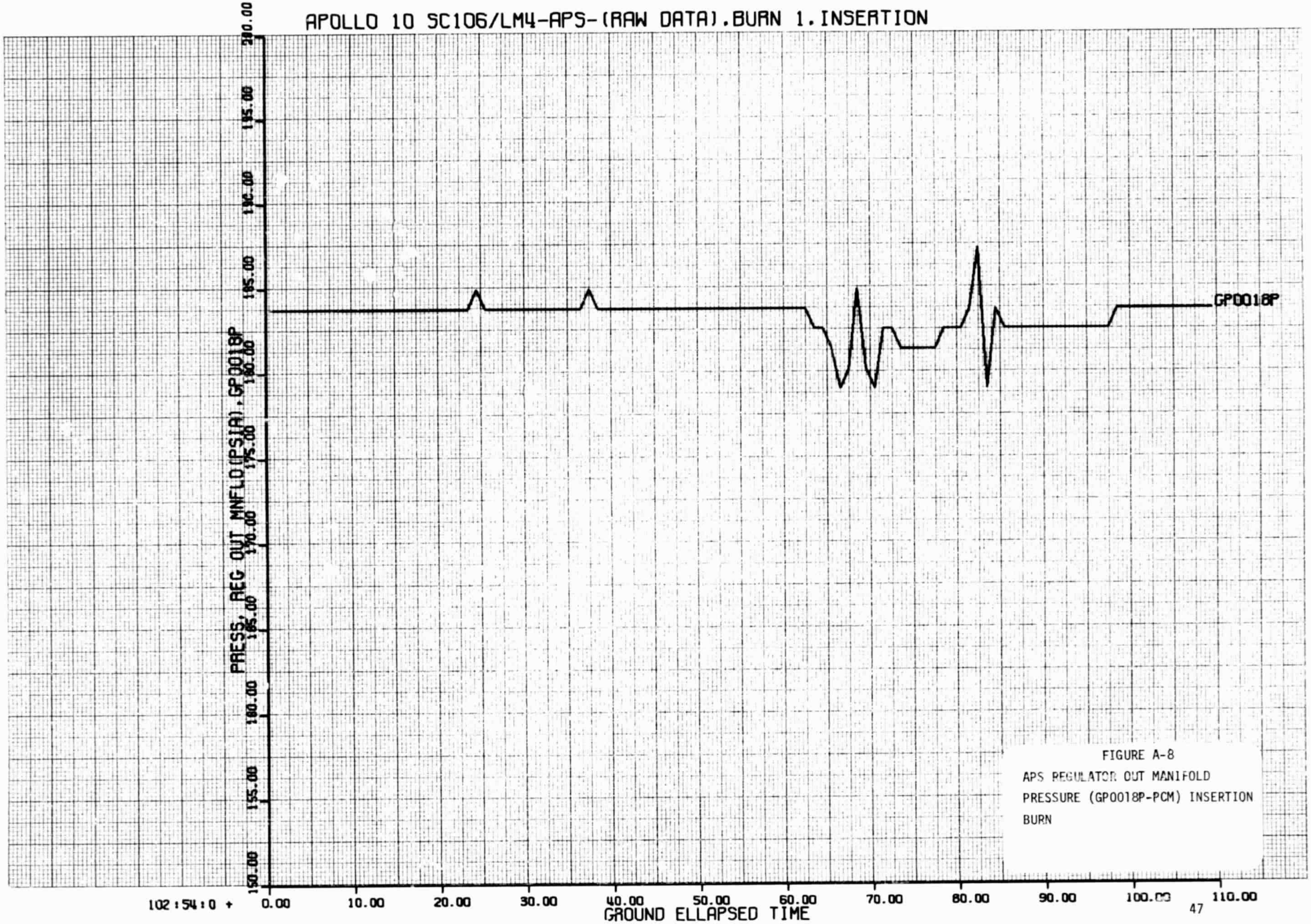


FIGURE A-8
APS REGULATOR OUT MANIFOLD PRESSURE (GP0018P-PCM) INSERTION BURN

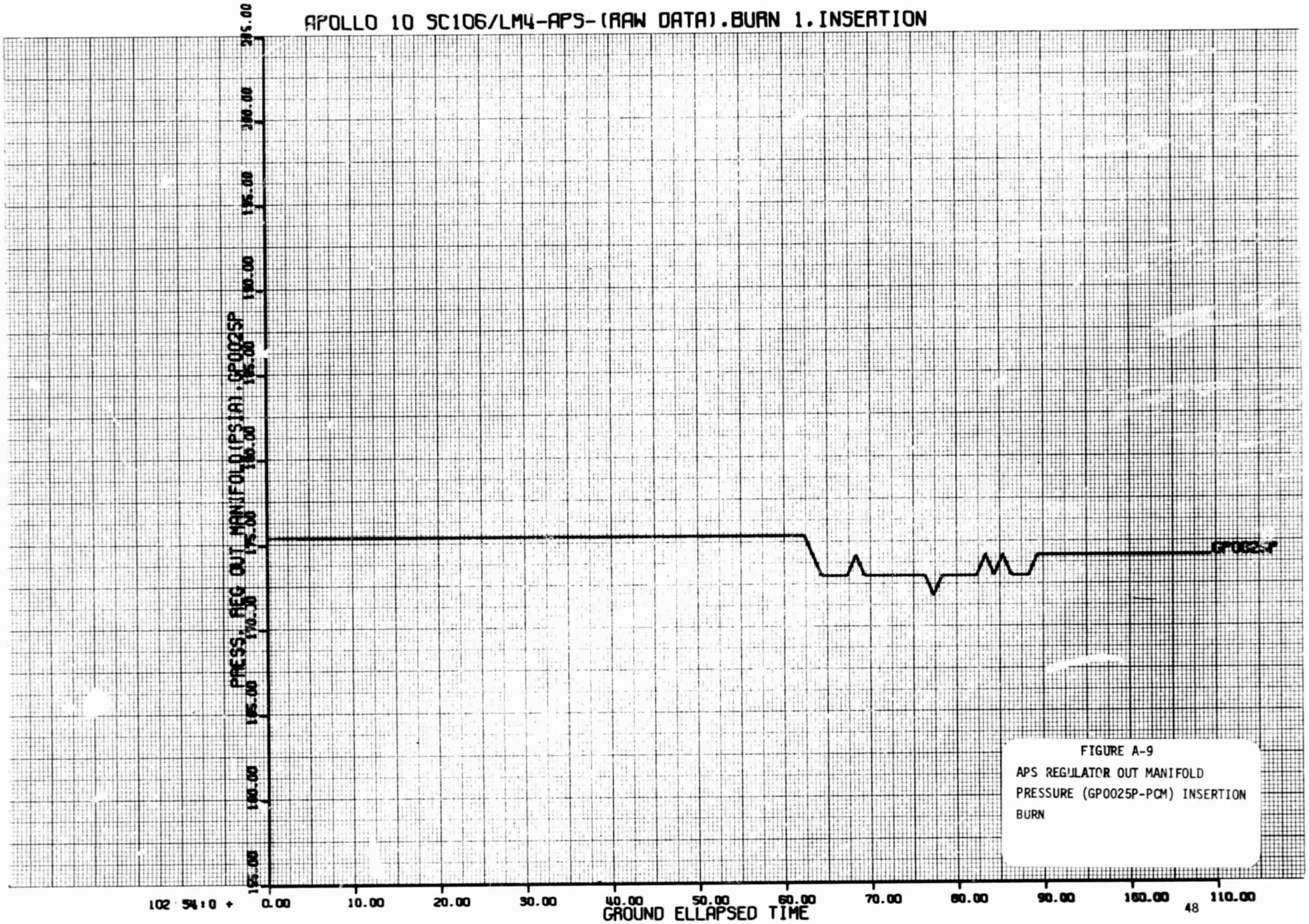


FIGURE A-9
APS REGULATOR OUT MANIFOLD PRESSURE (GP0025P-PCM) INSERTION BURN

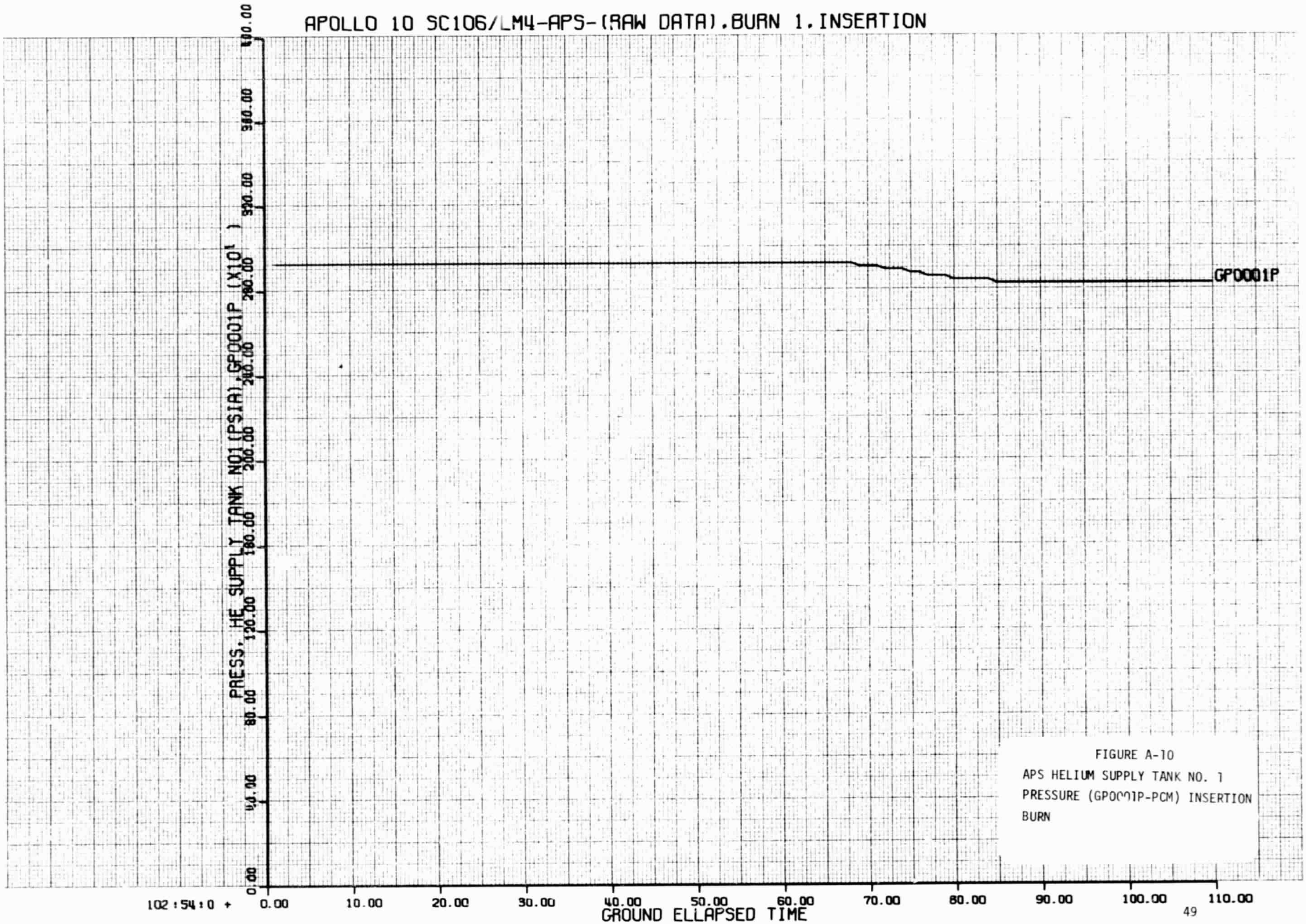


FIGURE A-10

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

