MSC-01855
SUPPLEMENT 5

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

APOLLO 12 MISSION REPORT

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DESCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
SEPTEMBER 1972

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DESCENT PROPULSION SYSTEM

FINAL FLIGHT EVALUATION

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
September 1972

11176-H585-RO-00

PROJECT TECHNICAL REPORT

APOLLO 12

LM-6

DESCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

NAS 9-8166

July 1970

Prepared for
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
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CONTENTS

TABLES

ILLUSTRATIONS

Figure | Page

ILLUSTRATIONS (Continued)

1. PURPOSE AND SCOPE

The purpose of this report is to present the results of the postflight analysis of the Descent Propulsion System (DPS) performance during the Apollo 12 Mission. The primary objective of the analysis was to determine the steady-state performance of the DPS during the descent phase of the manned lunar landing.

This report is a supplement to the Apollo 12 Mission Report. In addition to further analysis of the DPS, this report brings together information from other reports and memorandums analyzing specific anomalies and performance in order to present a comprehensive description of the DPS operation during the Apollo 12 Mission.

The following items are the major additions and changes to the results are reported in Reference 1:

(1) The performance values for the second DPS burn are presented.

(2) The analysis techniques, problems and assumptions are discussed.

(3) The analysis results are compared to the preflight performance prediction.

(4) The Propellant Quantity Gaging System performance is discussed in greater detail.

(5) Engine transient performance and throttle response is discussed.

(6) Estimated propellant consumption and residuals are revised.

2. SUMMARY

The performance of the LM-6 Descent Propulsion System during the Apollo 12 Mission was evaluated and found to be satisfactory. The average engine effective specific impulse for the Descent Burn was less than predicted, but well within the prediction 3σ uncertainty. The engine performance corrected to standard inlet conditions for the FTP portion of the Descent Burn was as follows: thrust, 9734 pounds; specific impulse, 304.5 seconds; and propellant mixture ratio, 1.598. These values are -0.15, 0.16, and 0.0 percent different, respectively, from the values reported from engine acceptance tests and were within specification limits.

Several flight measurement discrepancies existed during the flight. 1) Comparison of the inflight chamber pressure to ground test and post-flight computed values indicated that the flight transducer may have incurred a drift error, probably due to thermal effects during the burn. This problem also occurred during the Apollo 9 and Apollo 11 Missions. 2) The oxidizer interface pressure measurement appeared low throughout the flight. This discrepancy was assumed to be a measurement bias. Also, pressure oscillations as high as 59 psi peak-to-peak were recorded by the oxidizer interface pressure transducer. It was concluded that the oscillations were instrumentation phenomenon and not indicative of actual flow conditions. 3) The Propellant Quantity Gaging System did not perform within its specifications. The deviations occurred generally at the beginning and middle portions of the burn. Tests at the White Sands Test Facility had indicated earlier that the specifications could not be met. The accuracy was within the expected range based on those tests. 4) The propellant low level sensor was triggered early. The apparent cause was propellant sloshing, ranging in amplitude from 1.5 to 2.0 percent (peak-to-peak) of the gaged propellant levels. Due to the early signal, the capability to perform at least 19 seconds of hover before propellant depletion was not known to be available.

In-flight bulk propellant temperatures were lower than predicted. This also occurred during the Apollo 11 Mission. It is recommended that the temperatures used in future predictions be adjusted to more closely agree with those measured during these two flights.

In-flight supercritical helium tank pressure rise following the DOI Burn was higher than predicted. This also occurred during the Apollo 11 Mission. It is recommended that the pressure rise rate used in future predictions be increased to more closely agree with Apollo 11 and Apollo 12 flight data.

3. INTRODUCTION

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

At 109:23:40 hours, the first DPS maneuver, the Descent Orbit Insertion Burn (DOI) was accomplished. The burn duration was approximately 29 seconds and included operation at the minimum throttle setting and throttling to 40% of full thrust level. At 110:20:38 hours the Descent Burn was initiated and lasted for 717 seconds. The burn was started at the minimum throttling setting and after approximately 26 seconds, the thrust was increased to the fixed throttle position (FTP). At 383 seconds after ignition, the engine was throttled to 59%. Approximately 227 seconds after throttle down, the Spacecraft Commander assumed partial control of the descent. In this semi-automatic mode, the astronaut selected the rate of spacecraft descent but the LM Guidance Computer (LGC) controlled the engine to obtain the desired descent rate. As the spacecraft attitude was changed by the astronaut, a different thrust level was required to maintain the constant rate of descent. The engine was thus commanded through a substantial number of throttling changes between 19 and 44 percent thrust. Lunar landing occurred at 110:32:36 ending the DPS mission duty cycle. The propellant tanks were vented at about 110 hr 33 min GET.

The actual ignition and shutdown times for the two DPS firings are shown in Table 1. The throttling profile for the second DPS burn is shown if Figure 1.

The Apollo 12 Mission utilized LM-6 which was equipped with DPS engine S/N 1040. The engine and feed system characteristics are presented in Table 2.

Each DPS burn was preceded by a two jet +X Reaction Control System (RCS) ullage maneuver to settle propellants.

4. FTP STEADY-STATE PERFORMANCE

Analysis Technique

The major analysis effort for this report was concentrated on determining the flight steady-state performance of the DPS during the fixed throttle position (FTP) portion of the Descent Burn. Analysis of the DOI Burn was not practical due to the low data sampling rate and the duration of burn. The throttled portions of the Descent Burn were of insufficient duration at a given throttle position to allow for a meaningful detailed performance analysis. The performance analysis was accomplished by use of the Apollo Propulsion Analysis Program which utilizes a minimum variance technique to "best" correlate the available flight and ground test data. The program embodies error models for the various flight and ground test data that are used as inputs, and by iterative methods arrives at estimations of the system performance history and propellant weights which "best" (minimum-variance sense) reconcile the data.

Since a minimum variance estimate of performance during the throttled portion (i.e. other than FTP) of the Descent Burn could not be made, a simulation of the latter portion of the burn was performed. The end point conditions of the FTP analysis were used as initial values for this simulation.

Analysis Results

The engine performance during the FTP portion of the Descent Burn was satisfactory.

The Apollo Propulsion Analysis Program (PAP) results presented in this report are based on reconstructions using data from the flight measurements listed in Table 3.

The propellant densities were calculated from sample specific gravity data from KSC, assumed interface temperatures based on the flight bulk propellant temperatures, and the flight interface pressures.

The preliminary estimated spacecraft damp weight (LM minus DPS propellants) at ignition of the DOI maneuver was obtained from the Apollo Spacecraft Program Office. This weight was checked against the LM-6 Weight and Balance History Log from KSC and the itemized weight data from Grumman Aerospace Corporation. Weight discrepancies of -24.4 lbm for the descent stage and -22.3 lbm for the ascent stage were identified. The discrepancies were due to mathematical errors in summing of the component weights. The initial estimates of the DPS propellant on board at the beginning of the analyzed time segment were calculated (based on a simulation of the DOI Burn and the first portion of the Descent Burn) from the loaded propellant weights. The damp weight was adjusted for consumables such as RCS propellant, water, etc., used between DOI ignition and the beginning of the analyzed time segment. During the Descent Burn approximately 153 lbm of consumables other than DPS propellant were used. Of that amount, 118 lbm was RCS propellant. Since there was little RCS activity during the analyzed portion of the burn, it was assumed that the non-DPS consumed weight was used at a rate of 0.09 lbm/sec.

The DPS steady-state FTP performance was determined from the analysis of a 340 second segment of the burn. The segment of the burn analyzed commenced approximately 42 seconds after DPS ignition (FS-1) and included the flight time between 110:21:20 (397280 sec) and 110:27:00 (397620 sec) ground elapsed time (G.E.T.). Engine throttle down to 59% occurred two seconds after the end point of the analyzed segment.

The results of the Propulsion Analysis Program reconstruction of the FTP portion of the Descent Burn are presented in Table 4 along with the preflight values. The values presented are end point conditions of the segment analyzed and are considered representative of the actual flight values throughout the segment. In general, the actual values are within 1.0 percent of the predicted values. A portion of this difference can be attributed to the difference between predicted and actual throat erosion.

Based on the acceleration match during the postflight analysis, it was determined that the actual throat erosion rate was somewhat different than predicted. Subsequent to the publishing of the LM-6/DPS Preflight Performance Report (Reference 3), the throat erosion formulation was revised based on recently completed ground test programs. These programs indicated that the throat erosion as predicted by the former model was conservative. The analysis results tend to substantiate the ground test results. Figure 2 compares the inflight calculated erosion, the predicted erosion using the former model and the predicted erosion using the current model. At the end of the FTP portion of the burn, the inflight calculated erosion was substantially greater than the originally predicted value, but was within the 3σ uncertainty of the predicted value using the current erosion model.

Critique of Analysis Results

Figures 3 through 11 show the analysis program output plots which present the filtered flight data and the accuracy with which the data was matched by the PAP Program. The accuracy is represented by the residual, which is defined as the difference between the filtered data and the program calculated value. The figures presented are thrust acceleration, oxidizer interface pressure, fuel interface pressure, quantity gaging system for oxidizer tank 1 and 2, quantity gaging system for fuel tank 1 and 2, chamber pressure, and regulator outlet pressure.

A strong indication of the validity of the analysis program simulation can be obtained by comparing the thrust acceleration as determined from the LM Guidance Computer (LGC) ΔV data to that computed in the simulation. Figure 3 shows the thrust acceleration derived from the ΔV data and the residual between the measured and computed values. The time history of the residual has an essentially zero mean and a small, but acceptable, negative trend.

Several problems were encountered with flight data while analyzing the steady-state performance at FTP. Several assumptions were necessary in order to obtain an acceptable match to the flight data. These problems are discussed below.

The regulator outlet pressure is redundantly sampled by measurements GQ3018P and GQ3025P. The pressure indicated by GQ3025P was several psia less than that from GQ3018P. Based on early analysis and preflight tests, the data from GQ3018P was used for the analysis. A 1.2 psi step increase in the regulator pressure as measured by GQ3018P was noted about 10 seconds into the analyzed segment. This increase was equivalent to one PCM measurement count. Interface pressure data indicated that the regulator outlet pressure should have increased smoothly during this time from approximately 244 to 245 psia. The data also indicated that the regulator outlet pressure should have continued to slowly increase with time. This trend agreed with the expected regulator characteristics when the supercritical helium supply pressure profile was considered. A smooth pressure profile was therefore substituted for the measured regulator outlet pressure data. Figure 11 presents the adjusted profile that was used as the driver in the PAP analysis.

Comparison of the inflight chamber pressure data to other flight and ground test data indicated that the flight transducer may have incurred a drift error probably due to thermal effects during the burn. A similar problem was noted during the Apollo 9 and Apollo 11 Missions although the shifts were much greater during those flights. Chamber pressure was retained in the analysis although the residual match, Figure 10, was poor.

During the powered descent burn, pressure oscillations as high as 59 psi peak-to-peak were recorded by the oxidizer interface pressure transducer (GQ4111P). The oscillations were recorded at 200 samples per second and show peak-to-peak magnitudes of 25 psi at FTP, 20 psi for most of the 55-57% thrust, a period of high oscillations of 59 psi at approximately 57-60% thrust, and short periods of oscillations of as high as 35 psi during the random throttling period. The fuel interface pressure transducer recorded maximum oscillations of 11 psi. Figure 20 shows the oxidizer pressure oscillations observed. Since oscillations of that magnitude were not observed at the propellant tank pressurization manifold or the engine chamber, it was concluded that the oscillations were not actually present at the interface and that engine performance was not affected. Similar pressure oscillations were noted during the Apollo 11 Mission. A further discussion may be found in Reference 4.

The filtered value of the oxidizer interface pressure (GQ4111P) was less than expected by approximately 1.2 psia. This value is well within the accuracy of the measurement and was assumed to be a measurement bias.

Comparison with Preflight Performance Predictions

Prior to the Apollo 12 Mission the expected inflight performance of the DPS was presented in Reference 3. The preflight performance report was intended to bring together all the information relating to the entire Descent Propulsion System and to present the results of the simulation of its operation in the space environment.

The predicted steady-state and related three sigma dispersions for the specific impulse, mixture ratio and thrust during the FTP portion of the Descent Burn are presented in Figure 12. The corresponding analysis program computed flight performance values are also presented for comparison. With the exception of thrust, all flight values are within 0.3% of the predicted values. The flight values of thrust are within 0.7% of the predicted values. The greater difference in thrust is due to the difference in predicted and actual throat erosion as discussed previously. All values are well within the three sigma limits. The difference between analyzed and predicted mixture ratio was due to propellant density differences (due to actual temperatures and specific gravities).

Engine Performance at Standard Inlet Conditions

The flight performance prediction of the DPS engine was based on the data obtained from the engine acceptance tests. In order to provide a common basis for comparing engine performance, the acceptance test and flight performance is adjusted to standard inlet conditions. This allows actual engine performance variations to be separated from pressurization system and propellant temperature induced variations. The standard inlet conditions performance values were calculated for the following conditions:

Standard Inlet Conditions

Oxidizer interface pressure, psia	221.9
Fuel interface pressure, psia	221.9
Oxidizer interface temperature, ^{o}F	70.0
Fuel interface temperature, ^{o}F	70.0
Oxidizer density, lbm/ft^3	90.13
Fuel density, lbm/ft^3	56.36
Thrust acceleration, lbf/lbm	1.0
Throat area, in^2	54.4

The following table presents ground test data and flight test data adjusted to standard inlet conditions. Comparing the corrected engine flight performance at FTP during the Descent Burn to the corrected ground test

Parameter / Data Source	Ground Test	Flight
	Engine Prediction Characterization	Analysis Results
Thrust, lbf	9749	9734
Specific Impulse, sec	304.1	304.5
Mixture Ratio	1.598	1.598
Thrust Coefficient, C_f	1.780	1.764
Characteristic Velocity, C*, ft/sec	5495	5552

data shows the flight data to be 0.15% less, 0.16% more and 0.0% for thrust, specific impulse and mixture ratio, respectively. These differences are due to the 0.90% decrease in C_f and 1.04% increase in C* derived from the flight analysis. These differences are well within the engine repeatability uncertainties and within the performance specification ranges.