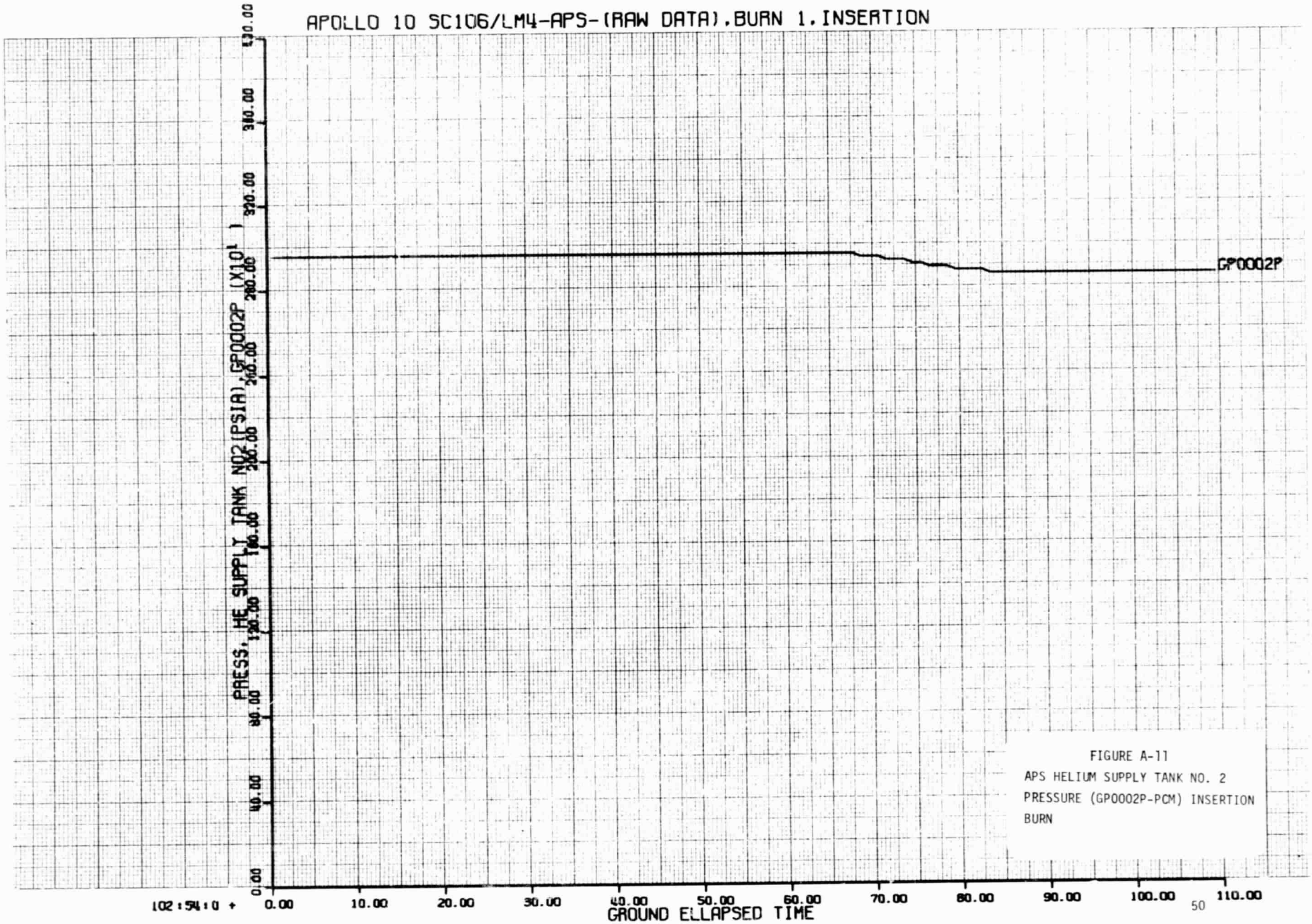
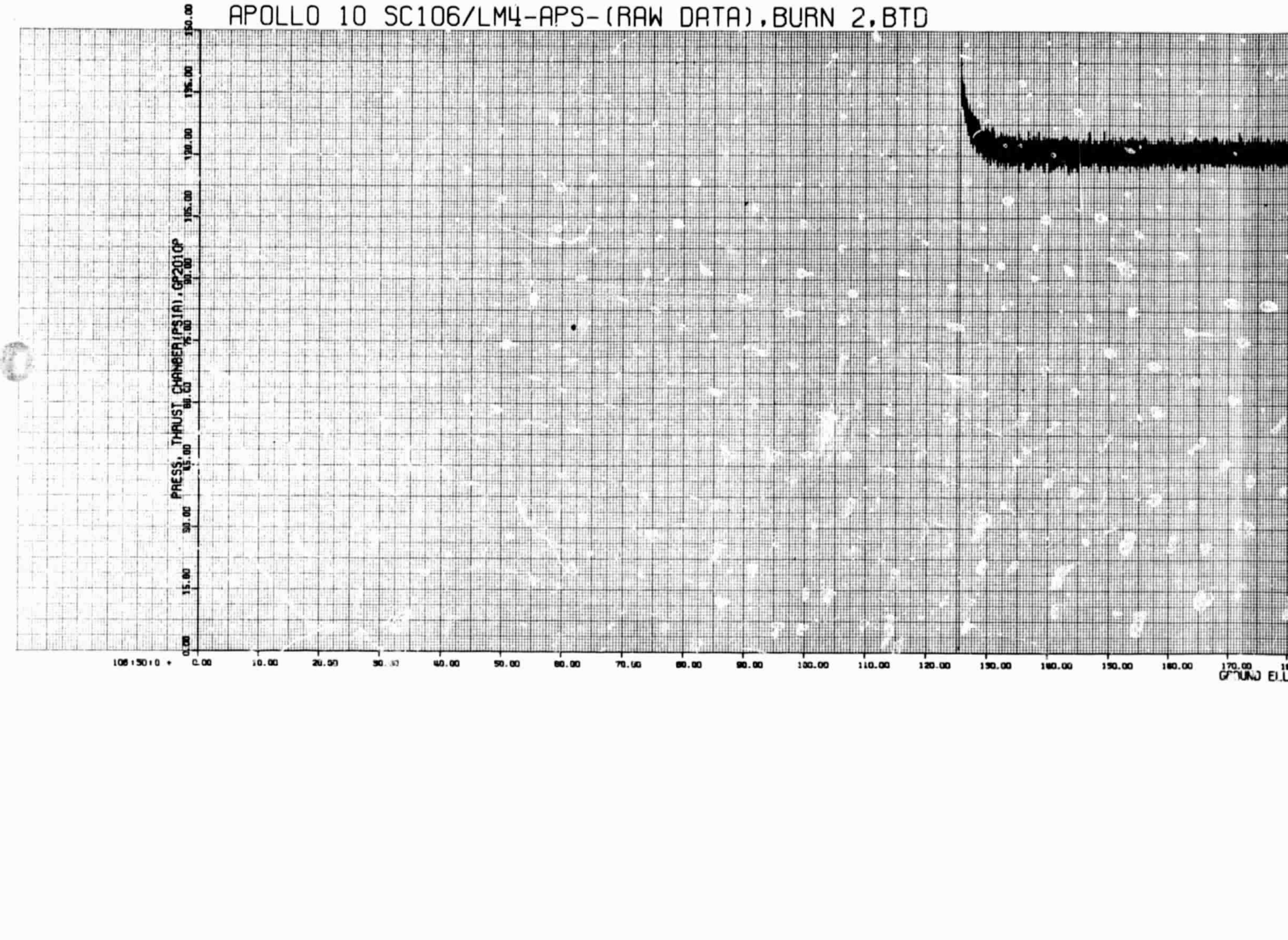


FIGURE A-11
APS HELIUM SUPPLY TANK NO. 2
PRESSURE (GP0002P-PCM) INSERTION
BURN

APOLLO 10 SC106/LM4-APS-(RAW DATA),BURN 2,BTD



FOLDOUT FRAME

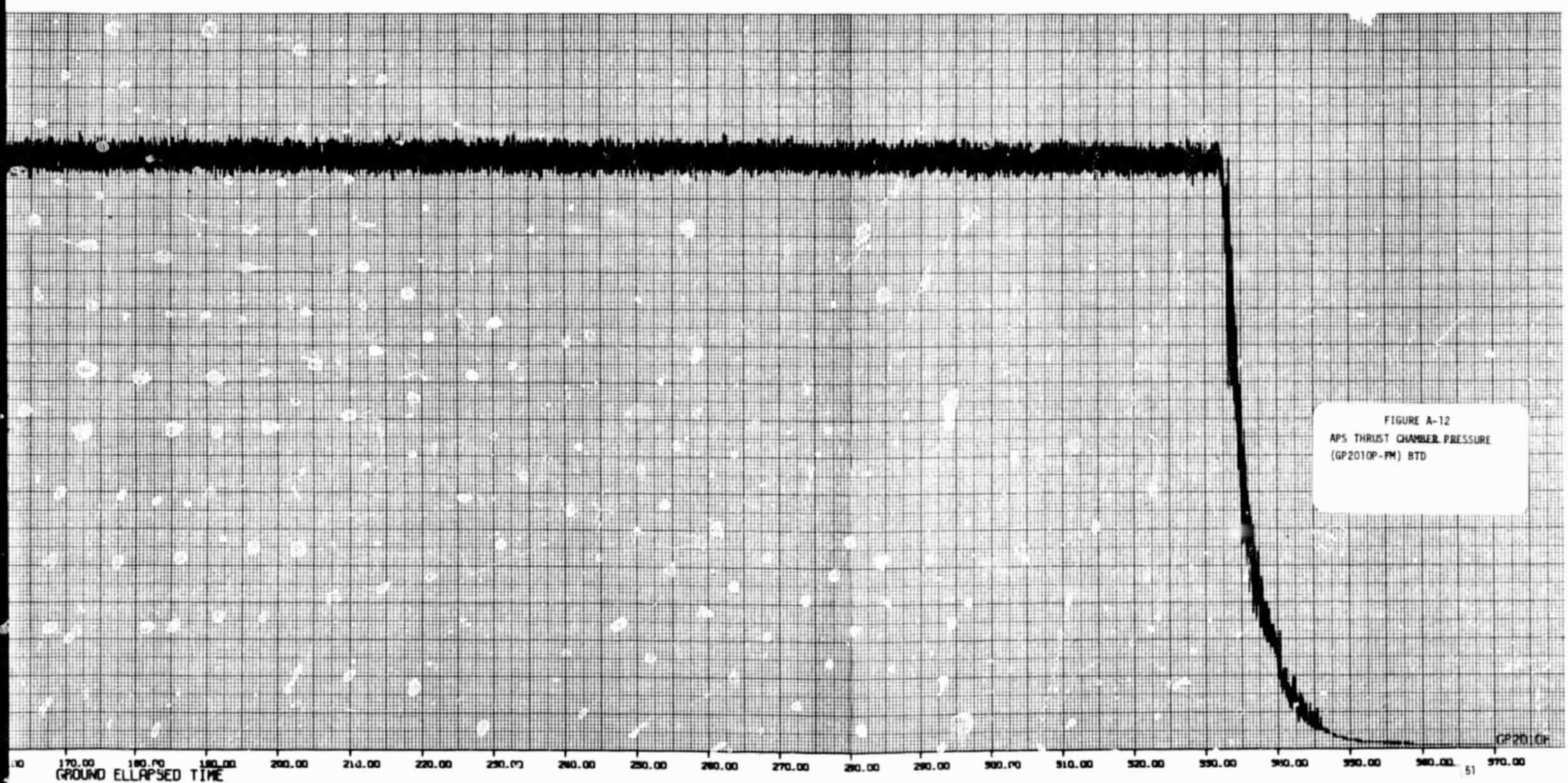
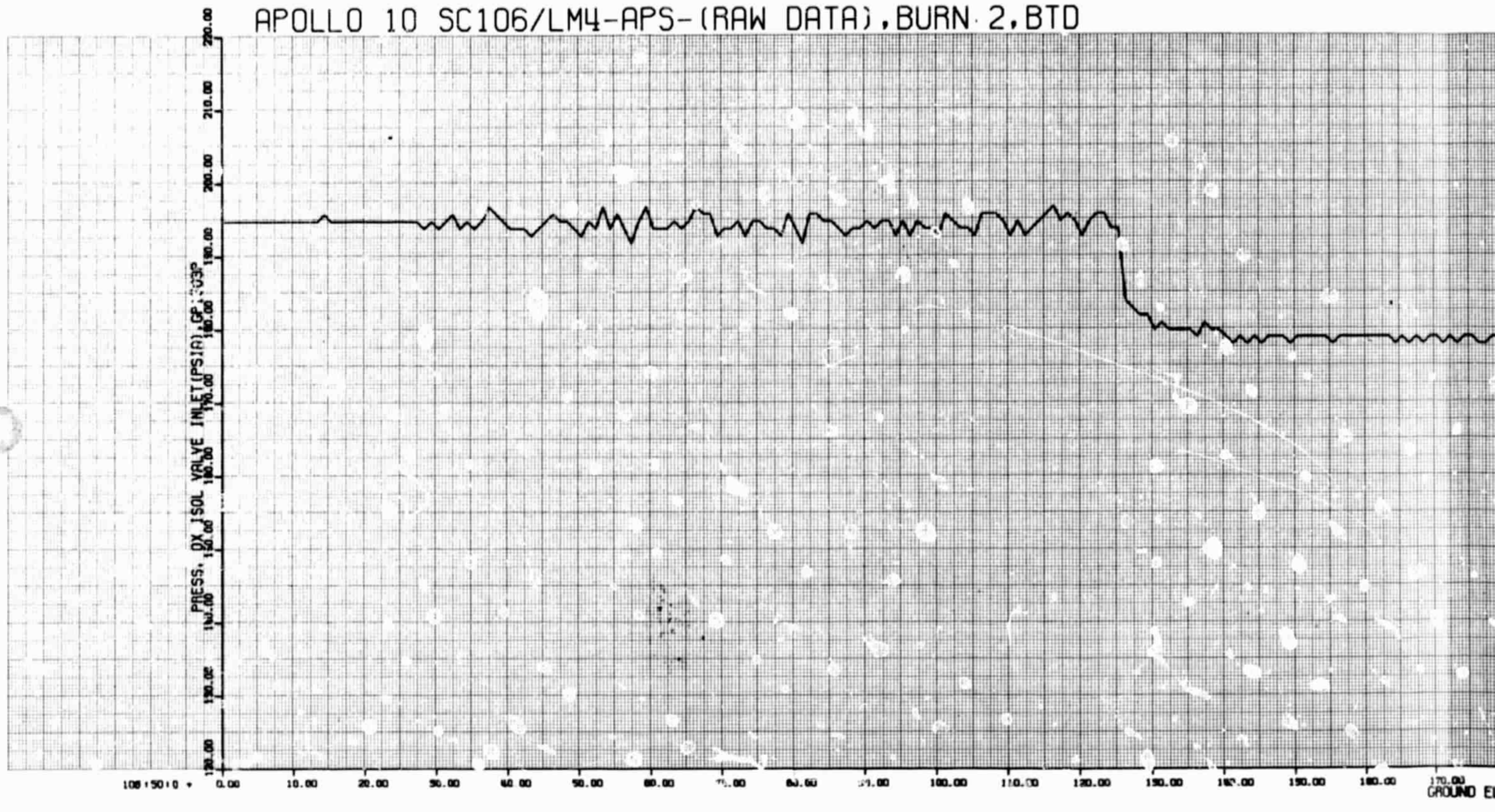


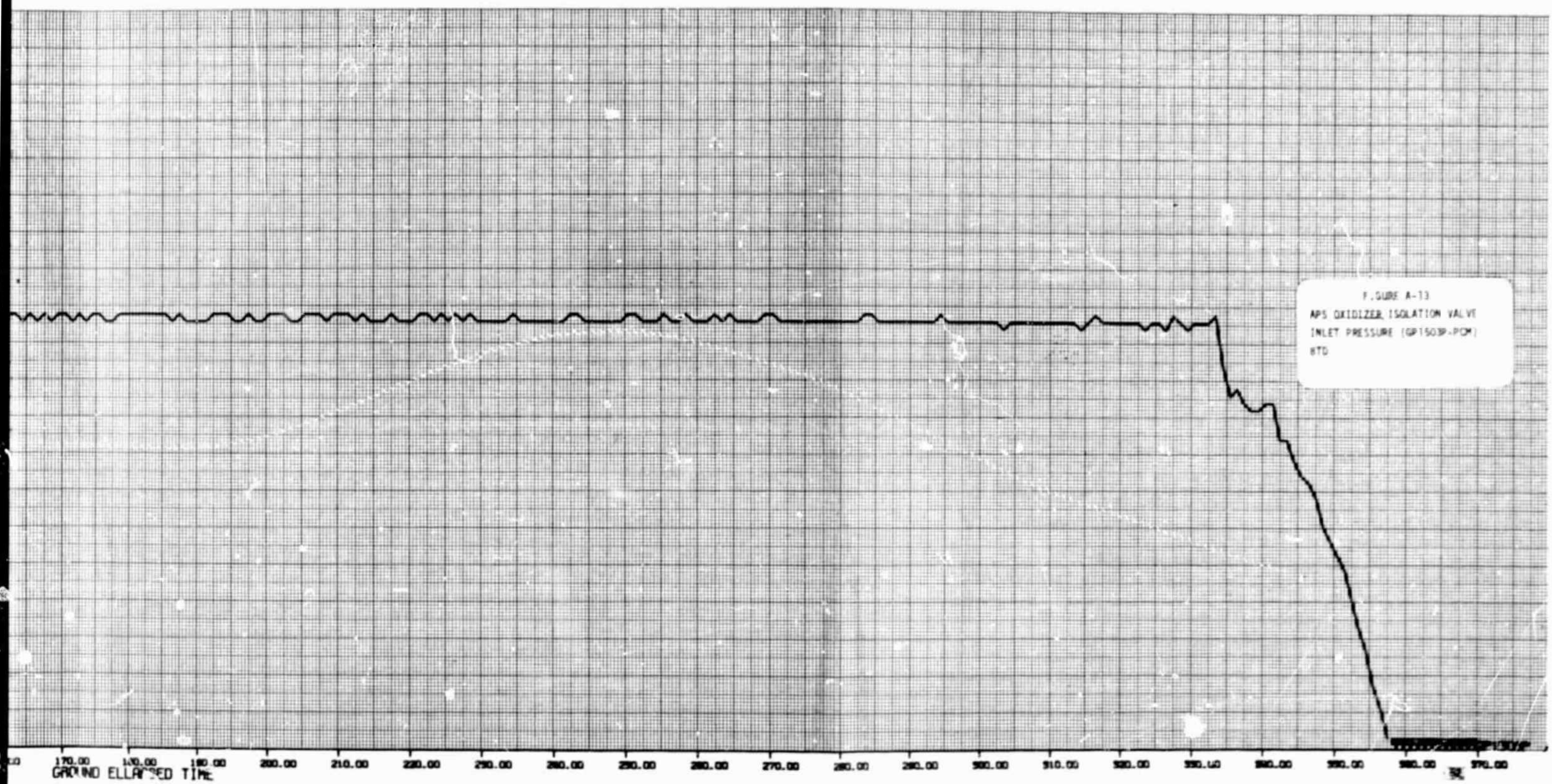
FIGURE A-12
APS THRUST CHAMBER PRESSURE
(GP2010P-FM) BTD