5. SIMULATION OF THROTTLED PERFORMANCE RESULTS

The DPS throttling performance was simulated by utilizing the prediction mode of the Apollo Propulsion Analysis Program. By this method, the measured value of the regulator outlet pressure (GQ3018P) drives the program and the measured value of throttle command voltage (GH1331V) determines the engine throttle setting. The program then calculates values of the remaining flight measurements and engine performance. In this mode the program does not compare calculated measurements with flight measurements and minimum variance analysis is not performed.

Based on the FTP analysis results, it was determined that a correction of +1.0 psia should be made to the regulator outlet pressure (GQ3018P). For the simulation, the initial values of throat erosion and of LM vehicle and propellant weights were obtained from the end point conditions of the FTP analysis. The damp weight was adjusted for non-DPS consumables, as in the FTP analysis, at the rate of 0.39 lbm/sec to account for the remainder of that weight lost during the burn.

The DPS throttling performance simulation was conducted starting at the end of the FTP analysis (FS-1 + 382 seconds) and continued for 334 seconds. This includes all of the powered descent burn after throttle down with the exception of the last second prior to engine shutdown and includes the flight time between 110:27:00 (397620 sec) and 110:32:34 (397954 sec) G.E.T.

Typical values of the simulation results are presented in Table 5. The preflight values as well as the flight measured values are also presented. The time slices presented are FS-1 + 396 seconds (397634 sec) and FS-1 + 600 seconds (397838 sec). The chamber pressure drift error, discussed in the FTP analysis, is also apparent during the throttling analysis. The differences between the predicted and simulated performance values are attributed to the difference between predicted and measured throttle command voltage, regulator outlet pressure, spacecraft weight, propellant density and vapor pressure, and throat erosion. As previously noted in the FTP analysis, the actual throat erosion was substantially greater than the predicted erosion profile from a model which has since been replaced with a more current formulation. The actual erosion is in better agreement with the current model than with the former model and is within the 3σ uncertainty of the current model (Figure 2).

Figures 13 through 16 present plots comparing the preflight predicted and the analysis program simulated values of throttle command voltage, mixture ratio, thrust and specific impulse.

It can be seen in Figure 13 that the throttle command voltage for the simulation (which was measured in flight) generally agreed with the predicted values until the Hi-gate target, which occurred 6 seconds later than predicted. The difference due to this time lag (which was within the 3σ uncertainties) remained essentially constant until the astronaut assumed partial control of the spacecraft guidance. In this mode, the astronaut selected the rate of descent but the LM Guidance Computer (LGC) controlled the engine. This occurred approximately 609 seconds after ignition. The throttle voltage commanded in flight is dependent on the LGC estimate of the state vector of the spacecraft; that is, its position, velocity gain and acceleration with respect to the guidance target. In addition, the command voltage is affected by the actual engine thrust and vehicle weight. Without the aid of a minimum variance analysis, including guidance system models as well as the DPS model, it is impossible to fully determine the affects and interactions of these variables. The simulation did indicate, however, that with the measured voltage and acceleration, an error in simulated thrust was possible. The magnitude of this error was generally less than 90 lbf and was within the predicted 3σ uncertainty.

Deviations in predicted and simulated propellant mixture ratio (Figure 14) were due to differences between predicted and actual propellant densities and vapor pressures. The difference in mixture ratio was within the predicted 3σ band. The predicted propellant densities were based on nominal specification values at 70^{o}F. Propellant samples taken prior to flight determined that the fuel density was greater than nominal and the oxidizer density was less than nominal. The inflight oxidizer and fuel temperatures were approximately 4^{o}F less than predicted. The decreased propellant temperature caused decreased propellant vapor pressures which greatly determine flowrates to the engine and further affect mixture ratio. For both the Apollo 11 and Apollo 12 Missions, the predicted propellant temperature was 70^{o} while the inflight temperatures were actually less. It is recommended that the propellant temperatures used in future predictions be adjusted to more closely agree with those temperatures measured inflight. Figures 17 through 28 present the inflight values of the measured propulsion parameters. The major portion of the FTP data has been deleted to obtain better resolution. In general, the FTP data shown is representative of the deleted segment.

6. OVERALL PERFORMANCE

When the results of the FTP analysis and the simulation of throttled operation are combined, the overall performance during the Descent Burn and the total propellant consumption for the mission can be evaluated. The following table presents a comparison of the propellant consumption, average mixture ratio (MR) and overall effective specific impulse (Isp). While the propellant consumption and mixture ratio are for the entire mission, the effective specific impulse is for the Descent Burn only. The vehicle effective specific impulse was computed based on spacecraft weight reductions

	Propellant Consumption (lbm)		Average MR	Vehicle Effective	Engine Effective
	Oxidizer	Fuel	(O/F)	Isp (sec)	Isp (sec)
Preflight Prediction	10547	6620	1.593	300.3	302.4
Analysis Program	10596	6630	1.598	297.5	301.4
Gaging System*	10568	6644	1.591	297.8	301.7

*Based on gaging system and ΔV measurements

due to both DPS propellant consumption and non-DPS consumables (approximately 0.09 lbm/sec during FTP and 0.39 lbm/sec during throttled operation). The engine effective specific impulse was calculated considering only weight reductions due to DPS propellant usage. Contributions from RCS activity is not included.

The measured propellant quantities consumed are based on final gage readings and measured initial loads. Due to loading and gaging system inaccuracies, the uncertainties in the consumed propellants are ±85 lbm and ±53 lbm (3σ) for oxidizer and fuel, respectively. The uncertainties in mixture ratio and engine effective specific impulse resulting from these uncertainties are ±0.016 and ±1.59 respectively. Both the predicted and analysis program results are within these uncertainties.

1 Calculated from FS-1 plus 33 seconds.

The values of effective specific impulse presented in the table are dependent on both the vehicle weight change and the thrust velocity gain. The analysis indicated a thrust velocity gain of 6805 ft/sec. The velocity gain used in computing the values of Isp using the gaging system readings were taken directly from the acceleration data (GG0001X) which measures the gain in velocity for each two second segment of the burn. The total measured thrust velocity gain, 6804 ft/sec, includes the contribution of both the DPS engine and RCS activity. The uncertainty in effective specific impulse due to measured propellant usage and velocity gain uncertainties is ±1.2 seconds. The engine effective specific impulse for both the prediction and analysis are within this uncertainty. The difference between the predicted and actual throat erosion accounts for the higher predicted engine effective specific impulse. Due to the rather large uncertainties related to the gaging system, it is felt that the best estimate of DPS performance is given by the analysis results.

Both the analysis results and measured results are within the prediction 3 uncertainties of ±5.91 sec and ±0.0225 for effective specific impulse and mixture ratio, respectively. The difference between the predicted vehicle effective specific impulse and that calculated from the PQGS measurements and the analysis program was due to more RCS usage (non-DPS consumable) than predicted.

7. PQGS EVALUATION AND PROPELLANT LOADING

Propellant Quantity Gaging System

During the DOI Burn and at ignition of the Descent Burn, all propellant gages were reading off scale as expected. The fuel tank 2 (Fu2) gage did, however, intermittently drop slightly below the maximum reading of 95%, particularly during the early portion of the Descent Burn. During both the Apollo 10 and Apollo 11 Missions (References 5 and 4, respectively) the fuel gages did not read off scale as expected when propellant was above the maximum gageable level. The Fu2 gage was reading off scale within 27 seconds after ignition of the Descent Burn.

All gages were indicating propellant consumption by 43.5 sec after Descent Burn ignition. At that time, the gages were reading 94.4, 95.0, 93.6, and 92.1 percent for Ox1, Ox2, Fu1 and Fu2, respectively. The reconstruction analysis indicates that the Ox2 gage was reading high by nearly two percent, the Ox1 gage was high by approximately one percent and the Fu2 gage was low by approximately one percent. As the burn continued the oxidizer gage readings diverged until approximately 400 seconds after ignition. At that time the readings of the Ox1 gage were approximately 1.3 percent greater than the Ox2 reading. The difference in propellant levels was due to propellant transfer between the oxidizer tanks caused by gimbaling of the engine commanding the thrust vector to pass through the vehicle center of gravity. As the burn continued, the engine gimbal angles decreased thus allowing some propellant to be transferred back from the Ox1 tank to the Ox2 tank. At 43.5 seconds after ignition the fuel gages displayed a difference of approximately 1.4 percent (Fu1 high). This difference remained essentially constant during the first 400 seconds of the burn. Subsequently, the difference decreased to approximately one half percent. The initial difference in propellant level may have been caused by preburn maneuvering. The engine gimbal angle which was relatively constant during the first 400 seconds could have maintained this difference. At approximately 400 seconds, the gimbal angle was decre[illegible] accounting for the convergence of the fuel gage readings. At the end of [illegible] burn, the propellant gages were reading approximately 6.4, 6.1, 5.6 and 5.4 percent for Ox1, Ox2, Fu1, and Fu2, respectively.

The expected accuracies for the gaging system, based on tests conducted at WSTF (Reference 6) are presented in the following table:

EXPECTED PROPELLANT GAGING SYSTEM ACCURACY			
Quantity Remaining in Tank	Accuracy for Each Oxidizer Gage	Quantity Remaining in Tank	Accuracy for Each Fuel Gage (% of Full Tank)
100-50%	2.7%	100-60%	3.5%
50-25%	1%	60-20%	2%
25-8%	0.5%	20-0%	1%
8-0%	1%		

The specification limit of the PQGS is ±1% of full tank capacity for quantities above 25% load and below 8% load. When the PQGS is integrated into the vehicle and telemetry effects are considered, the ±1% value is increased to ±1.3%. In the 8% to 25% range, the specification requirement is ±0.5% of full tank capacity. However, the WSTF tests indicate that these specifications cannot be met.

In the analysis of FTP and the simulation of the throttled portion of the burn, the transfer of approximately 81 lbm of oxidizer from tank No. 1

t... tank No. 2 was modeled. No fuel transfer was simulated. Table 6 presents a comparison of the measured data and the best estimate of the actual values at various time points during the Descent Burn. While the difference between the measured and computed values were frequently outside the specification limits they were generally within the expected accuracy of the gaging ... based on WSTF results.

At engine shutdown, the quantities of propellants remaining in the tanks were computed to be 6.5, 5.5, 5.7 and 5.7 percent for Ox1, Ox2, Fu1 and Fu2, respectively. This is equivalent to remaining quantities of 761 lbm of oxidizer and 446 lbm of fuel. Of these quantities, 657 lbm of oxidizer and 408 lbm of fuel are usable before a propellant tank depletes. It should be noted that values of usable propellants are contingent on the actual amount of oxidizer transferred between tanks. Based on the measured propellant quantities, usable quantities of 627 lbm of oxidizer and 389 lbm of fuel remained. Applying the propellant flowrates at engine shutdown, 111.6 seconds of hover time remained based on computed residual propellants. The measured quantities indicated 106.6 seconds of remaining hover time.

The propellant low level was triggered at 110:31:59.6 GET, 682 seconds after ignition and 35 seconds before engine shutdown. Gaging system data indicates that the sensor was triggered in the Fu2 tank. At the time of the signal, the mean measured readings were 8.0, 7.3, 7.4 and 6.7 percent for the Ox1, Ox2, Fu1, and Fu2 gages, respectively. Based on the predicted time of the low level signal, the sensor was triggered early. This may have been due to the 1.5 to 2.0 percent oscillation (peak-to-peak) of the propellant levels as indicated by the gaging probes. It is likely that the propellant level oscillations were due to slosh. Observing the mean of the measured readings, the low level signal should have been activated 701 seconds after ignition by the sensor in the fuel tank No. 2. Thus, the sensor was triggered approximately 19 seconds early. Approximately 113 seconds of burn time remain when the low level indication is given. Due to the early signal, a remaining burn time of 78 seconds was indicated at engine shutdown. The availability of 19 seconds of hover time is significant. A similar early signal occurred during the Apollo 11 Mission. It is believed that propellant slosh is the cause of the premature low level indications. Due to the low gaging probe measurement sample rate (1 sample/second), it is difficult to substantiate this hypothesis. Ground tests are being conducted to determine the effect of propellant slosh. It should be noted that preliminary ground testing to determine the effects of propellant slosh on the gaging system readings indicate that the mean propellant level as measured by the probe can be lower than the quiescent level of propellants. Thus, the low level sensor signal may have been more than 19 seconds early. It is planned to increase the PQGS sample rate to 100 samples/second for the Apollo 13 Mission in an effort to identify the slosh modes occurring in flight.

Propellant Loading

Prior to propellant loading, density determinations were made for each propellant to establish the amount of off-loading of the planned overfill. An average oxidizer density of 90.25 lbm/ft^3 and an average fuel density of 56.49 lbm/ft^3 at a pressure of 240 psia and a temperature of 70°F were determined from the samples. In off-loading propellant to obtain the desired ullage volume, more propellant than planned (5.2 lbm of oxidizer and 3.2 lbm of fuel) was removed from the spacecraft. The quantities loaded were 11345.7 lbm of oxidizer at a temperature of 67.7°F and a pressure of 53.6 psig and 7080.4 lbm of fuel at a temperature of 70.0°F and a pressure of 41.8 psig (Reference 7). The total quantity of DPS propellant on board at launch was 18426.1 lbm.

8. PRESSURIZATION SYSTEM EVALUATION

The pressurization system performed satisfactorily throughout the mission.

The ambient start bottle was loaded with approximately 1.1 lbm of helium at a pressure of 1632 psia and temperature of 70°F. At launch, the pressure was approximately 1625 psia. Five days prior to launch, the oxidizer and fuel tank pressures were increased from their pressures at loading to 140.2 and 134.9 psia, respectively. At launch the propellant tank pressures had decreased to approximately 94 and 123 psia, respectively. Approximately 33.6 hours prior to launch, the supercritical helium (SHe) tank was filled with 48 lbm of supercritical helium at a bottle pressure of approximately 110.5 psia. At launch the bottle pressure had risen to approximately 364 psia.

The following table presents measured pressurization system values for various times during the mission.

	Initial Pressurization (psia)	Launch (psia)	Pressure at DPS Activation (90:23 hrs G.E.T.) (psia)	Post Start Tank Pressurization (~107:48 G.E.T.) (psia)
Start Bottle	1632	1625	1605	491
Oxidizer Tank	140.2	94	60	247
Fuel Tank	134.9	123	105	248
SHe Tank	110.5	364	917	1036

The large amount of pressure decay in the propellant tanks was due primarily to helium absorption by the propellants and was as expected. The decay in ambient start bottle pressure prior to the DOI Burn was attributed to helium temperature drop and was within the expected range.

The best estimate of the ground SHe tank pressure rise rate was approximately 8.0 psi/hr and was caused by normal heat leak into the system from the environment. SHe tank pressures after liftoff indicated that the coast pressure rise rate was approximately 6.2 psi/hr. Both values compare favorably with previous flights. A peak SHe tank pressure of 1305 psia was measured during the Descent Burn. A comparison of preflight predicted and flight measured SHe tank pressure is shown in Figure 29. The predicted pressure was obtained utilizing the DPS Supercritical Helium Pressurization System Computer Program. The difference between preflight predicted and flight measured pressure at the end of the Descent Burn was approximately 85 psi, the predicted value being greater. A postflight simulation, using flight data as input to the computer program, resulted in the predicted value being approximately 120 psi greater than the flight measured value. The reason for the larger computed values has not been resolved; however, it is hypothesized that the difference is due to incorrect SHe thermal properties in the 500 psia, 150°R region. An analysis is now being made of the differences between program thermal properties and new National Bureau of Standards data.

Flight data from LM-5 and LM-6 indicates that the pressure rise rate which occurs during the coast period following the DOI Burn is greater than previously predicted. SHe tank pressure rise was 73 psi for LM-5 and 77 psi for LM-6. The predicted value was 50 psi. Future SHe system predictions will use an average of the measured values (75 psi).

DPS storage tank venting was initiated approximately 35 seconds after Descent Burn engine shutdown. This was successfully accomplished by closing the SHe isolation valve, firing the vent line squib valves and opening the vent line solenoid valves (Figure 30). The SHe tank was vented approximately 20 minutes prior to APS ignition by opening the SHe isolation valve and the vent line solenoid valves. During the period of lunar stay, the SHe tank had an average pressure rise rate of 4.9 psi/hr. Interface pressure rises, which occurred on LM-5 during venting (Reference 4), were prevented by the sequential venting procedure. This was due to venting the SHe tank after engine heat soakback had occurred. Pressure buildup occurs because venting of the SHe tank causes the stagnant fuel in the helium/fuel heat exchanger to freeze. Engine heat soakback then expands the fuel against the plugged heat exchanger causing a pressure buildup in the feedlines downstream which could exceed specification limits.

9. ENGINE TRANSIENT ANALYSIS

The mission duty cycle of the Descent Propulsion system for Apollo 12 included two starts at the minimum throttle setting, one shutdown at approximately 38% throttle and one shutdown at 23.4 throttle. Much throttling occurred during the Descent Burn, all of which was commanded by the LGC.

Start and Shutdown Transients

Due to the low data sample rate (1 s/s) during the DOI Burn, no analysis of the transients was made. Reference 8 presents the technique used in determining the time of engine fire switch signals (FS-1 and FS-2) for the Descent Burn. This method was developed from White Sands Test Facility (WSTF) test data and assumes that approximately 0.030 seconds after the engine start command (FS-1) an oscillation in the fuel interface pressure occurs, as observed from the WSTF tests. Similarly, 0.092 seconds after the engine shutdown signal (FS-2) another oscillation in the fuel interface pressure occurs. Thus, start and shutdown oscillations of the fuel interface pressure were noted and the appropriate time lead applied.

The ignition delay from FS-1 to first rise in chamber pressure was approximately 0.66 seconds (second burn). From past flights it has been shown that the first start delay of a duty cycle is generally twice as long as subsequent start delays. This difference in time between the first and subsequent start delays appears to be because of a difference in priming conditions. Prior to the DOI Burn there was no propellant between the engine prevalves (actuator isolation valves) and the engine valve actuators, and no propellant between the series engine shutoff valves. For the Descent Burn these volumes were primed with propellants. The delay time compared favorably to those experienced on the flights of Apollo 5, Apollo 9 and

Apollo 10. On Apollo 11, the response of the chamber pressure transducer was such that it gave no output until it sensed a pressure in excess of 4.4 psi. Thus, the time of initial chamber pressure rise and the start delay time could not be determined.

It was determined that there was a bias of 0.8 psia on the chamber pressure measurement at ignition. It was assumed that the bias decreased linearly from ignition (0.0 psia chamber pressure) to a chamber pressure of 106 psia. Thus at 90% (15.3 psia) and 100% (17 psia) of the minimum throttle setting the bias was 0.68 psia and 0.66 psia, respectively.

The start transient from FS-1 to 90% of the minimum steady-state throttle setting (16.2% of full thrust) required 1.77 seconds with a start impulse of 499 lbf-sec. The transient time was well within the specification limit of 4.0 seconds for a minimum throttle start. The start transient from 90% to 100% of the minimum throttle setting required 0.14 seconds with an impulse of 207 lbf-sec.

The shutdown transient required 2.06 seconds from FS-2 to 10% of the steady-state throttle setting (23.4%) with an impulse of 1540 lbf-sec. The specification limit on transient shutdown time is 0.25 seconds, however, this applies only to shutdowns from FTP. There is no specification limit on impulse. The impulse from 10% of the steady-state throttle setting to 0% thrust was 161 lbf-sec over a time increment of 0.79 seconds. Table 7 presents a summary of the transients for flights to date.

Throttle Response

During the Descent Burn the engine was commanded to many different thrust levels. All throttle commands were automatic. The first throttling maneuver, minimum (16% of full thrust) to FTP, which was executed 26 seconds into the burn, required approximately 1 second. The engine then remained at FTP for 357 seconds. The second command, from FTP to 59%, occurred 383 seconds after ignition and required approximately 0.5 seconds. This value compares favorably with similar maneuvers on previous flights. Little throttling was performed during the next 136 seconds. The LM Guidance Computer then commanded a ramping decrease in the throttle setting from 54% to 33% over 90 seconds. At this time the Spacecraft Commander selected guidance program P-66 which allowed him to select the vehicle rate of descent with the LGC still controlling the Descent Engine. During the subsequent 108 seconds of the burn, the LGC commanded approximately 99 throttle changes in the 19% to 44% range. The command time from one throttle setting to the next was generally less than 0.20 seconds and the engine response time was less than 0.15 seconds in addition to the command times. The requirement for the large number of throttle changes was directly attributed to the spacecraft attitude. As the astronaut pitched or rolled the vehicle, a different engine throttle setting was necessary to maintain the selected rate of descent. While no throttle response specifications exist for commands of the type given during the latter portion of the burn, the response of the DPS engine was considered satisfactory.

10. REFERENCES

1. NASA Report MSC-01855, "Apollo 12 Mission Report," March 1970.

2. SPD9-R-051, "Mission Requirements, SA-507/CSM-108/LM-6, H-1 Type Mission, Lunar Landing," 18 July 1969.

3. TRW Report No. 11176-H392-R0-00, "Apollo Mission H1/LM-6/DPS Preflight Performance Report," R. K. M. Seto and A. T. Avvenire, October 1969.

4. TRW Report No. 11176-H496-R0-00, "Apollo 11, LM-5 Descent Propulsion System Final Flight Evaluation," R. K. M. Seto and R. L. Barrows, March 1970.

5. TRW Report No. 11176-H314-R0-00, "Apollo 10, LM-4 Descent Propulsion System Final Flight Evaluation," R. K. M. Seto, 8 August 1969.

6. GAC LED-271-98, "PQGS Accuracy Study," I. Glick and S. Newman, 14 May 1969.

7. SNA-8-D-027(III)Rev. 2., "CSM/LM Spacecraft Operational Data Book," Volume III, Mass Properties, 20 August 1969.

8. MSC Memorandum EP22-41-69, "Transient Analysis of Apollo 9 LMDE," from EP2/Systems Analysis Section to EP2/Chief, Primary Propulsion Branch, 5 May 1969.

TABLE 1

DPS MISSION DUTY CYCLE

Burn	FS-1 (hr:min:sec)	FS-2 (hr:min:sec)	Burn Duration (sec)	Velocity Change (ft/sec)
DPS-1 (1)	109:23:39.9	109:24:08.9	29.0	72.4
DPS-2	110:20:38.11 (3)	110:32:35.07 (3)	717.0	6804 (2)

(1) Reference 1

(2) LM Guidance Computer Downlink Data - GG0001X

(3) Determined from DPS Fuel Interface Pressure, GQ3611P

TABLE 2

LM-6 DESCENT PROPULSION ENGINE AND

FEED SYSTEM PHYSICAL CHARACTERISTICS

ENGINE	
Engine Number	1040
Chamber Throat Area, in^2	54.197[1]
Nozzle Exit Area, in^2	2569.7[4]
Nozzle Expansion Ratio	47.4[4]
Oxidizer Interface to Chamber Resistance at FTP $\frac{lbf\text{-}sec^2}{lbm\text{-}ft^5}$	3960.0[3]
Fuel Interface to Chamber Resistance at FTP $\frac{lbf\text{-}sec^2}{lbm\text{-}ft^5}$	6325.2
Fuel Film Coolant Tapoff Point to Combustion Chamber $\frac{lbf\text{-}sec^2}{lbm\text{-}ft^5}$	465069
FEED SYSTEM	
Oxidizer Propellant Tanks, Total Ambient Volume, Ft^3	126.0[4]
Fuel Propellant Tanks, Total Ambient Volume, Ft^3	126.0[4]
Oxidizer Tank to Interface Resistance, $\frac{lbf\text{-}sec^2}{lbm\text{-}ft^5}$	427.03[2]
Fuel Tank to Interface Resistance, $\frac{lbf\text{-}sec^2}{lbm\text{-}ft^5}$	674.53[2]

TABLE 2 (CONTINUED)

[1]TRW No. 01827-6173-R-00, TRW LEM Descent Engine Serial No. 1040 Acceptance Test Performance Report Paragraph 6.9, dated 19 July 1968.

[2]GAEC Cold Flow Tests.

[3]TRW No. 4721.3.69-63, LM-6, Engine Serial No. 1040 Descent Engine Characteristic Equations, March 1969.

[4]Approximate Values.

TABLE 3

FLIGHT DATA USED IN FTP STEADY-STATE ANALYSIS

Measurement Number	Description	Range	Sample Rate Sample/Sec
GQ3018P	Pressure, Helium Reg. Out. Manifold	0-300 psia	1
GQ3611P	Pressure, Engine Fuel Interface	0-300 psia	200
GQ4111P	Pressure, Engine Oxidizer Interface	0-300 psia	200
GQ6510P	Pressure, Thrust Chamber	0-200 psia	200
GQ3603Q	Quantity, Fuel Tank No. 1	0-95 percent	1
GQ3604Q	Quantity, Fuel Tank No. 2	0-95 percent	1
GQ4103Q	Quantity, Oxidizer Tank No. 1	0-95 percent	1
GQ4104Q	Quantity, Oxidizer Tank No. 2	0-95 percent	1
GQ3718T	Temperature, Fuel Bulk Tank No. 1	20-120^{o}F	1
GQ3719T	Temperature, Fuel Bulk Tank No. 2	20-120^{o}F	1
GQ4218T	Temperature, Oxidizer Bulk Tank No. 1	20-120^{o}F	1
GQ421[illegible]	Temperature, Oxidizer Bulk Tank No. 2	20-120^{o}F	1
GG0001[illegible]	PGNS Downlink Data	40 Bits	1/2

TABLE 4

DESCENT PROPULSION SYSTEM STEADY STATE FTP PERFORMANCE

PARAMETER	FS-1 + 42 SECONDS			FS-1 + 382 SECONDS		
INSTRUMENTED	Predicted	Measured	Calculated	Predicted	Measured	Calculated
Regulator Outlet Pressure, psia	245	244	244	245	244	245
Oxidizer Interface Pressure, psia	225	222	224	224	222	224
Fuel Interface Pressure, psia	225	223	224	224	224	224
Engine Chamber Pressure, psia	104	105	105	99	100	97.3
Oxidizer Bulk Temperature, Tank No. 1, °F	70	65.5	---	70	65.5	---
Oxidizer Bulk Temperature, Tank No. 2, °F	70	65.8	---	70	65.8	---
Fuel Bulk Temperature, Tank No. 1, °F	70	66.2	---	70	66.2	---
Fuel Bulk Temperature, Tank No. 2, °F	70	66.5	---	70	66.5	---
DERIVED						
Oxidizer Flowrate, lbm/sec	19.9	---	19.8	20.2	---	20.4
Fuel Flowrate, lbm/sec	12.5	---	12.4	12.7	---	12.8
Propellant Mixture Ratio	1.594	---	1.598	1.593	---	1.596
Vacuum Specific Impulse, sec	304.6	---	304.8	302.2	---	301.4
Vacuum Thrust, lbf	9356	---	9790	9953	---	9987
Throat Erosion,	0.27	---	0.0	7.65	---	11.61

TABLE 5

DESCENT PROPULSION SYSTEM THROTTLED PERFORMANCE

PARAMETER	FS-1 + 396 Seconds			FS-1 + 600 Seconds		
INSTRUMENTED	Predicted	Measured	Simulation	Predicted	Measured	Simulation
Regulator Outlet Pressure, psia	246	245	246	247	245	246
Oxidizer Interface Pressure, psia	237	234	237	241	239	241
Fuel Interface Pressure, psia	237	235	237	241	240	241
Engine Chamber Pressure, psia	59	[illegible]	56	33	38	35
Oxidizer Bulk Temperature, Tank No. 1, °F	70	65.5	65.5	70	65.5	65.5
Oxidizer Bulk Temperature, Tank No. 2, °F	70	65.8	65.8	70	65.3	65.8
Fuel Bulk Temperature, Tank No. 1, °F	70	66.2	66.2	70	66.2	66.2
Fuel Bulk Temperature, Tank No. 2, °F	70	66.5	66.5	70	66.5	66.5
Throttle Command Voltage, VDC	8.23	8.22	8.22	5.11	5.32	5.32
DERIVED						
Oxidizer Flowrate, lbm/sec	12.0	----	11.8	7.1	----	7.4
Fuel Flowrate, lbm/sec	7.6	----	7.4	4.4	----	4.6
Propellant Mixture Ratio, O/F	1.587	----	1.594	1.600	----	1.603
Vacuum Specific Impulse, sec	302.3	----	300.4	296.2	----	295.[illegible]
Vacuum Thrust, lbm	5924	----	5772	3401	----	3579
Throat Erosion,	7.82	----	11.75	10.19	----	[illegible]