FOLDOUT FRAME 2

APOLLO 10 SC106/LM4-APS-(RAW DATA),BURN 2,BTD

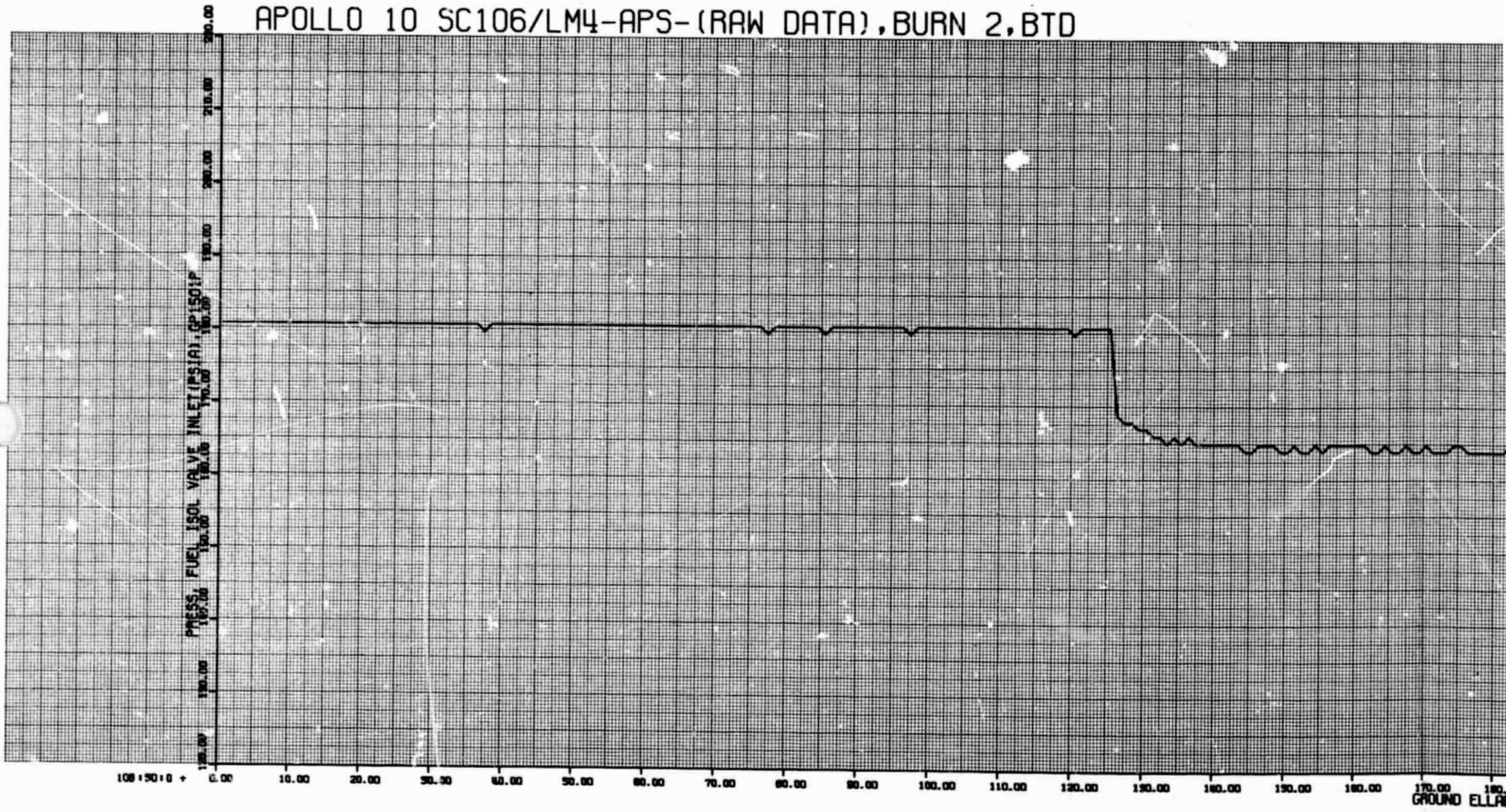


FOLDOUT FRAME 1



FOLDOUT FRAME 2

APOLLO 10 SC106/LM4-APS-(RAW DATA),BURN 2,BTD



FOLDOUT FRAME /

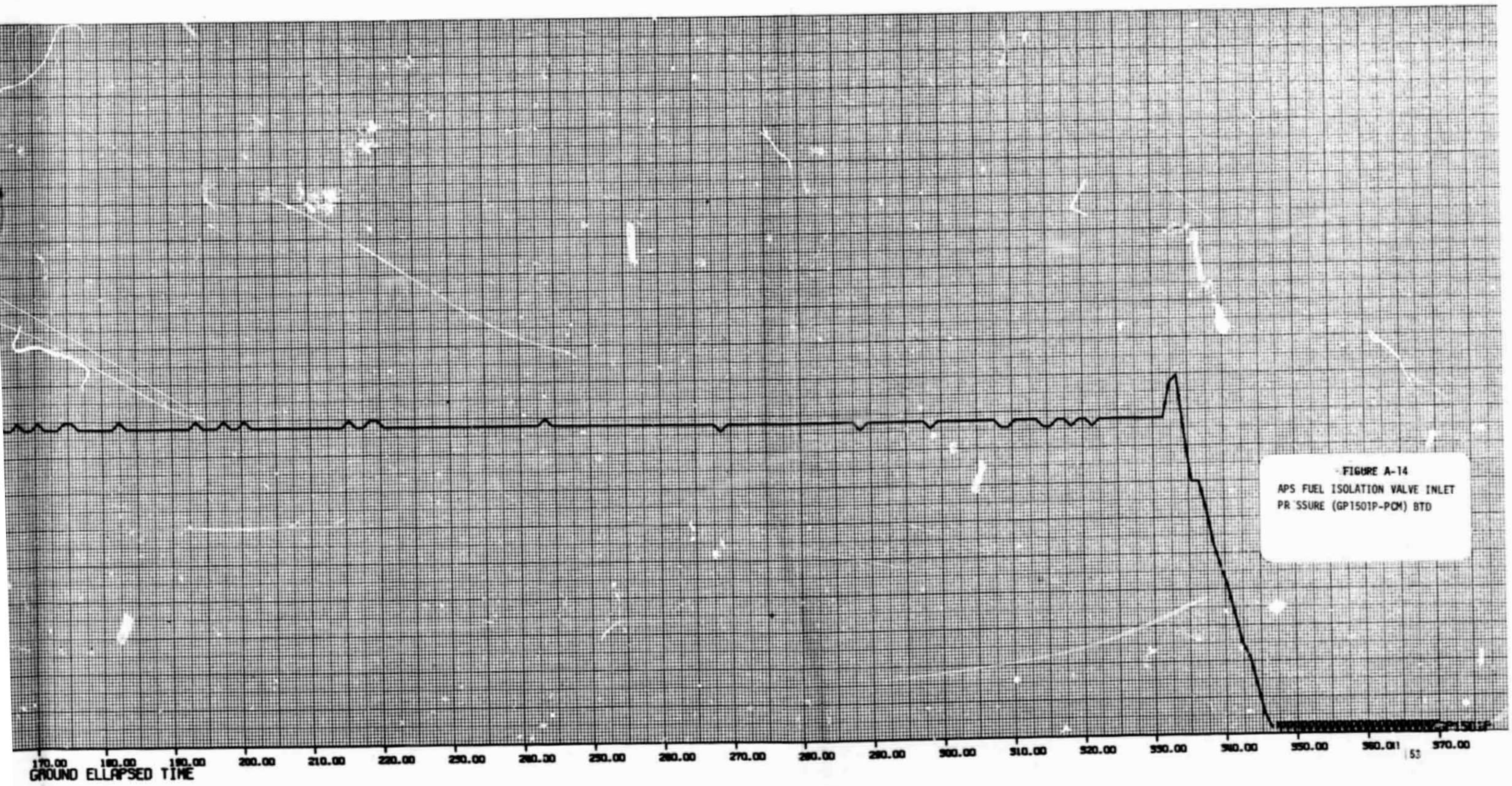
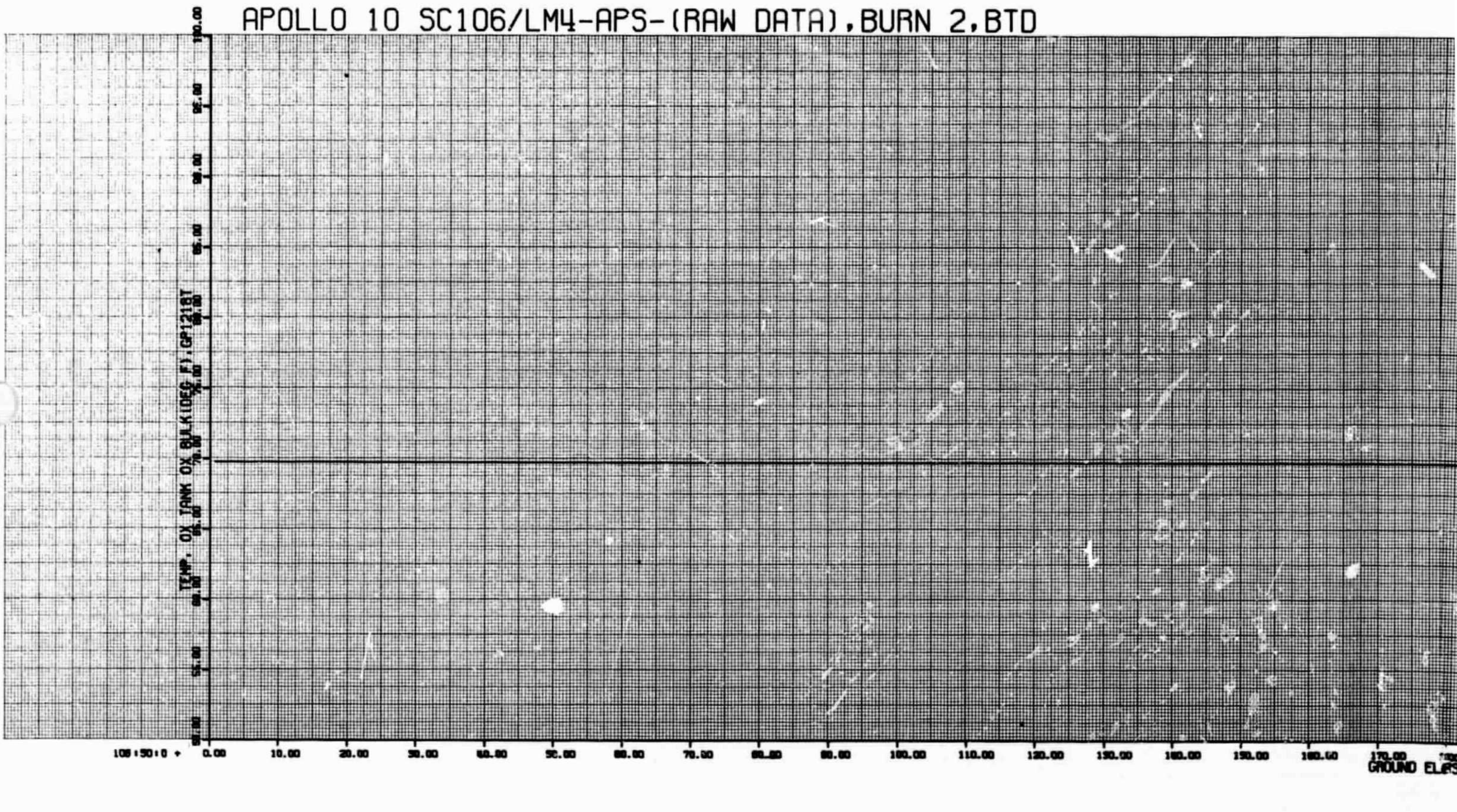


FIGURE A-14
APS FUEL ISOLATION VALVE INLET
PR SSURE (GP1501P-PCM) BTD

FOLDOUT FRAME 2



FOLDOUT FRAME /

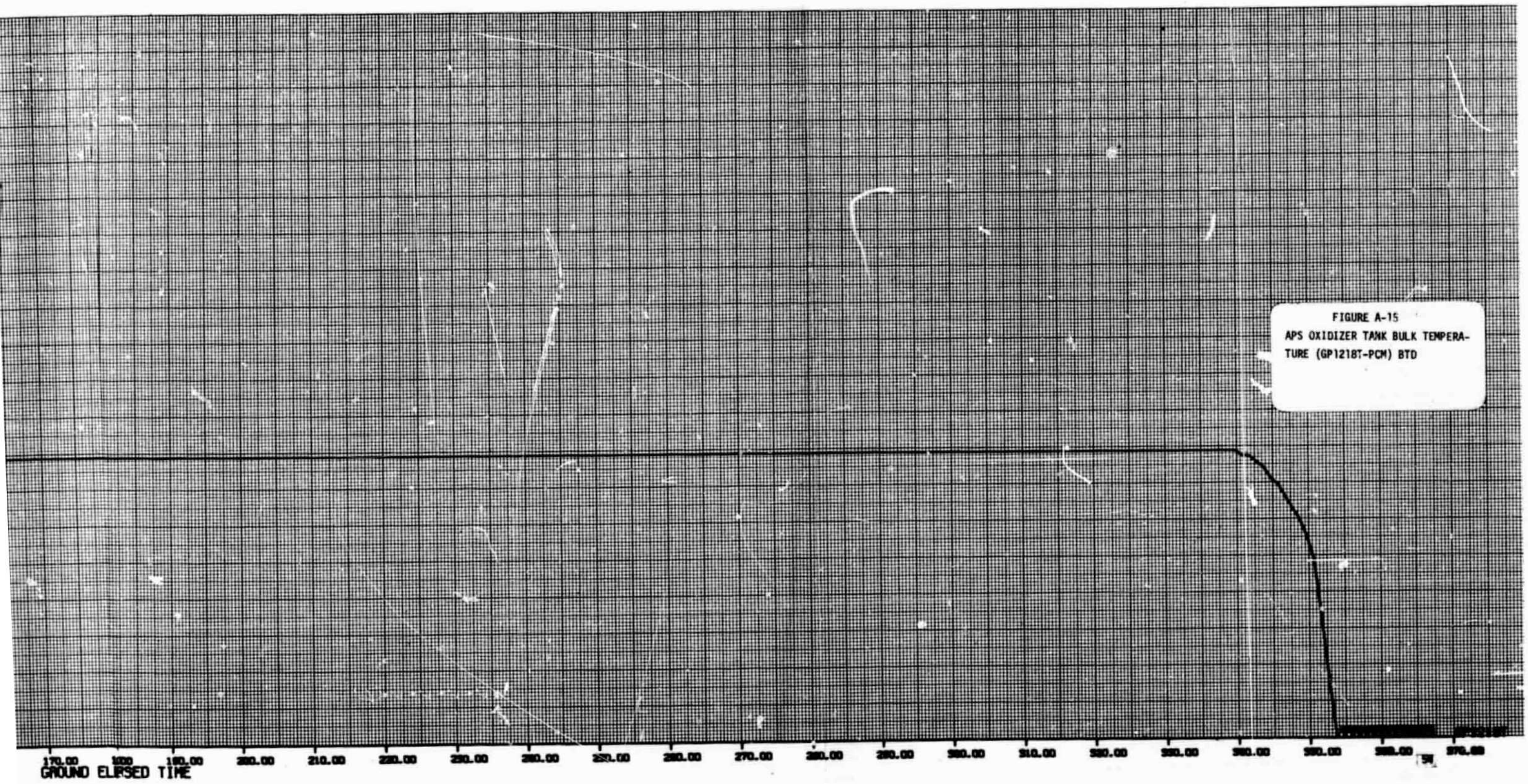
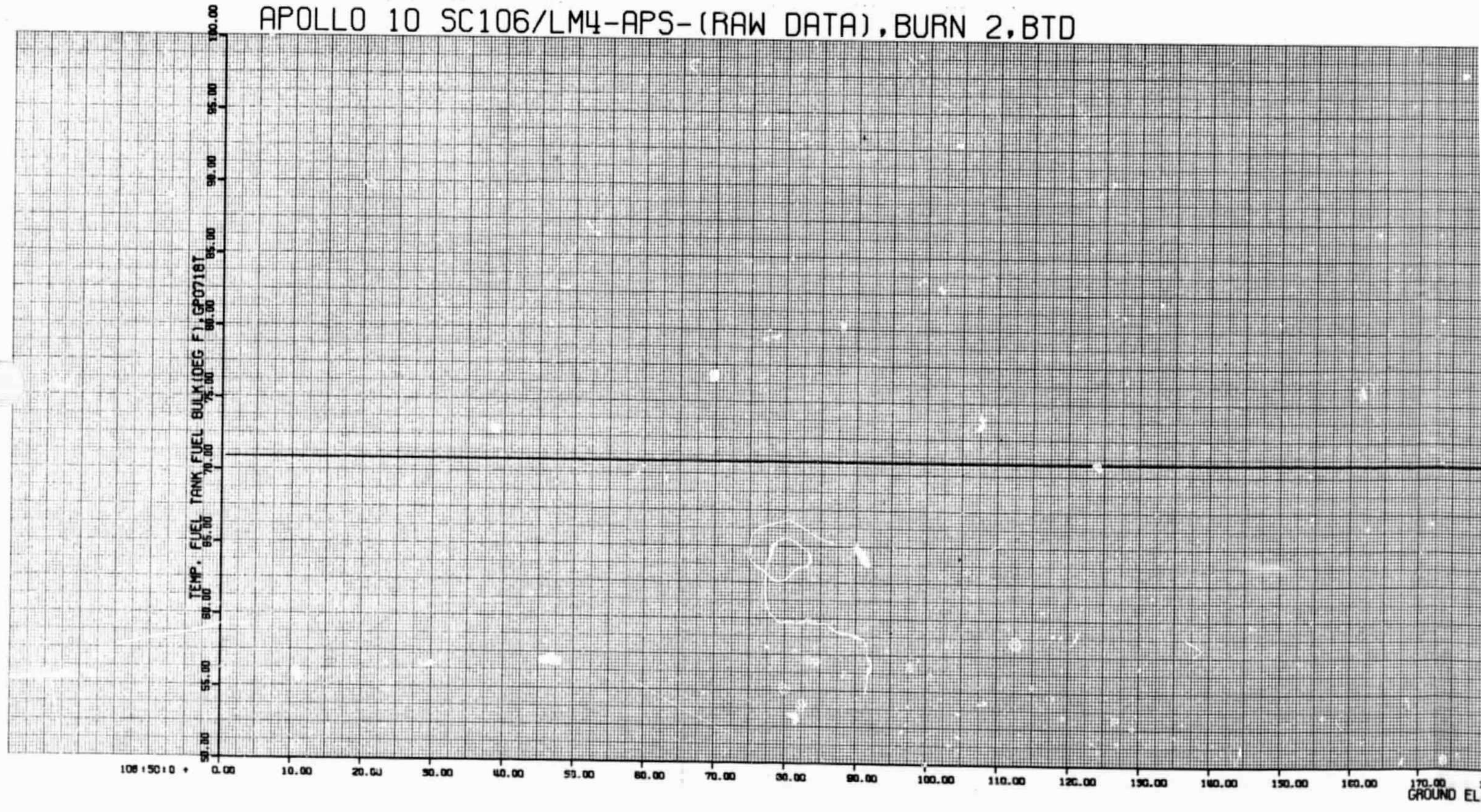


FIGURE A-15
APS OXIDIZER TANK BULK TEMPERATURE (GP1218T-PCM) BTD

FOLDOUT FRAME 2



FOLDOUT FRAME /

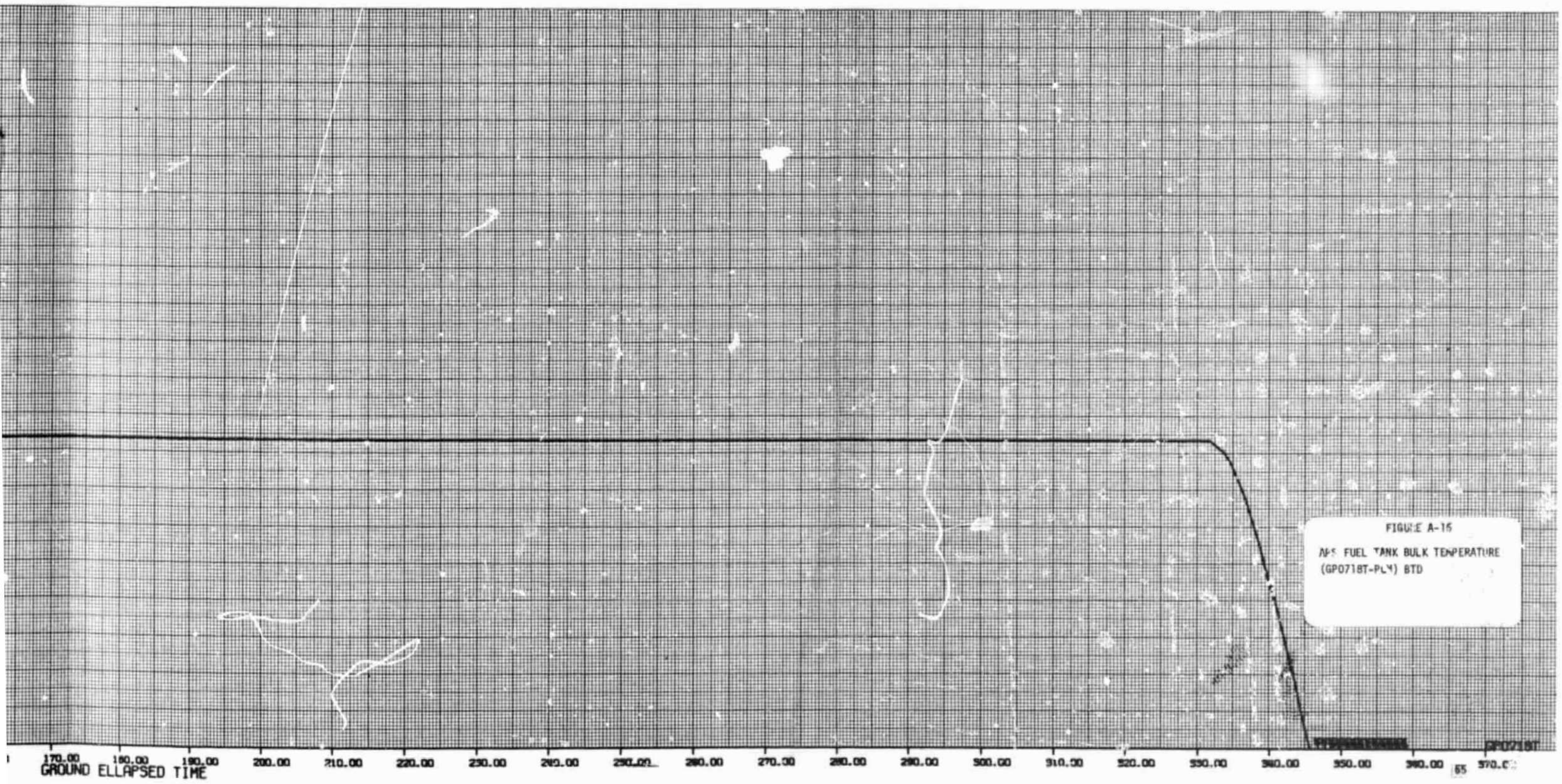


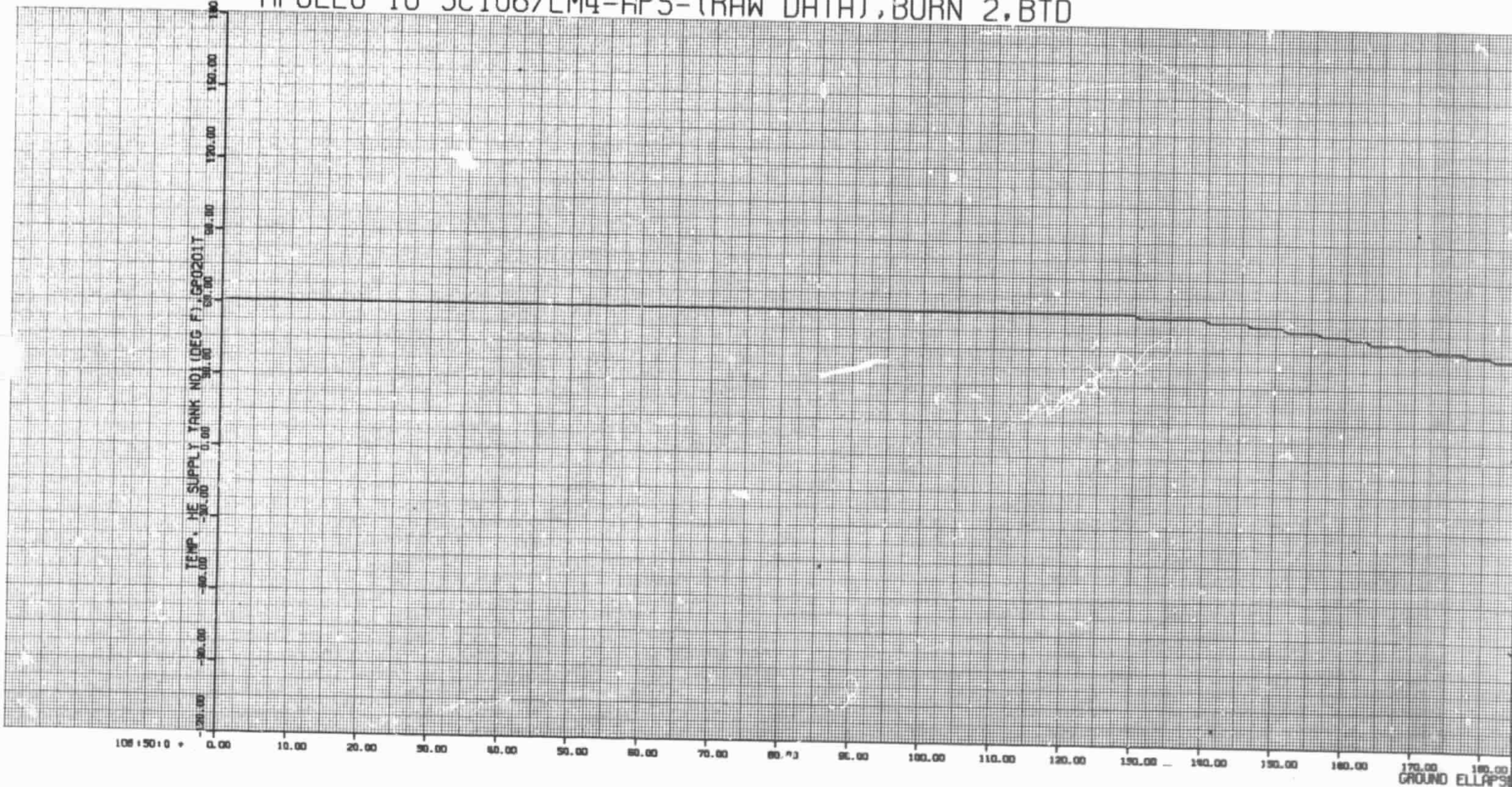
FIGURE A-15

APS FUEL TANK BULK TEMPERATURE
(GP0718T-PLM) BTD

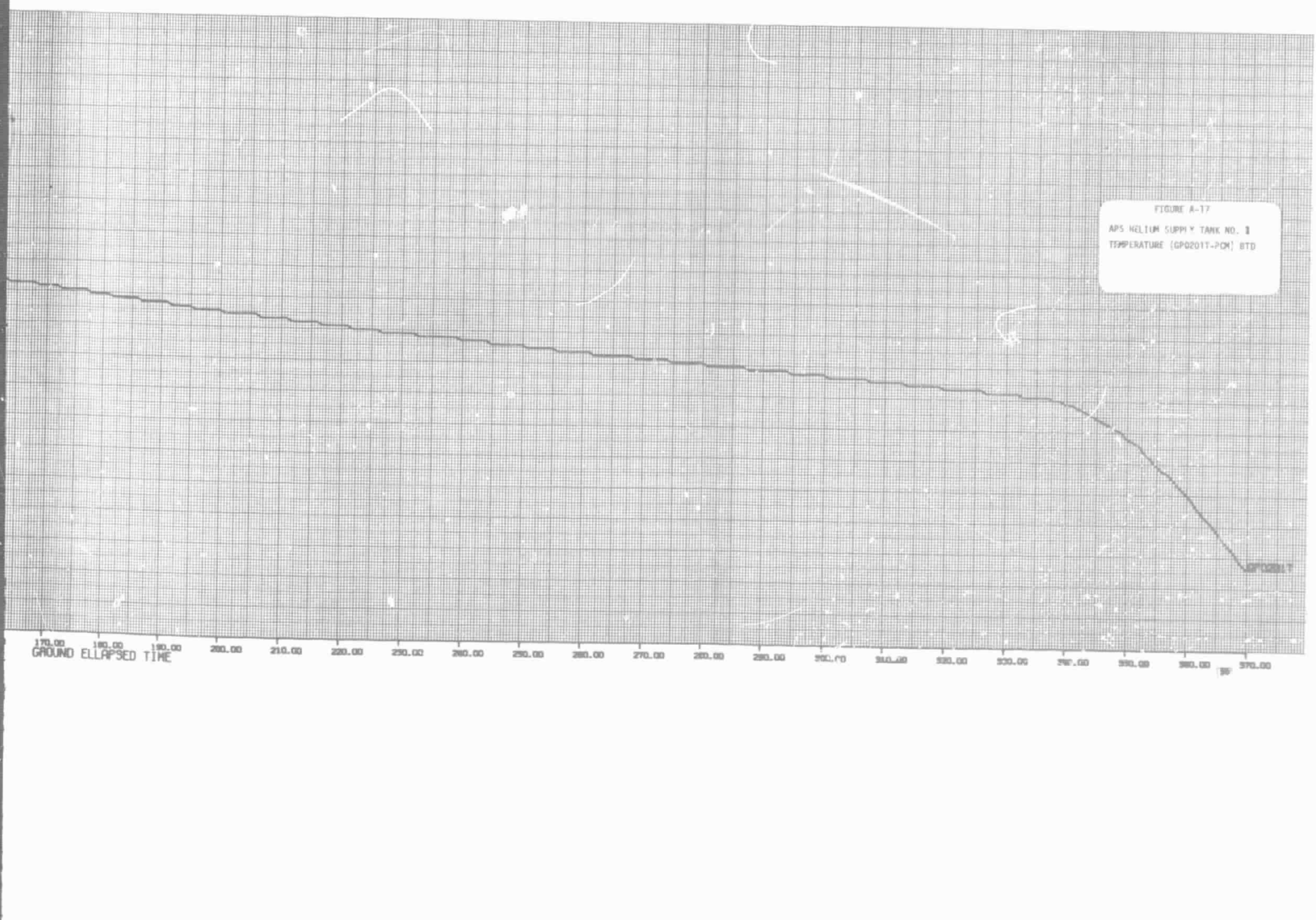
65