TABLE 6

DPS PROPELLANT QUANTITY GAGING SYSTEM PERFORMANCE

Parameter	Time (from Descent Burn ignition) sec							
	42	142	242	342	442	542	642	716
Oxidizer Tank No. 1								
Measured Quantity, percent	94.4	77.0	59.8	41.6	28.3	17.5	10.1	6.4
Calculated Quantity, percent	93.4	76.5	59.4	42.1	28.6	17.9	10.6	6.5
Difference, percent	+1.0	+0.5	+0.4	-0.5	-0.3	-0.4	-0.6	-0.1
Oxidizer Tank No. 2								
Measured Quantity, percent	95.0	76.6	58.9	40.0	27.0	16.6	9.3	6.1
Calculated Quantity, percent	93.1	75.4	57.5	39.5	26.1	16.0	9.3	5.5
Difference, percent	+1.9	+1.2	+1.4	+0.5	+0.9	+0.6	0.0	+0.6
Fuel Tank No. 1								
Measured Quantity, percent	93.6	75.7	58.4	39.6	26.7	16.3	9.2	5.6
Calculated Quantity, percent	93.2	75.8	58.3	40.6	27.1	16.6	9.6	5.7
Difference, percent	+0.4	-0.1	+0.1	-1.0	-0.4	-0.3	-0.4	-0.1
Fuel Tank No. 2								
Measured Quantity, percent	92.2	74.2	56.9	38.7	26.0	15.7	8.7	5.4
Calculated Quantity, percent	93.2	75.8	58.3	40.6	27.1	16.6	9.6	5.7
Difference, percent	-1.0	-1.6	-1.4	-1.9	-1.1	-0.9	-0.9	-0.3

TABLE 7

DPS START AND SHUTDOWN IMPULSE SUMMARY

	Apollo 12 LM-6/DPS-2	Apollo 10 LM-4/DPS-2	Apollo 9 LM-3/DPS-1	Apollo 9 LM-3/DPS-2	Apollo 9 LM-3/DPS-3	Apollo 5 LM-1/DPS-2	Apollo 5 LM-1/DPS-3	SPECIFICATION LIMITS
	STARTS							
Steady-State Throttle Position, Percent	16.2	13.1	12.7	12.7	12.7	12.4	12.4	
Total Vacuum Start Impulse (FS-1 to 90% steady state), lbf-sec	591	728	805	1029	950	894	574	
Start Time (FS-1 to 90% steady state), sec	1.77	2.13	2.5[1]	2.1	2.3[1]	2.66	2.13	4.0
Coast Time From Prior Burn, Minutes	56	72	From Launch	2640	111	131	0.5	
	SHUTDOWNS							
Steady-State Throttle Position, Percent	23.4	FTP	40	40	12.7	FTP	FTP	
Total Vacuum Shutdown Impulse (FS-2 to 10% Steady State), lbf-sec	1540	2041	---[2]	1730	748	1727	1713	
Shutdown Time (FS-2 to 10% steady-state), sec	2.06	0.34	1.1[1]	1.1	1.9[1]	0.26	0.50	0.25[4]
Repeatability, lbf-sec						1734 ± 7	1734 ± 7	±100[4]
Total Vacuum Shutdown Impulse (FS-2 to Zero Thrust) from Velocity Gain Data, lbf-sec	---[5]	2948	1777	---[2]	1040	2493	---[3]	

[1] Reference 8.

[2] Data Unavailable.

[3] Unavailable due to APS "Fire-in-the-Hole" maneuver

[4] Specification value for shutdowns performed from FTP [illegible]

[5] Not applicable to [illegible] shutdown

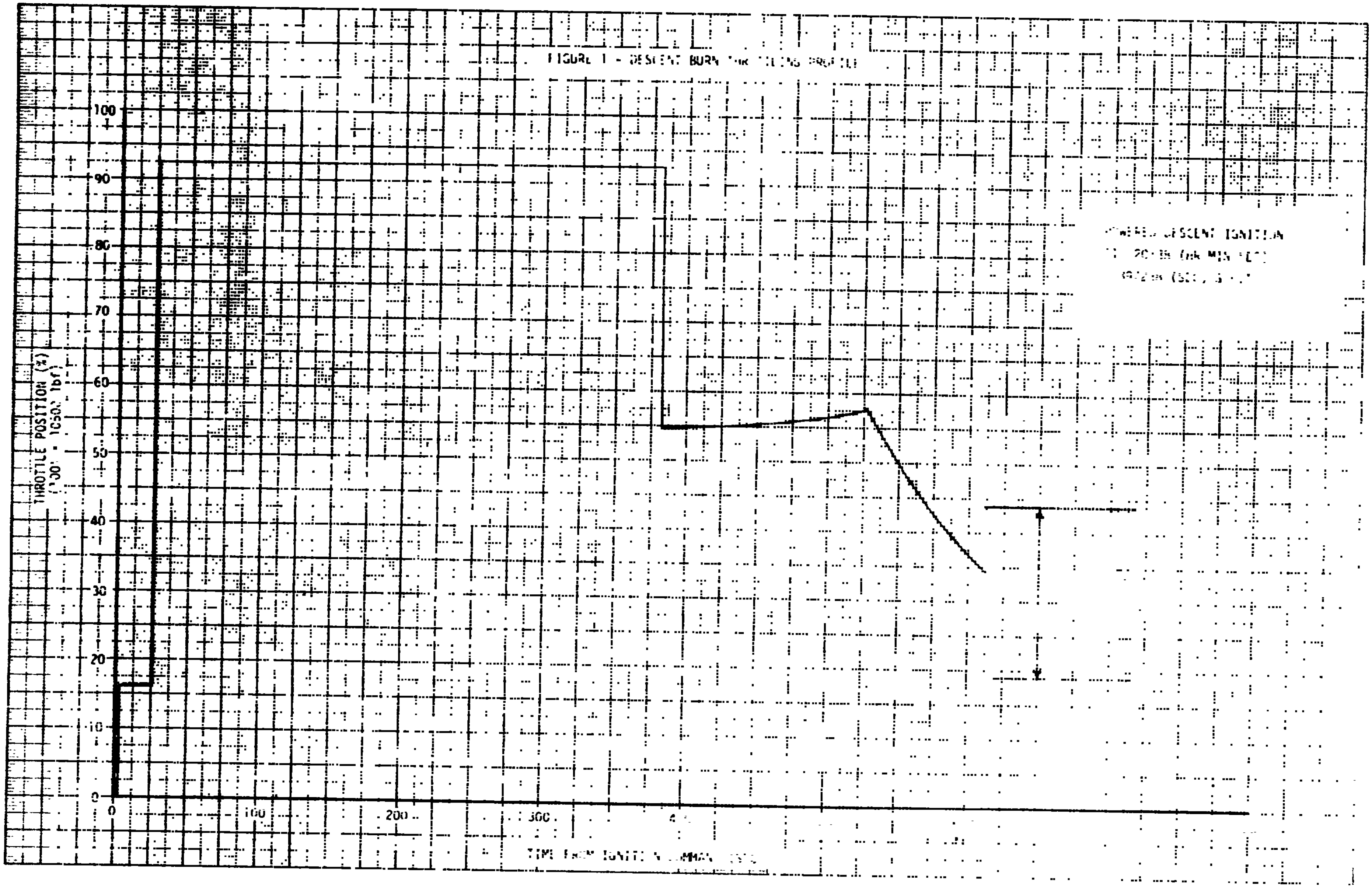
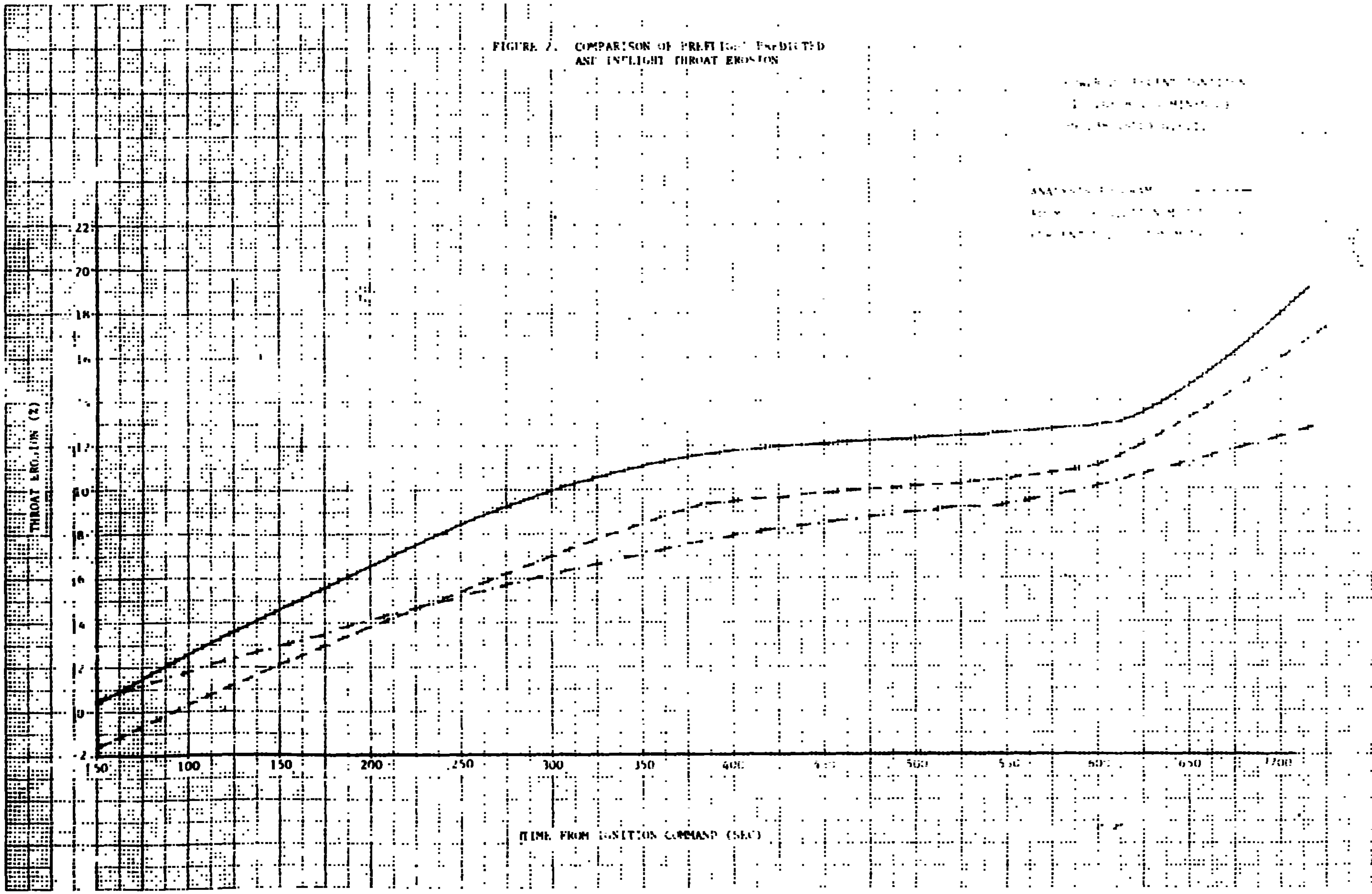
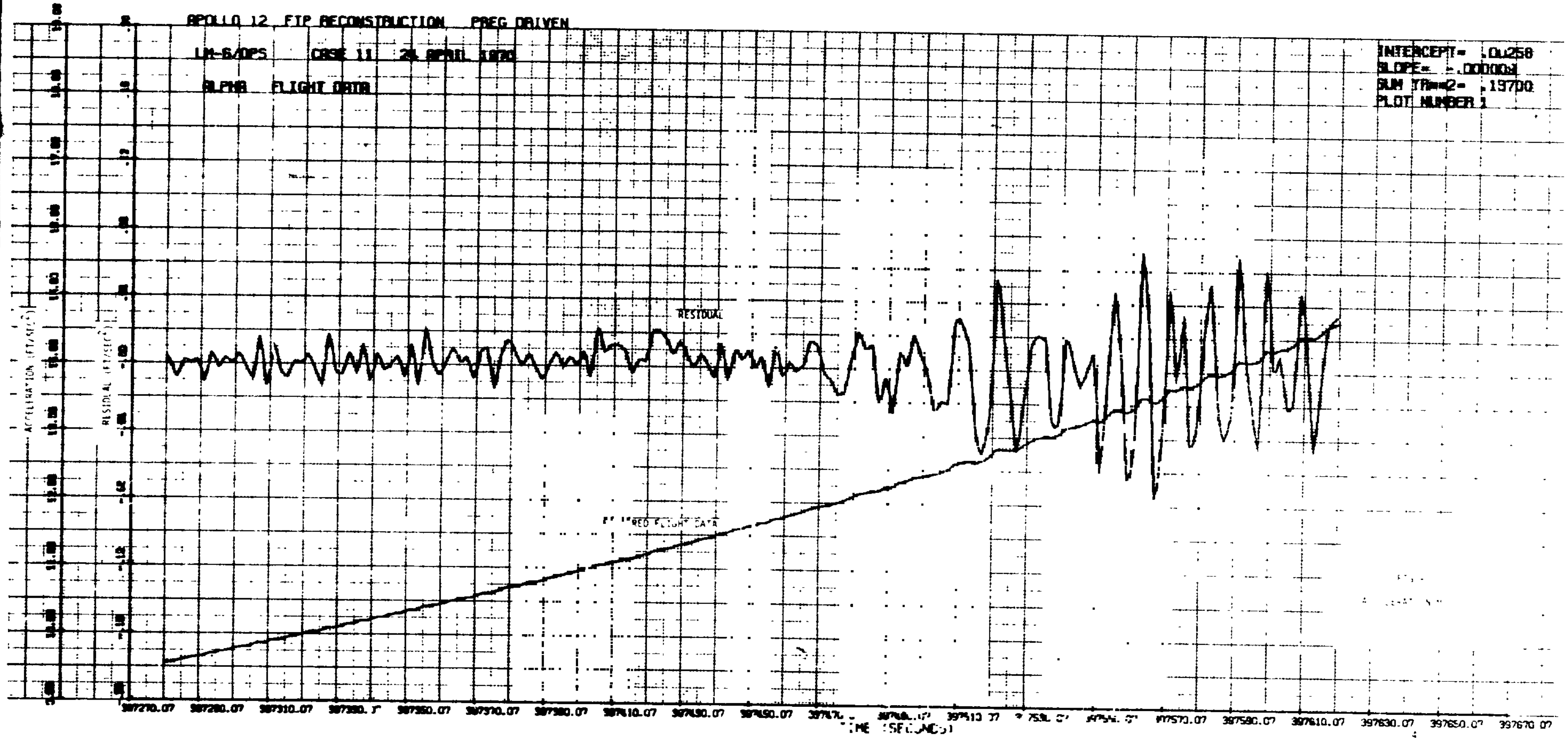


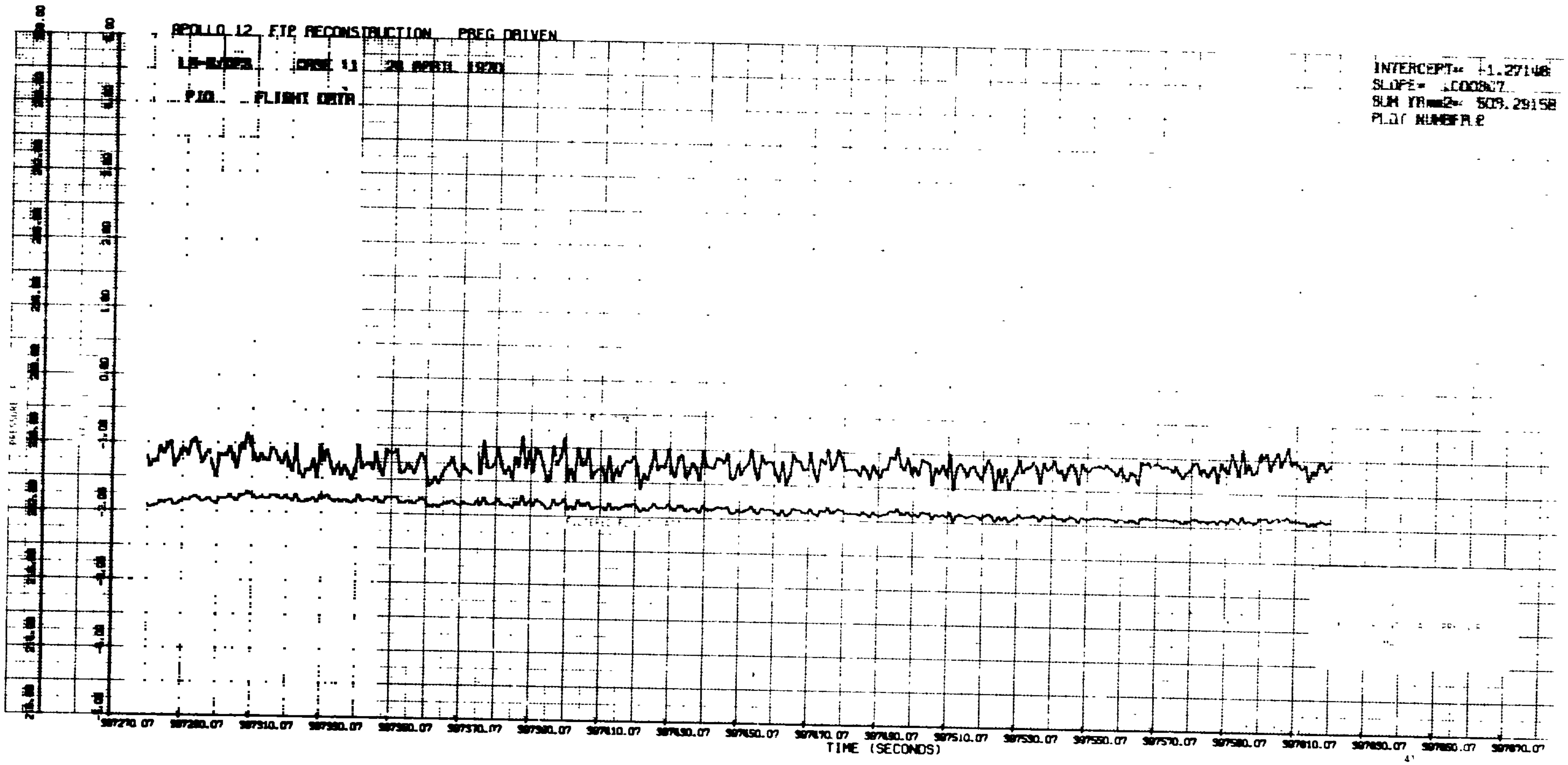
FIGURE 1 - DESCENT BURN THROTTLING PROFILE

FIGURE 7. COMPARISON OF PREFLIGHT PREDICTED AND INFLIGHT THROAT EROSION

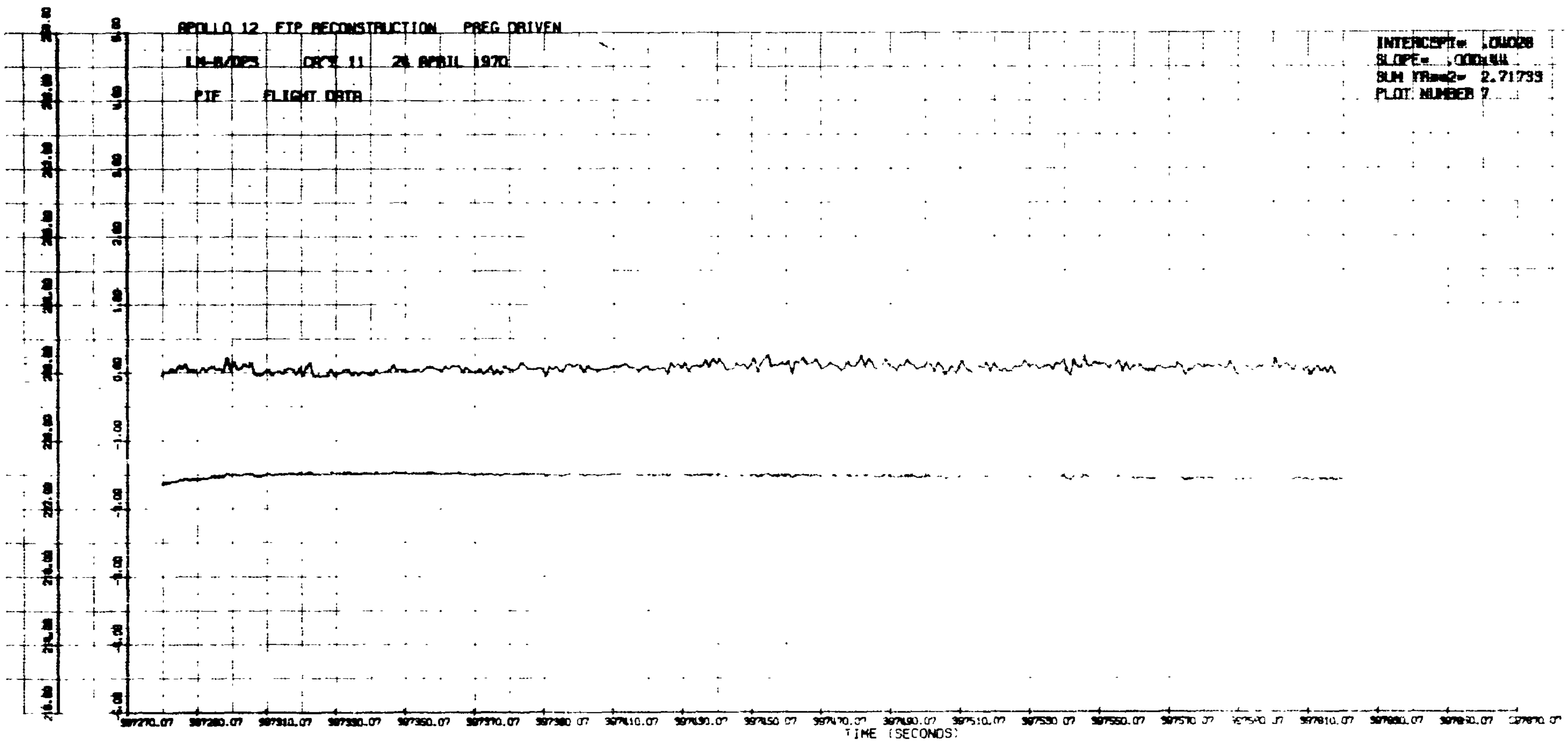




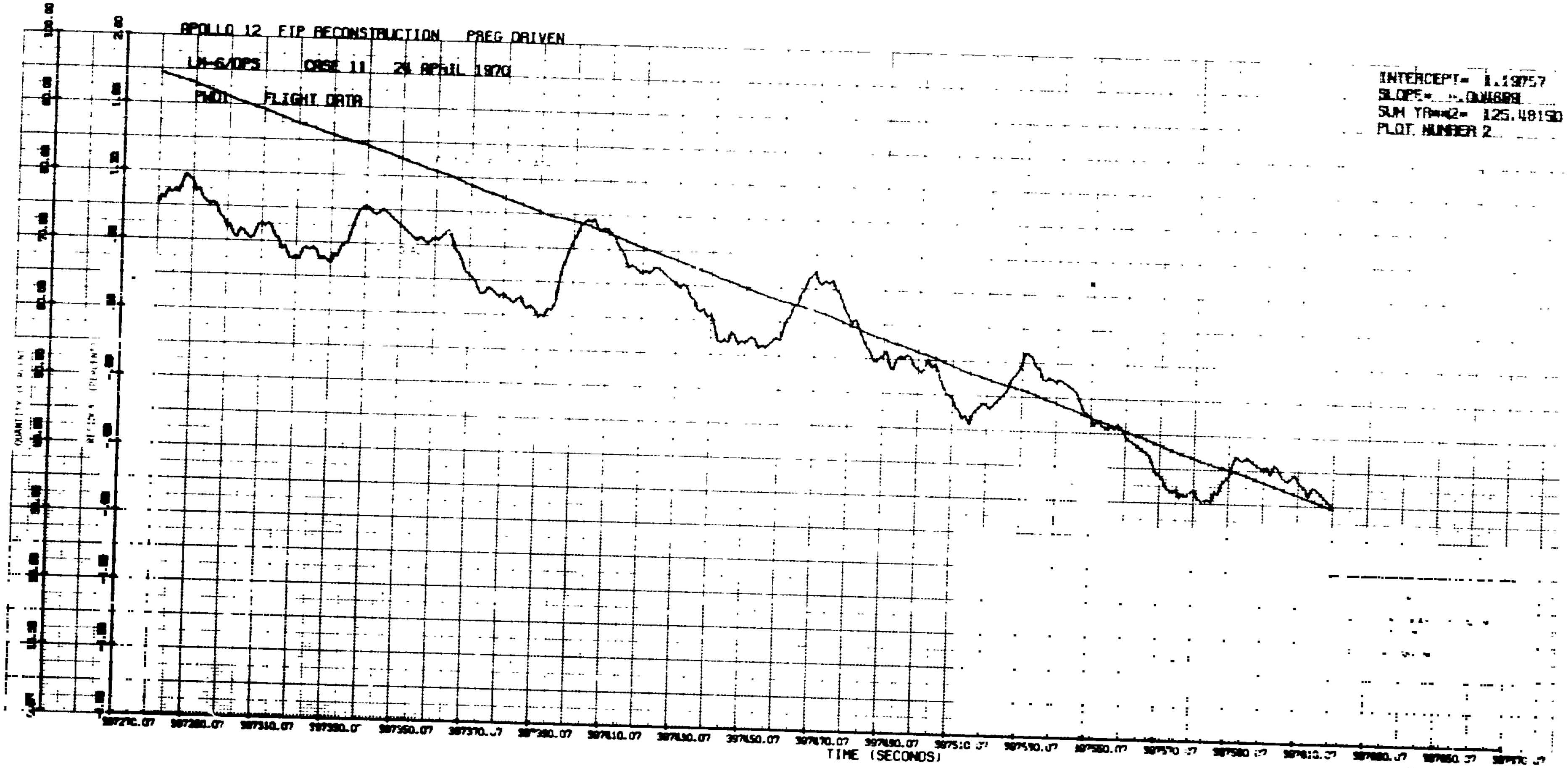
APOLLO 12 FIP RECONSTRUCTION PREG DRIVEN
LM-6/DPS CASE 11 24 APRIL 1970
ALPHA FLIGHT DATA
INTERCEPT= .04258
SLOPE= -.000003
SUM YR**2= .19700
PLOT NUMBER 1
RESIDUAL
ACCELERATION (FT/SEC²)
RESIDUAL (FT/SEC²)
TIME (SECONDS)
397270.07
397290.07
397310.07
397370.07
397390.07
397630.07
397650.07
397670.07