FOLDOUT FRAME 2

APOLLO 10 SC106/LM4-APS-(RAW DATA),BURN 2,BTD

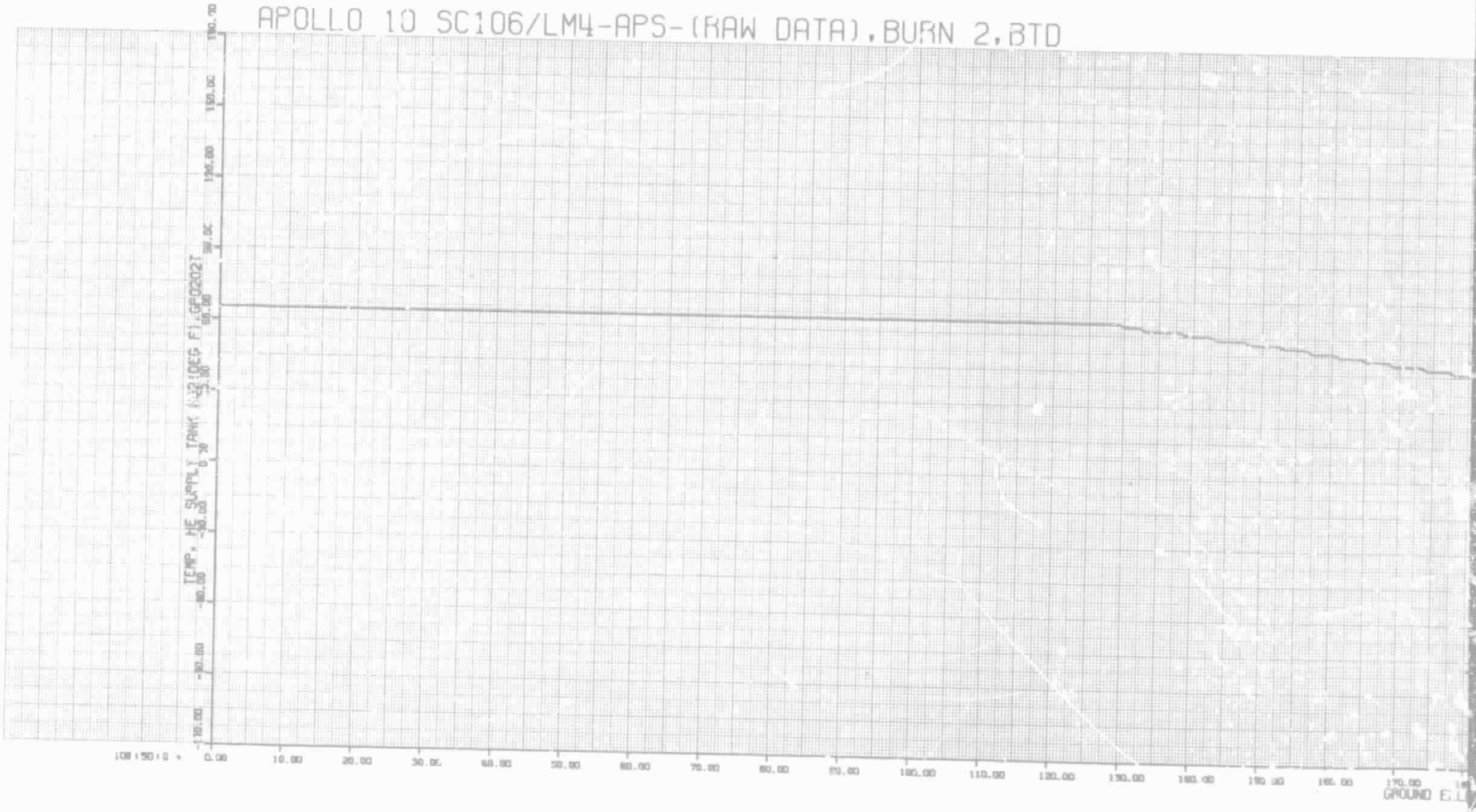


FOLDOUT FRAME /

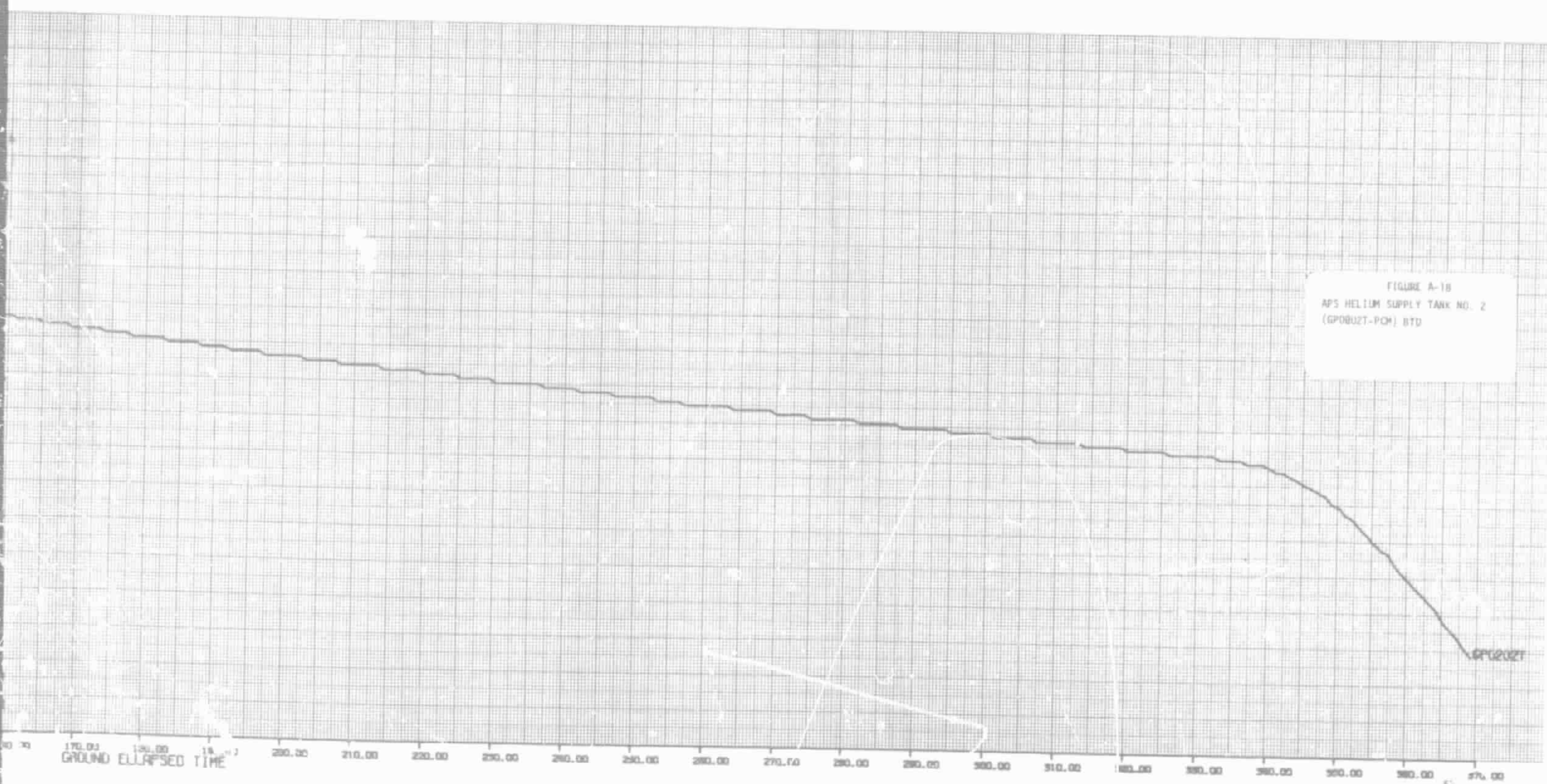


FOLDOUT FRAME 2

APOLLO 10 SC106/LM4-APS-(RAW DATA), BURN 2, BTD



FOLDOUT FRAME /

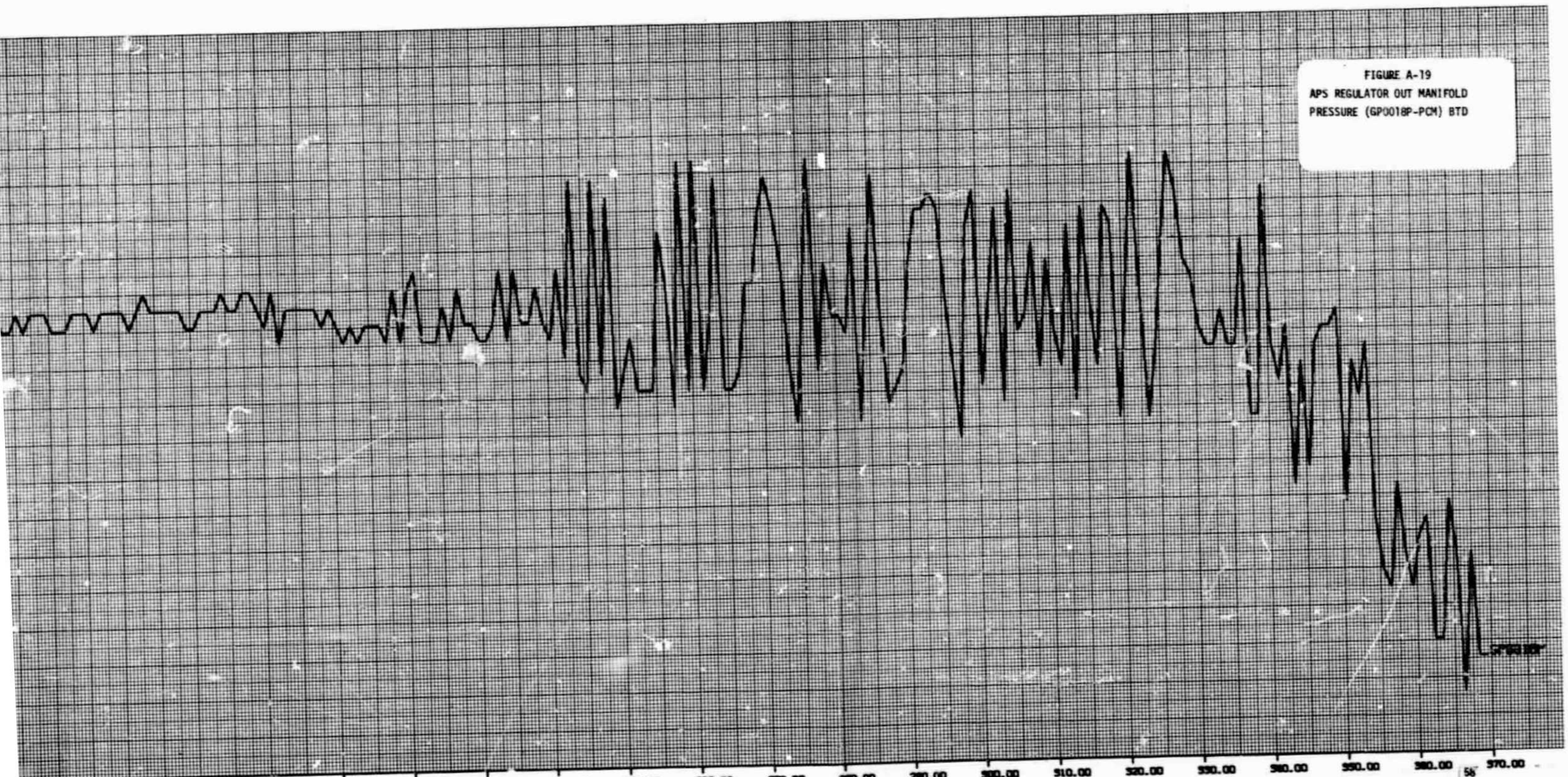


FOLDOUT FRAME 2