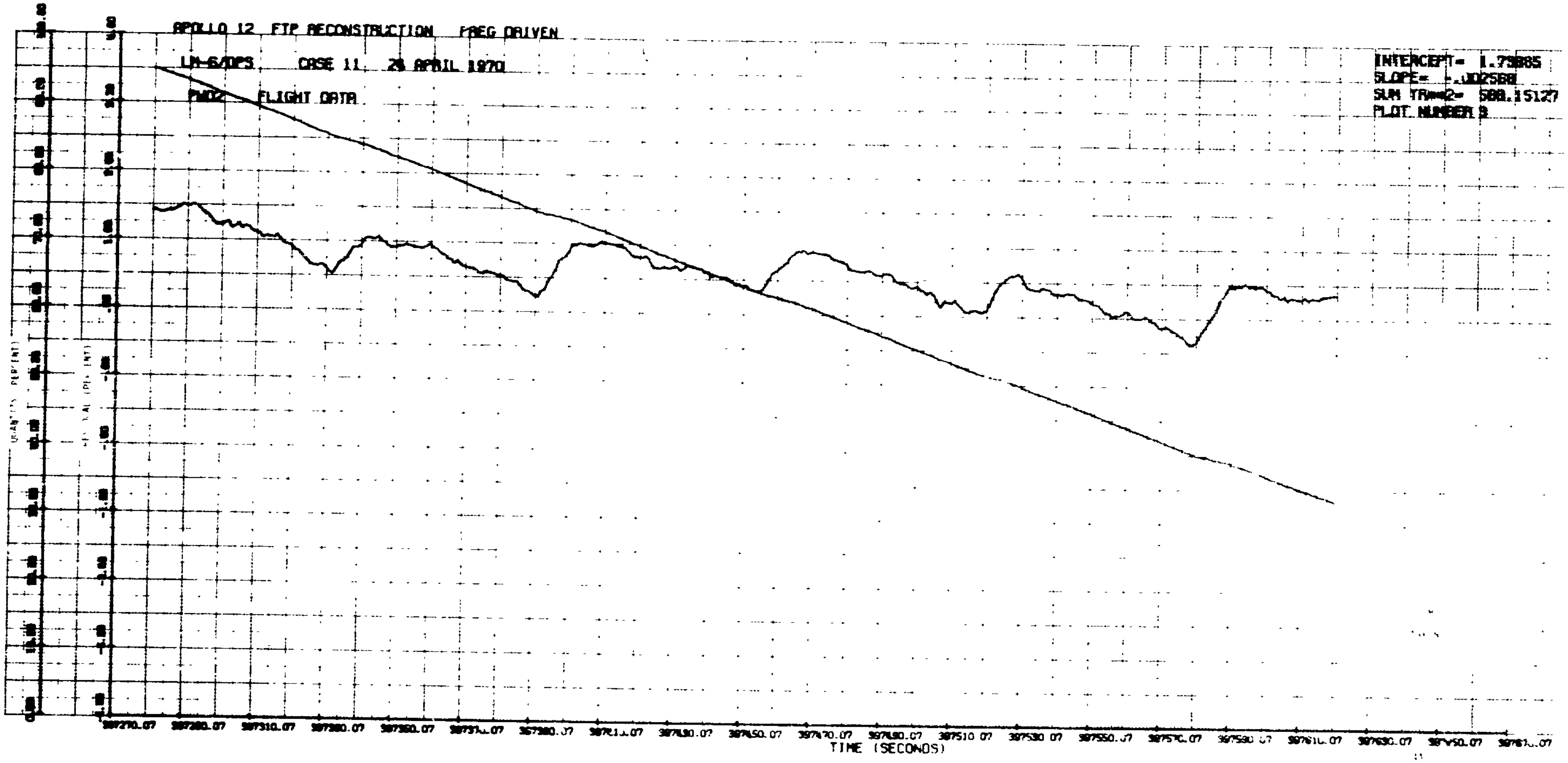


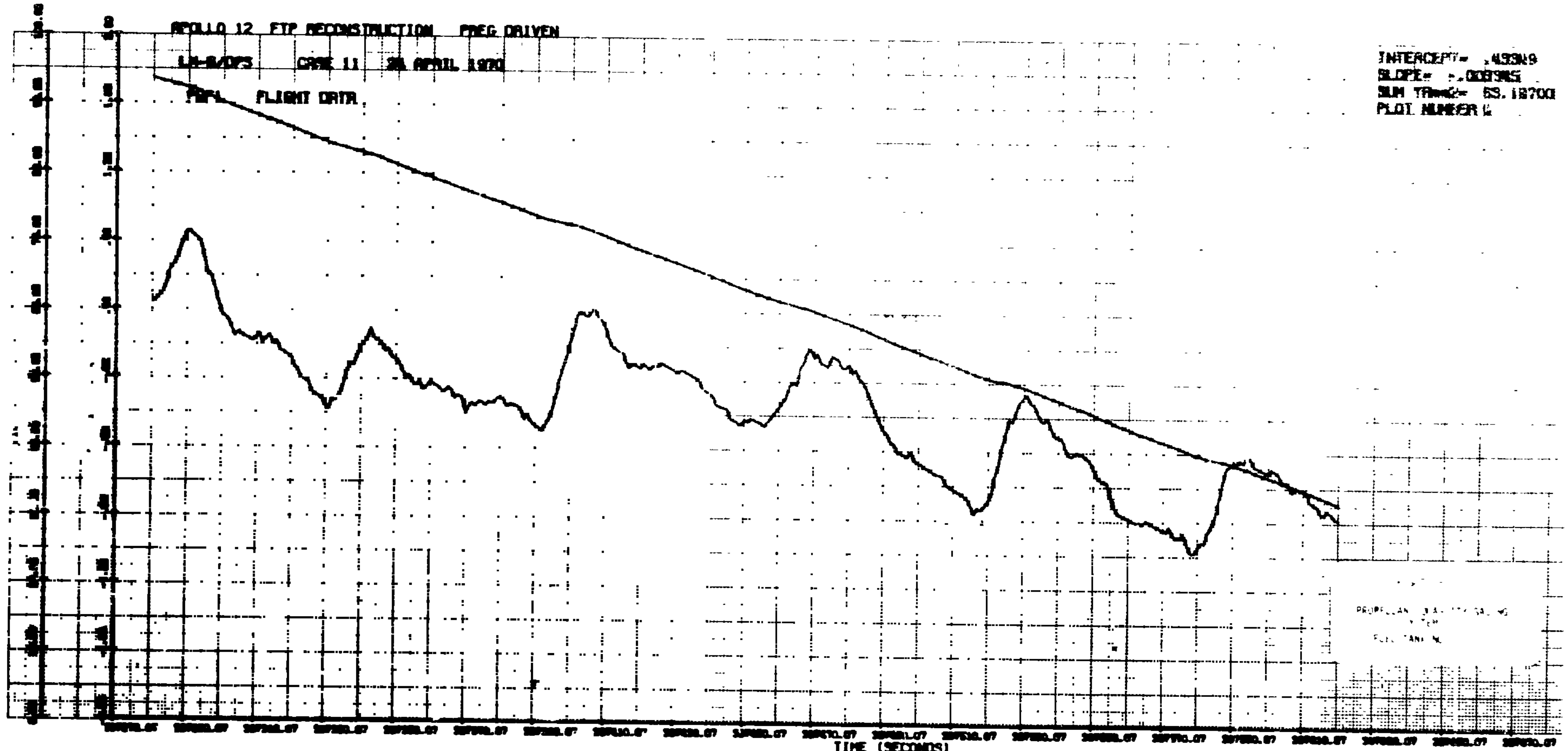
APOLLO 12 FTP RECONSTRUCTION PREG DRIVEN
FLIGHT DATA
INTERCEPT= -1.27148
SUM YR**2= 509.29158
TIME (SECONDS)



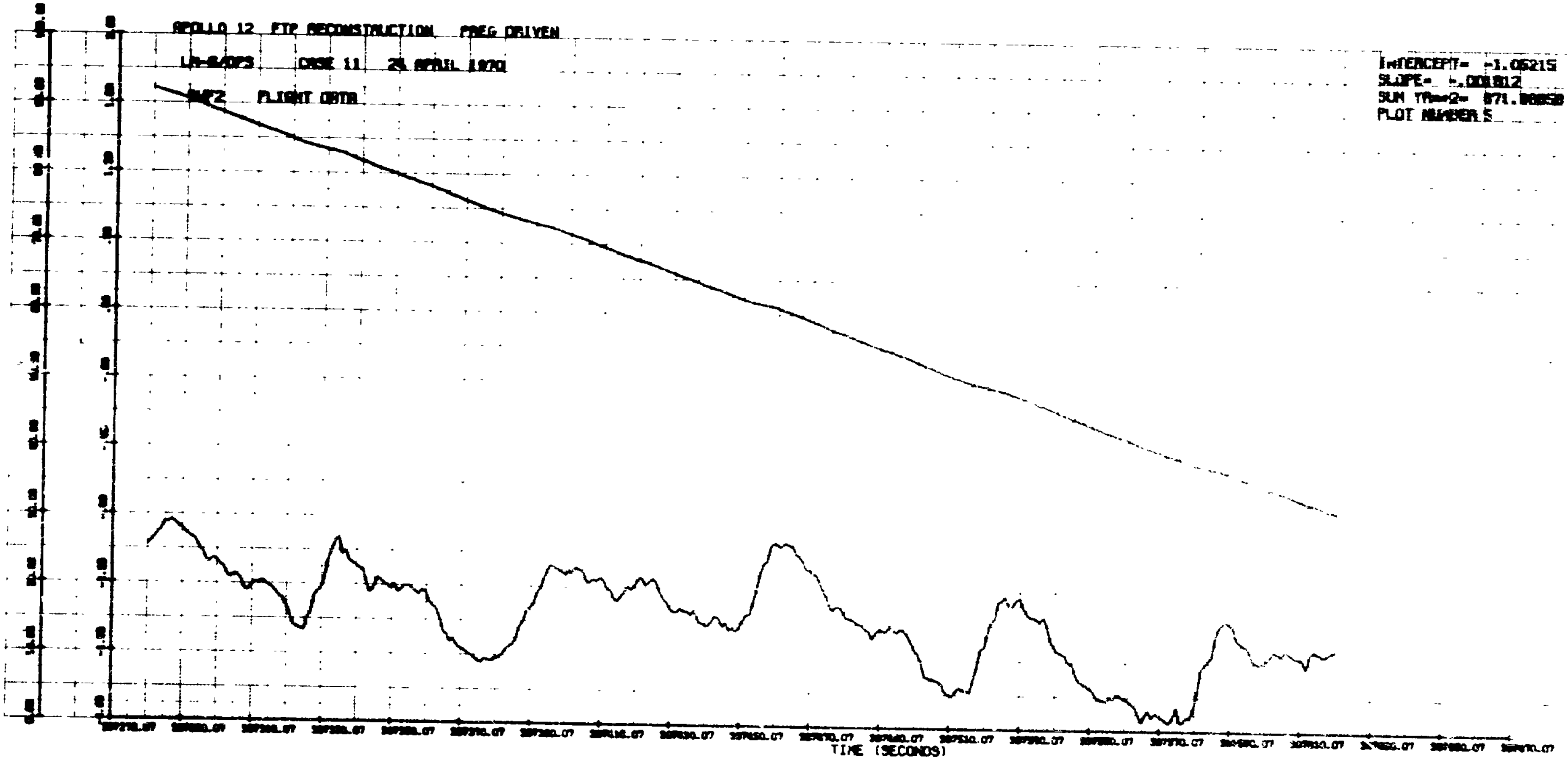
APOLLO 12 FTP RECONSTRUCTION PREG DRIVEN
LM-8/DPS CPCT 11 28 APRIL 1970
PIF FLIGHT DATA
INTERCEPT= .00028
SLOPE= .000144
SUM YR**2= 2.71733
PLOT NUMBER 7
TIME (SECONDS)


APOLLO 12 FTP RECONSTRUCTION PREG DRIVEN
LM-6/DPS CASE 11 24 APRIL 1970
PWDT FLIGHT DATA
INTERCEPT= 1.19757
SUM YR**2= 125.48150
PLOT NUMBER 2
TIME (SECONDS)

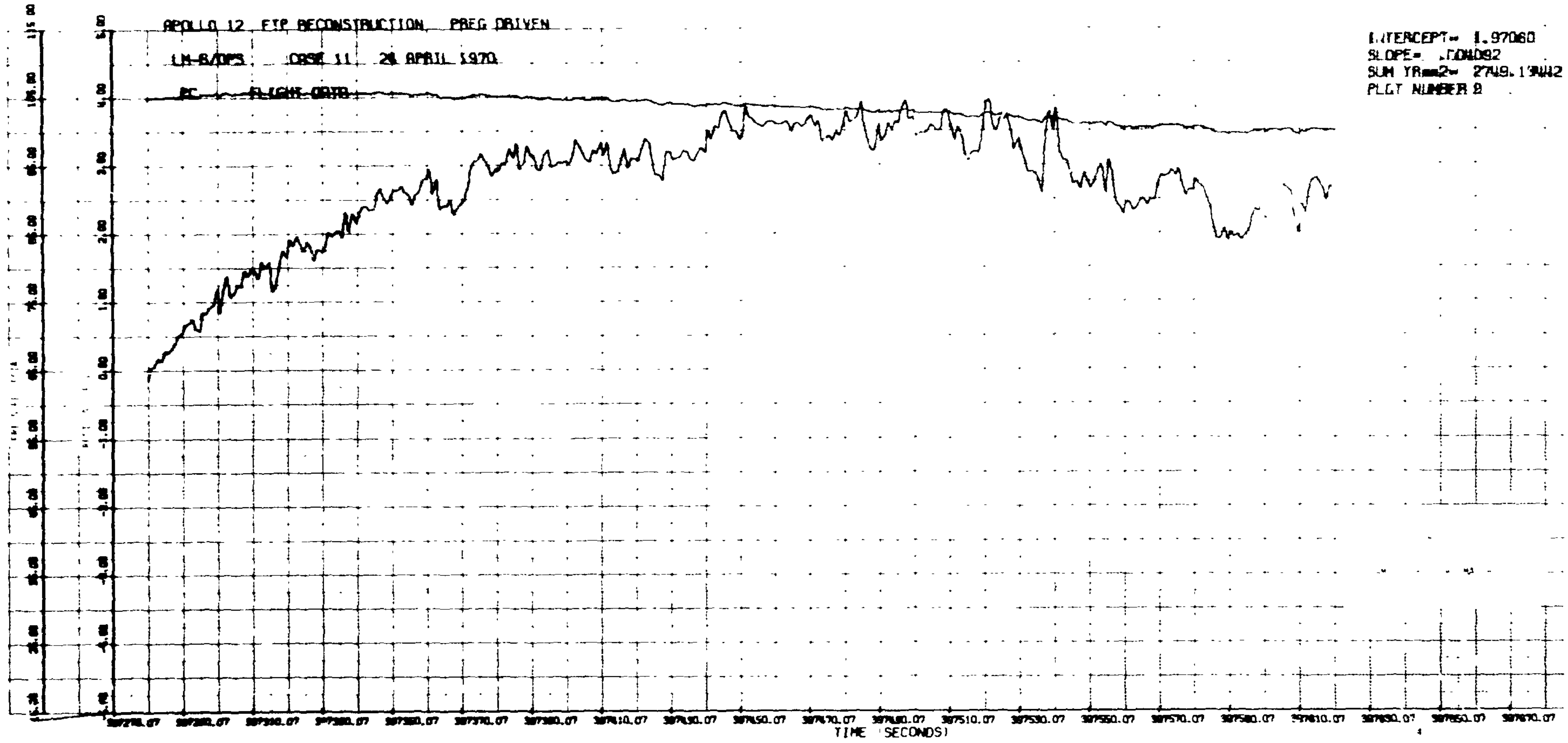

APOLLO 12 FTP RECONSTRUCTION FREQ DRIVEN
LM-6/DPS CASE 11 28 APRIL 1970
PWO2 FLIGHT DATA
INTERCEPT= 1.79885
SLOPE= -.002588
SUM TR**2= 588.15127
PLOT NUMBER 9
TIME (SECONDS)