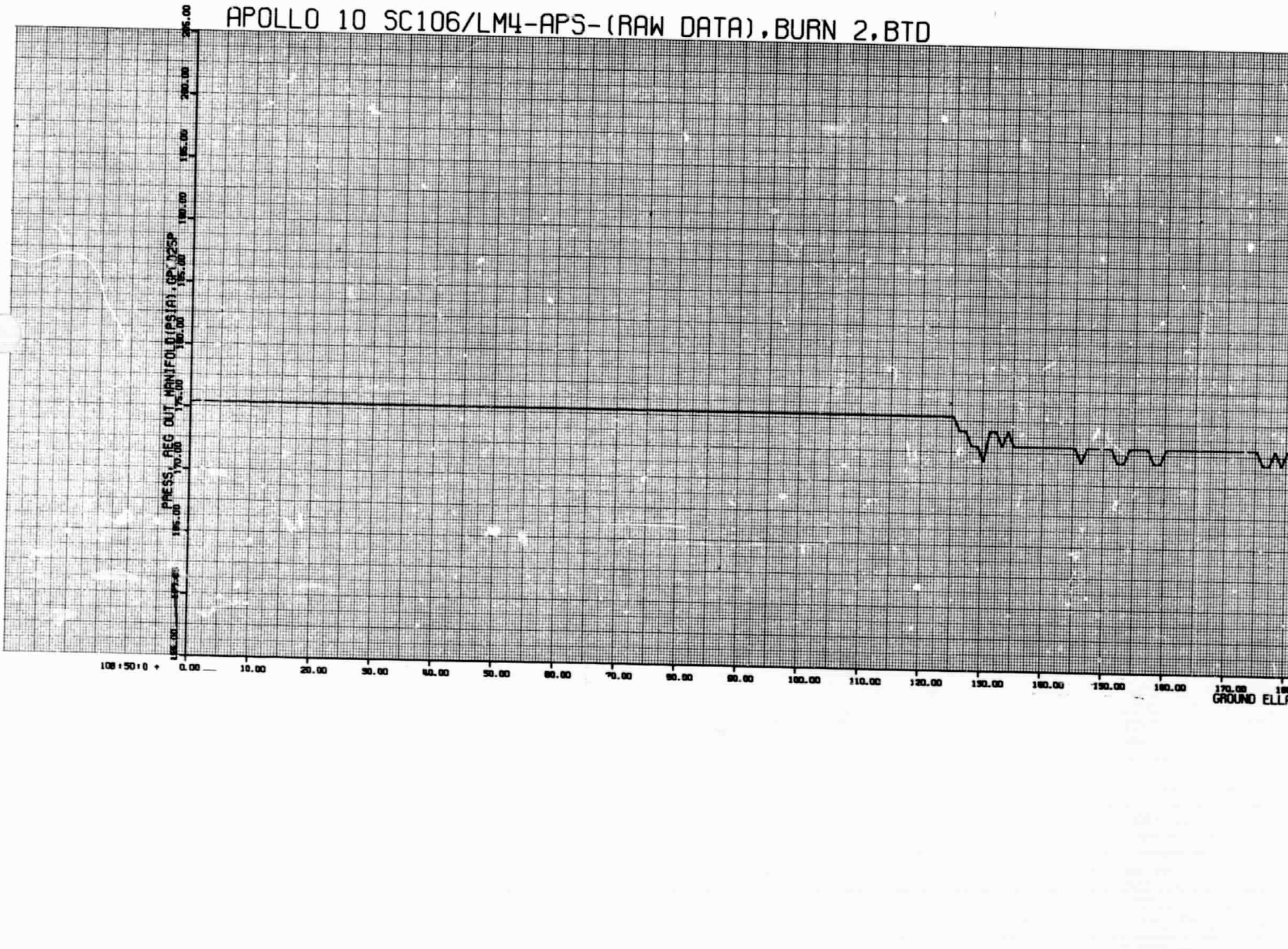
APOLLO 10 SC106/LM4-APS-(RAW DATA),BURN 2,BTD

FOLDOUT FRAME /

FIGURE A-19
APS REGULATOR OUT MANIFOLD
PRESSURE (GP0018P-PCM) BTD



FOLDOUT FRAME 2



FOLDOUT FRAME /

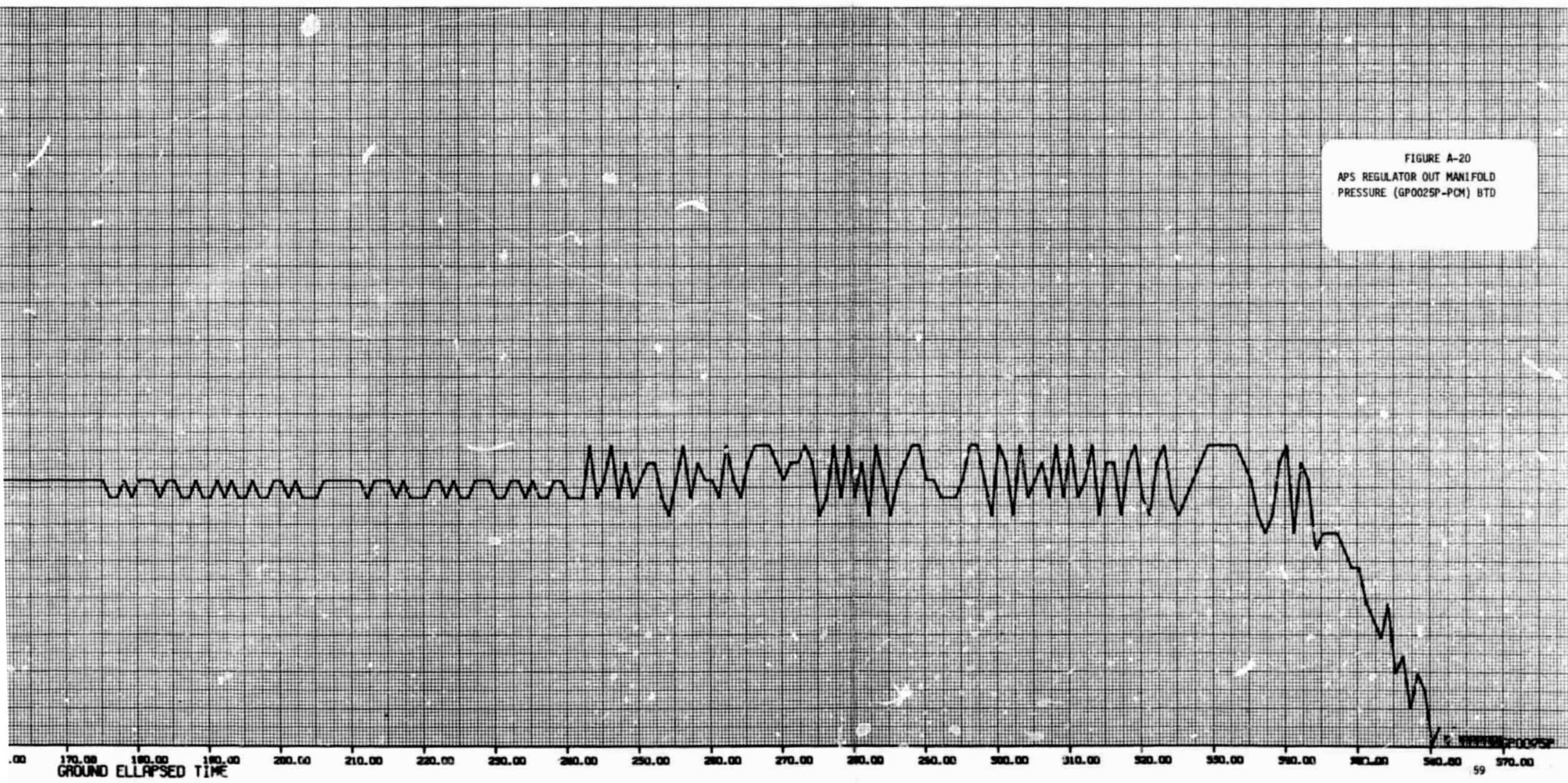
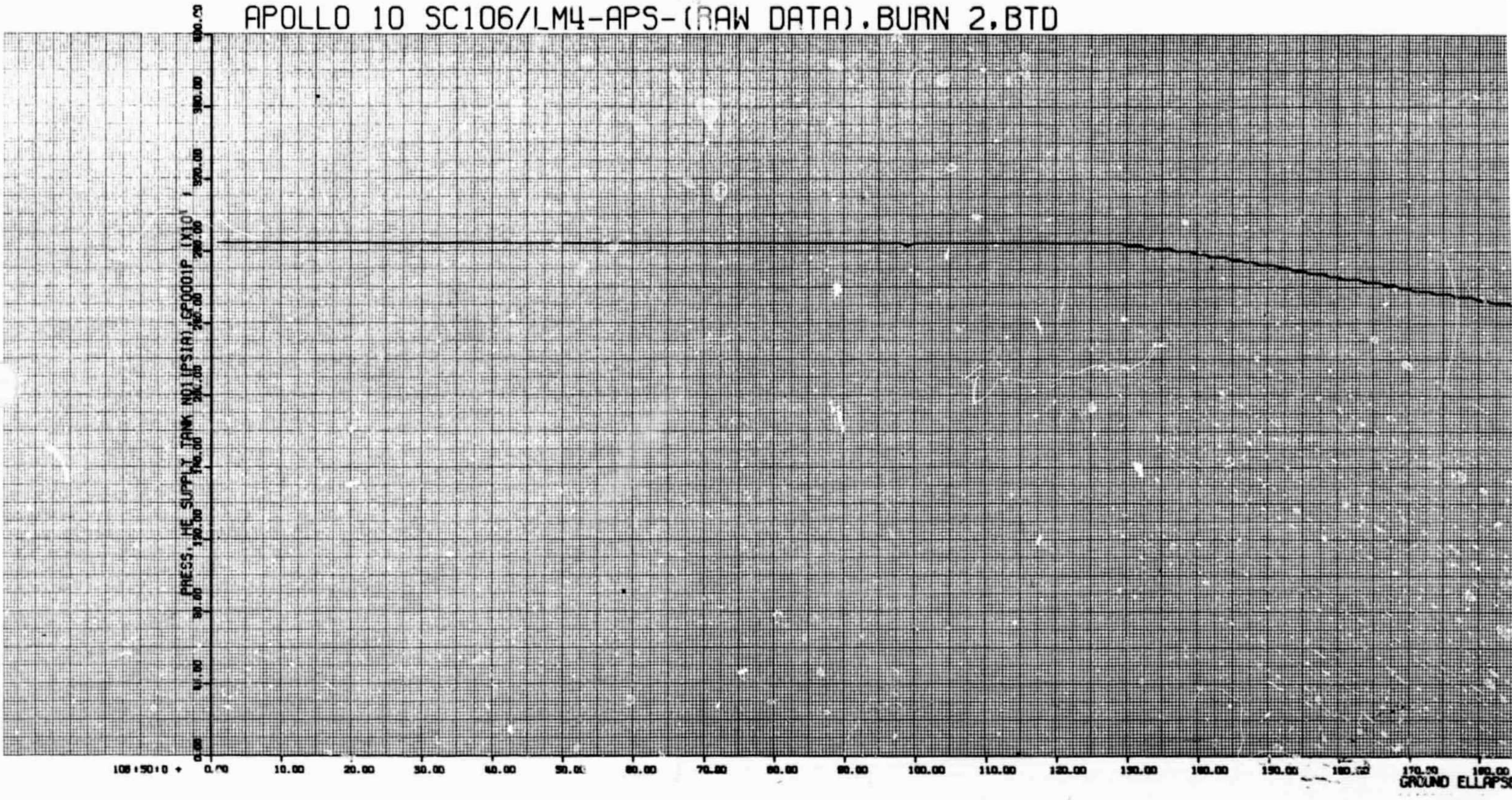