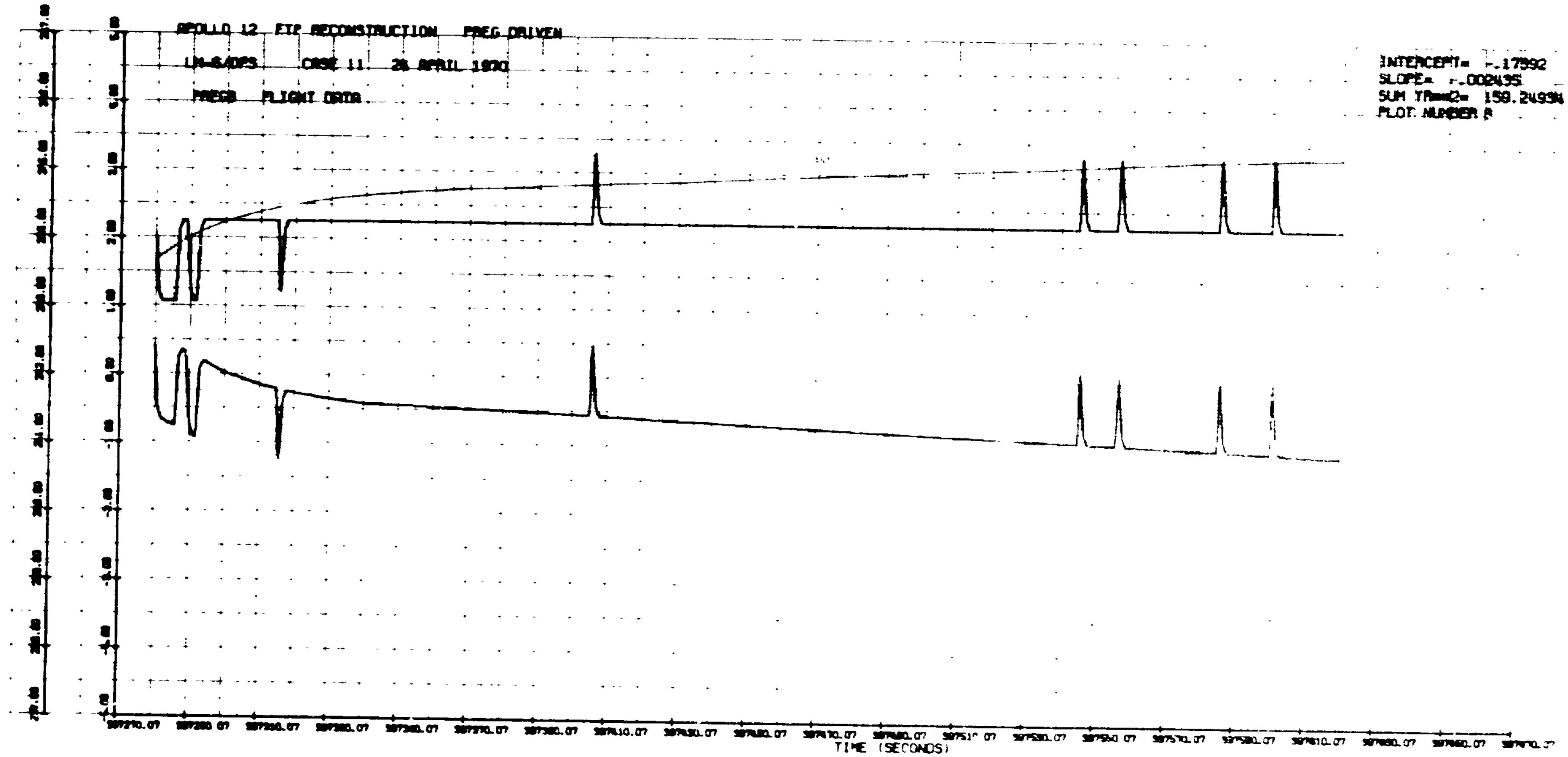
APOLLO 12 FTP RECONSTRUCTION PREG DRIVEN
LM-6/DPS CASE 11 28 APRIL 1970
PQFL FLIGHT DATA
INTERCEPT= .45349
SLOPE= -.00395
SUM TR**2= 53.18700
PLOT NUMBER 6
TIME (SECONDS)



APOLLO 12 FTP RECONSTRUCTION PREG DRIVEN
LM-8/DPS CASE 11 28 APRIL 1970
FLIGHT DATA
INTERCEPT= -1.06215
SUM TR**2= 871.80858
PLOT NUMBER 5
TIME (SECONDS)



APOLLO 12 FTP RECONSTRUCTION PREG DRIVEN
LM-6/DPS CASE 11 20 APRIL 1970
PC FLIGHT DATA
INTERCEPT= 1.97060
SLOPE= .00092
SUM YR**2= 2749.19412
PLOT NUMBER 8
TIME (SECONDS)



APOLLO 12 FTP RECONSTRUCTION PREG DRIVEN
LM-6/DPS CASE 11 28 APRIL 1970
PREGB FLIGHT DATA
INTERCEPT= -.17992
SLOPE= -.002435
SUM YR**2= 159.2493N
PLOT NUMBER R
TIME (SECONDS)

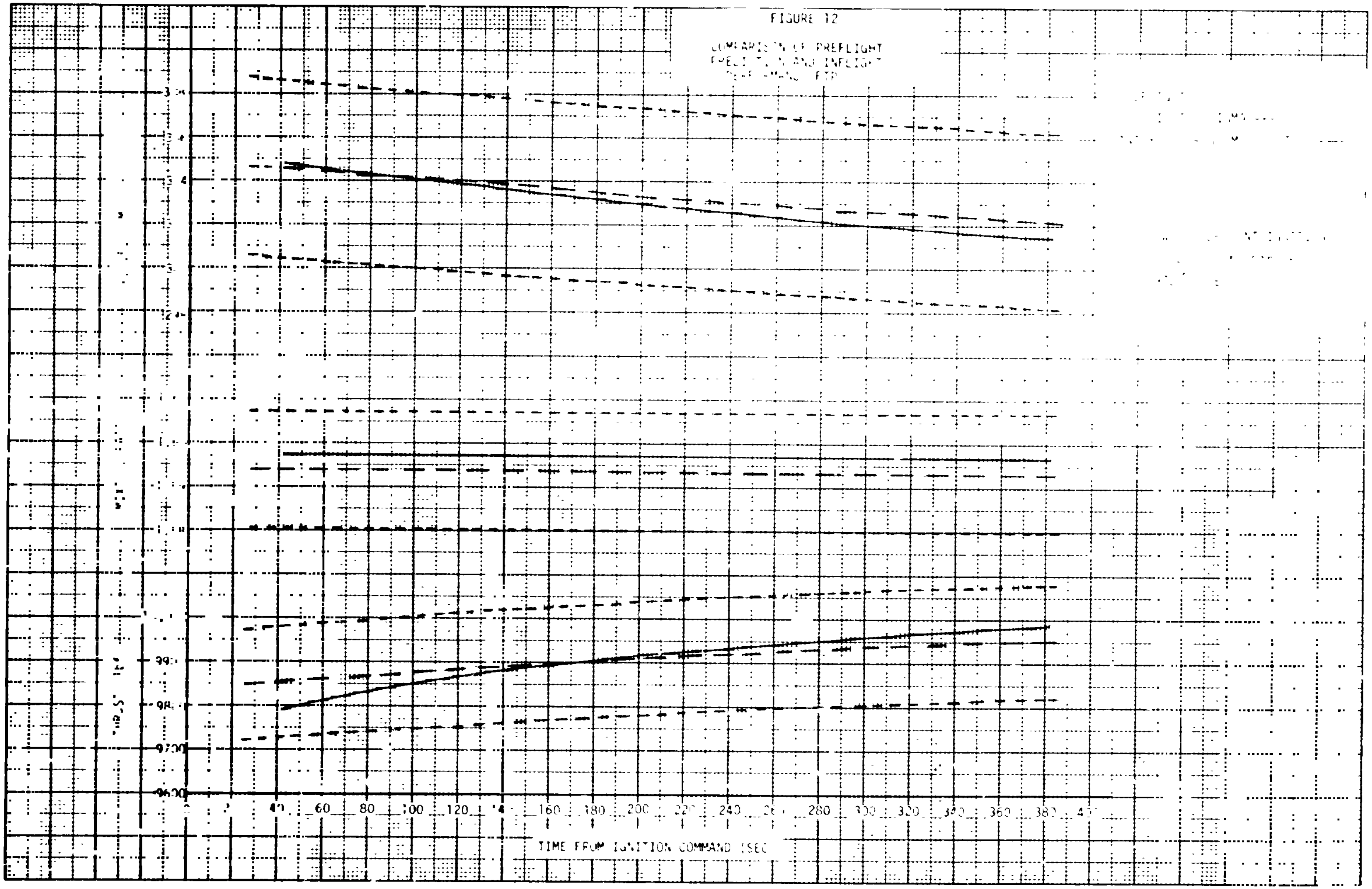
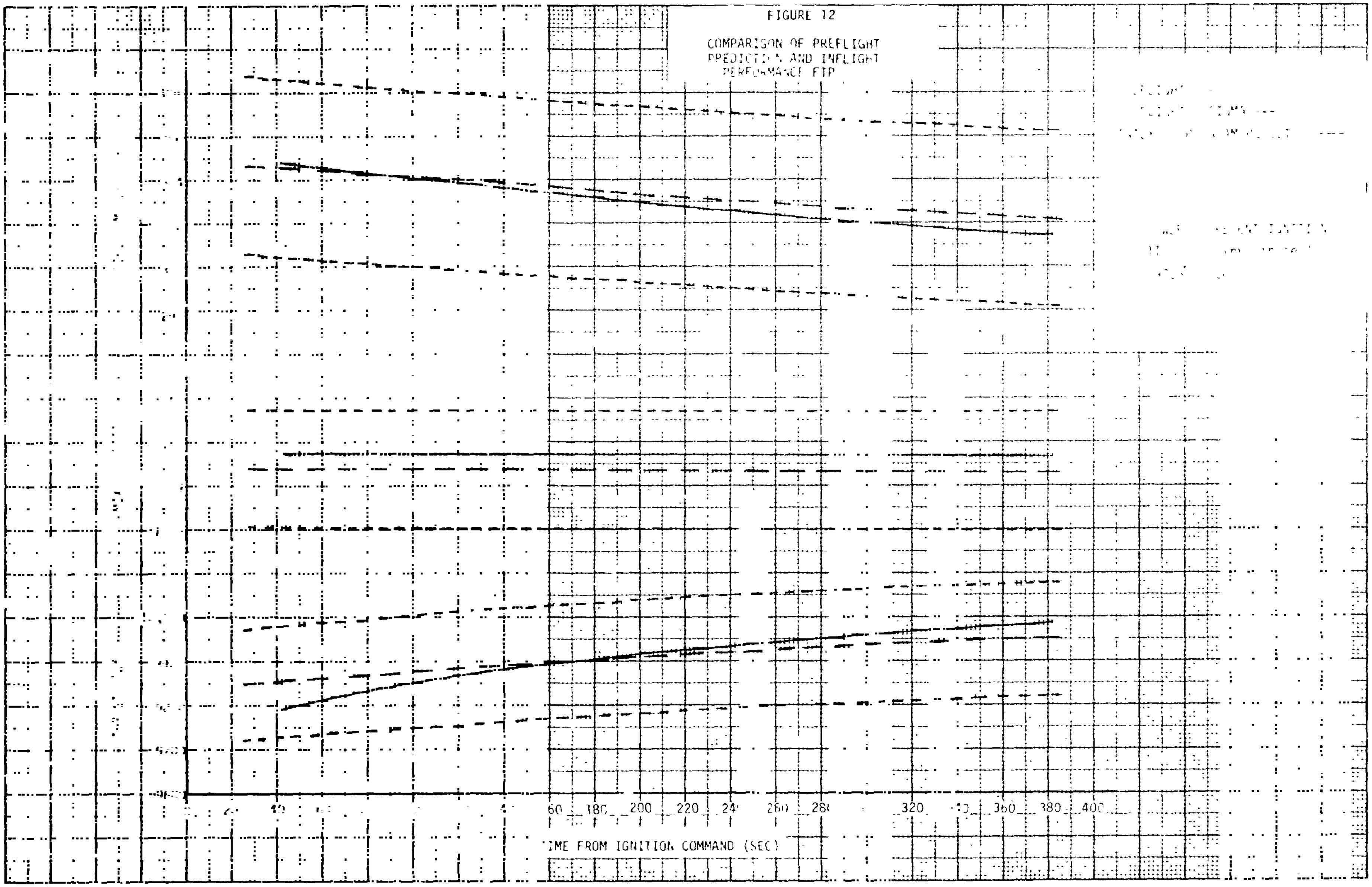