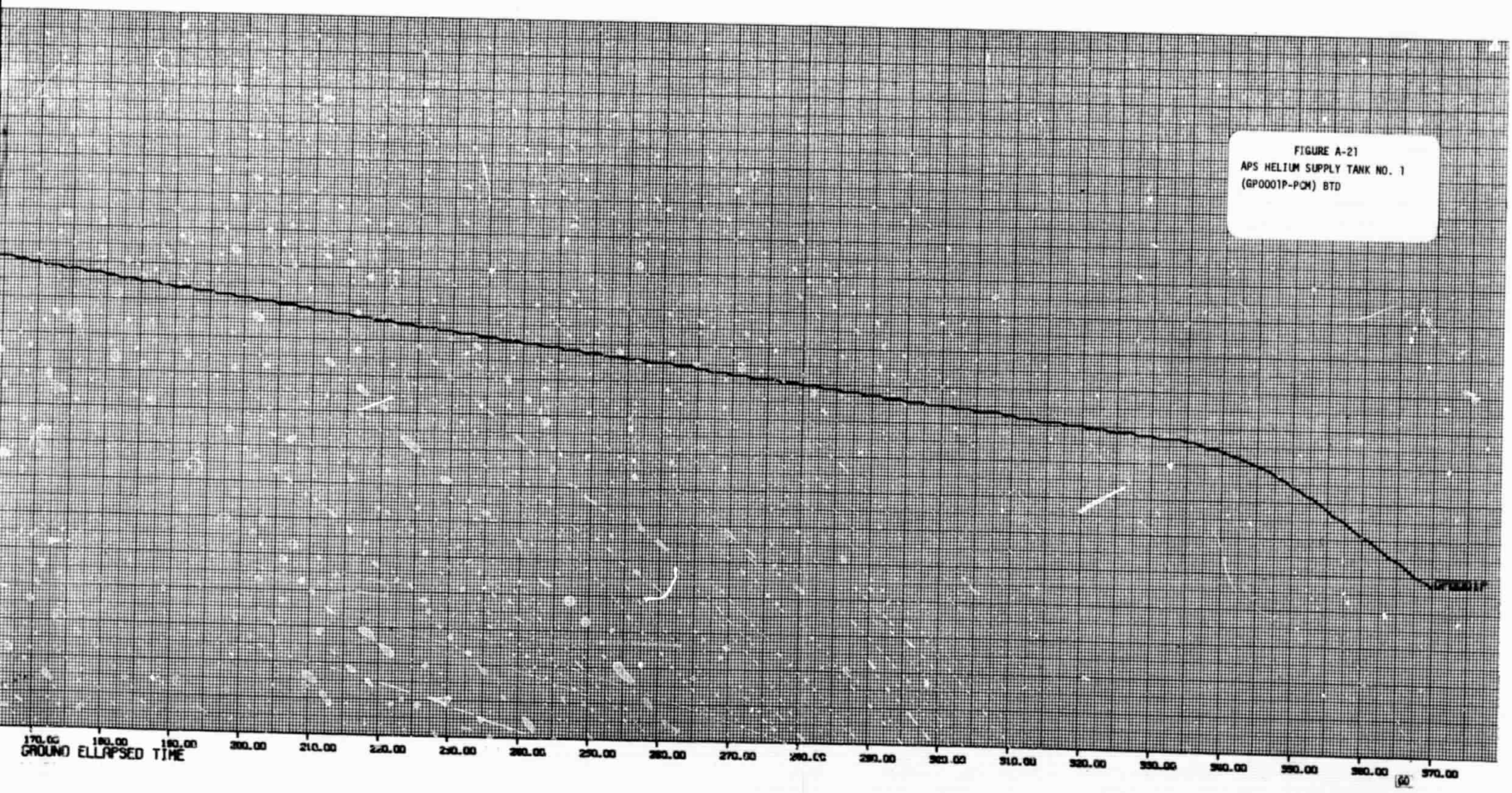
FIGURE A-20
APS REGULATOR OUT MANIFOLD PRESSURE (GP0025P-PCM) BTD

FOLDOUT FRAME 2

APOLLO 10 SC106/LM4-APS-(RAW DATA).BURN 2,BTD

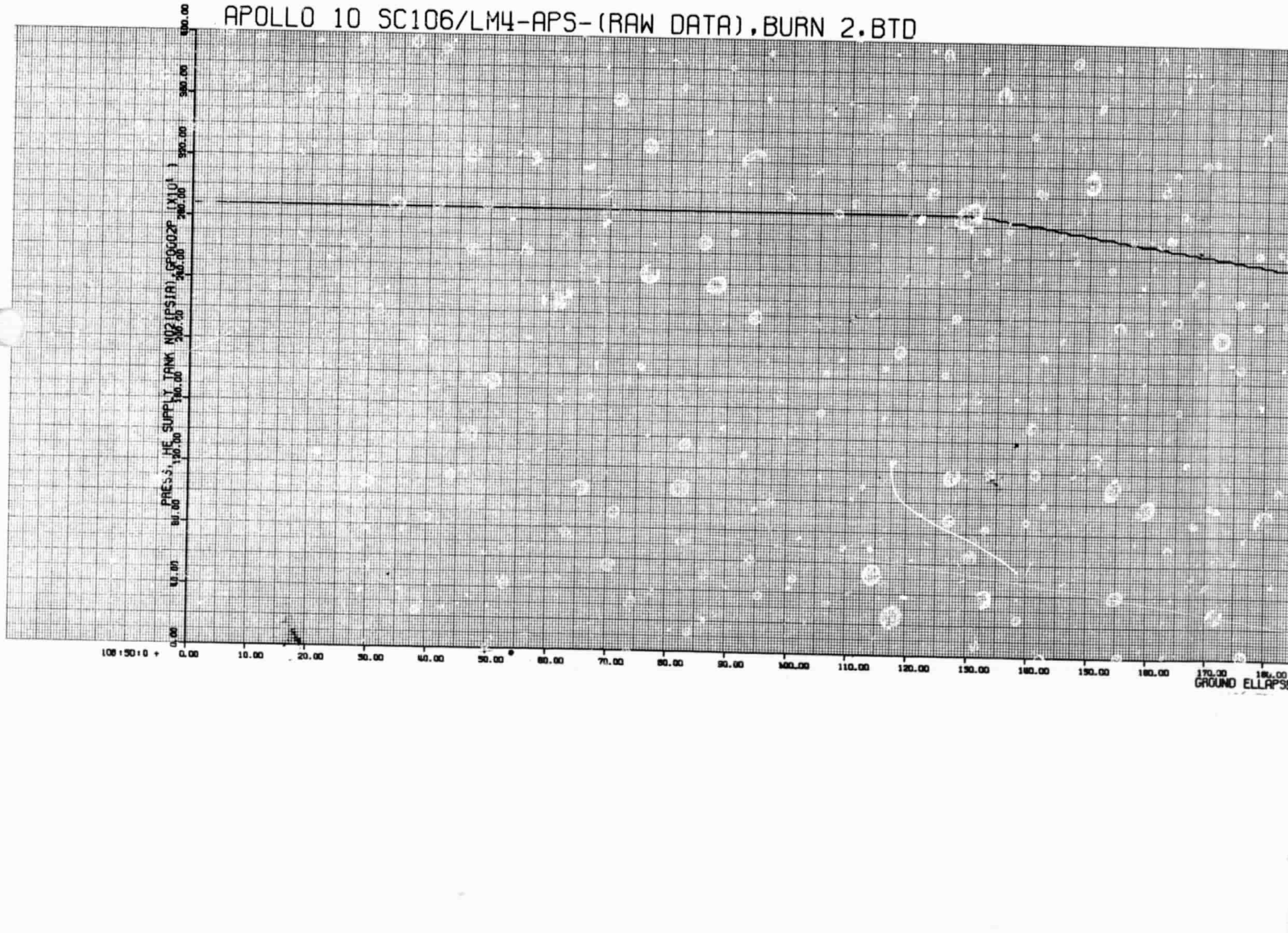


FOLDOUT FRAME /



FOLDOUT FRAME 2

APOLLO 10 SC106/LM4-APS-(RAW DATA),BURN 2.BTD



FOLDOUT FRAME /

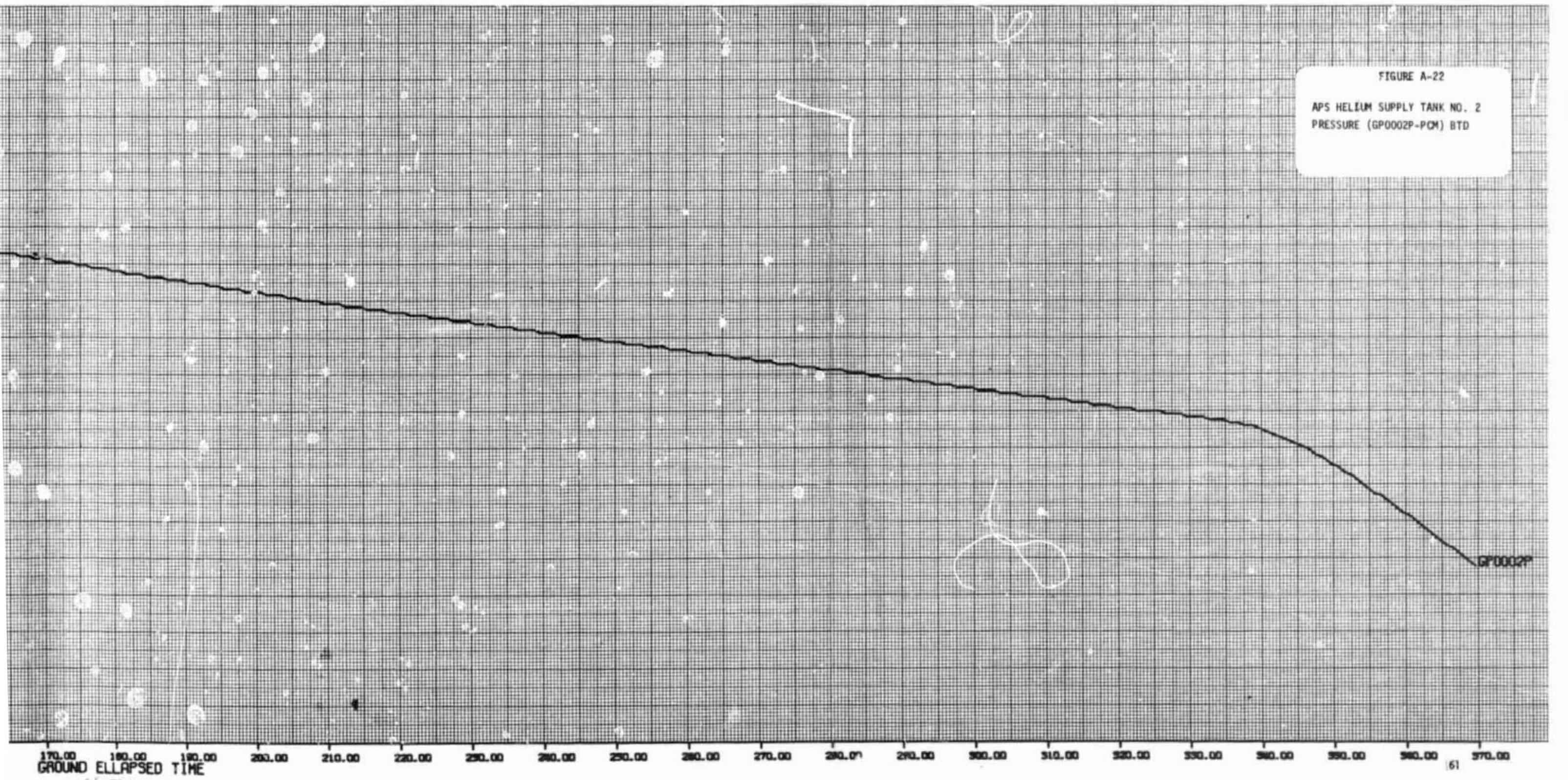


FIGURE A-22

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

FOLDOUT FRAME 2