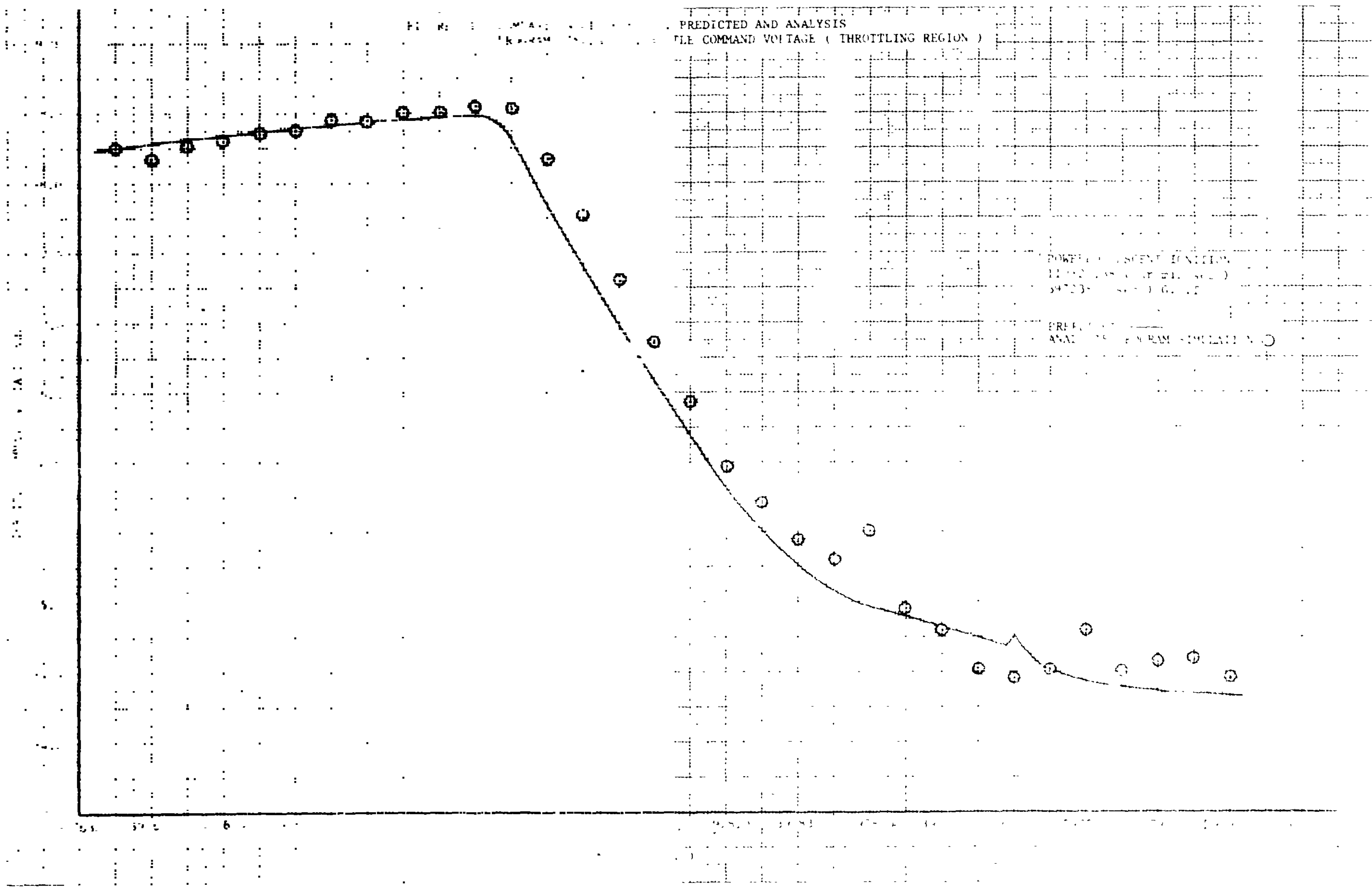
FIGURE 12

COMPARISON OF PREFLIGHT PREDICTION AND INFLIGHT

FIGURE 12

COMPARISON OF PREFLIGHT PREDICTION AND INFLIGHT PERFORMANCE FTP




PREDICTED AND ANALYSIS
COMMAND VOLTAGE (THROTTLING REGION)

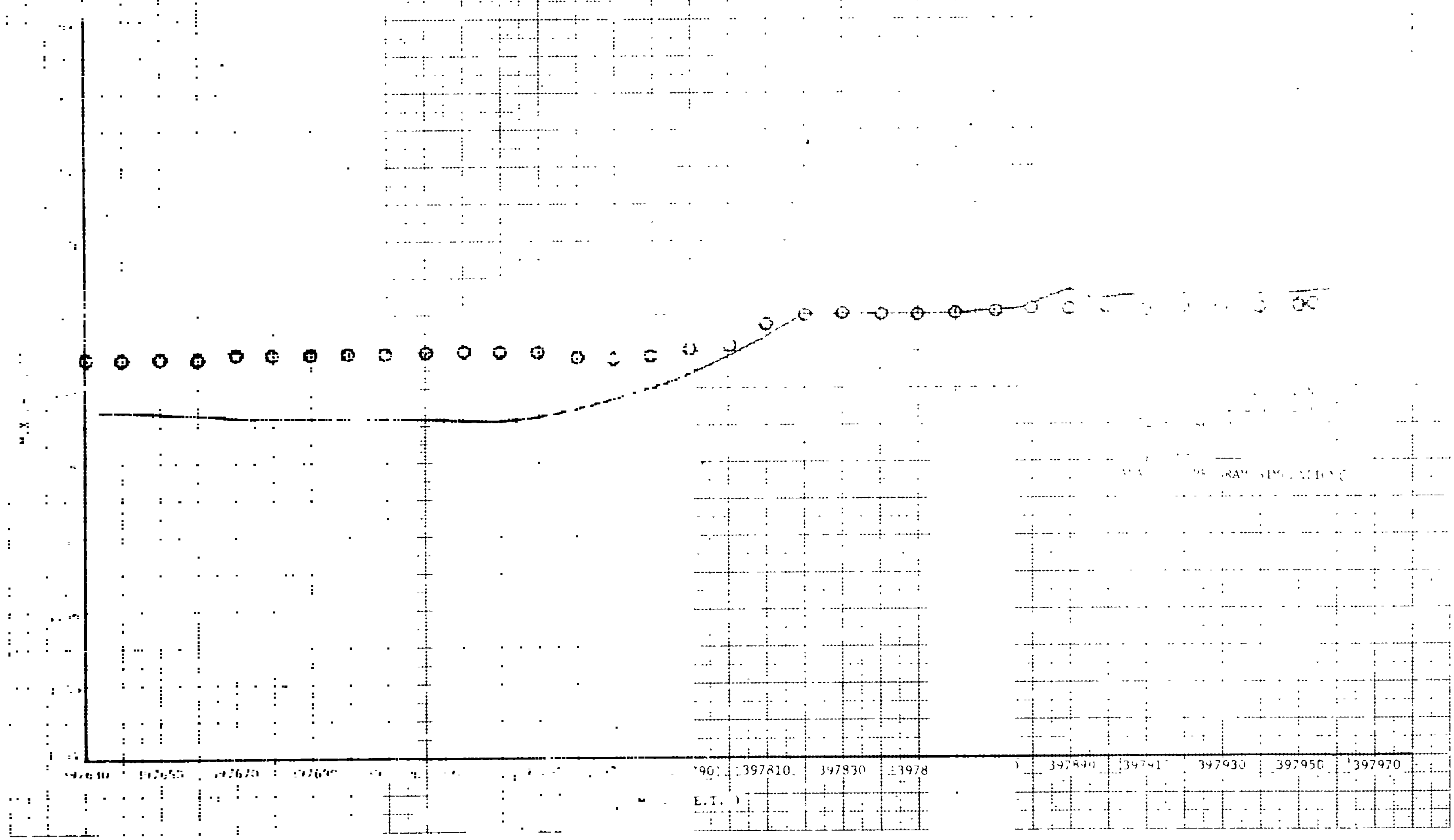


FIGURE 14. COMPARISON OF PREFLIGHT PREDI[illegible] ANALYSIS
PROGRAM SIMULATED MIXTURE RA[illegible]

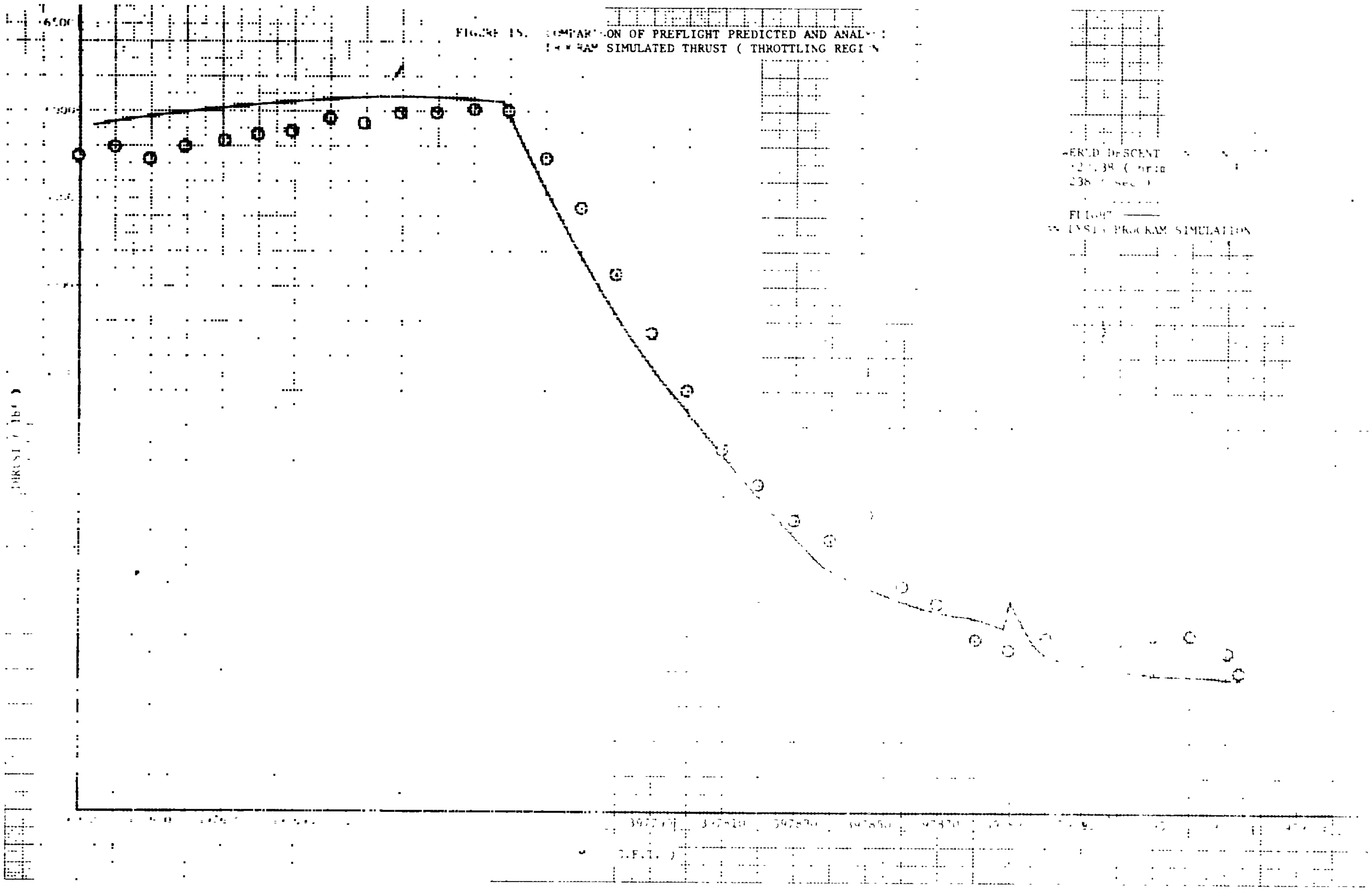
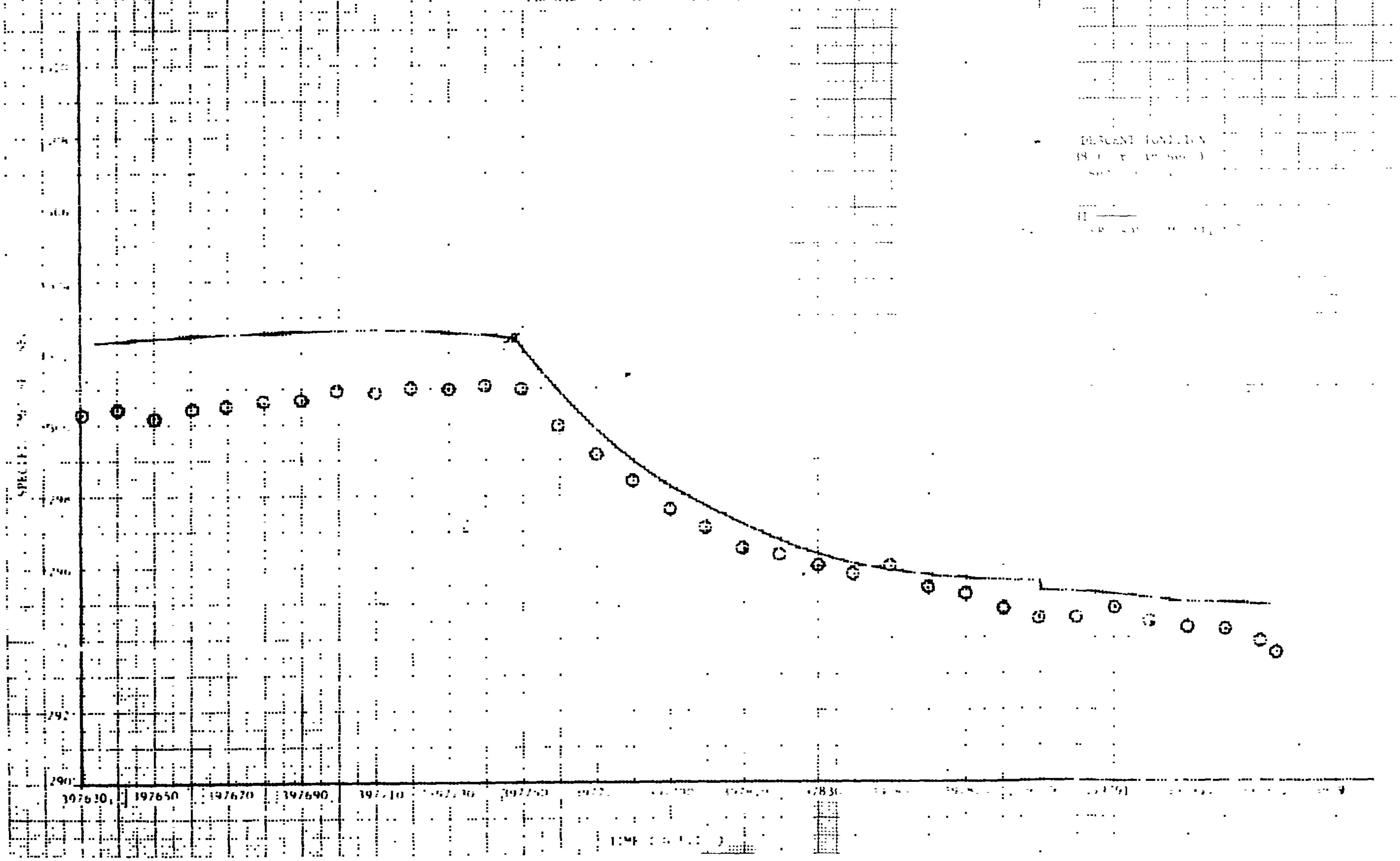
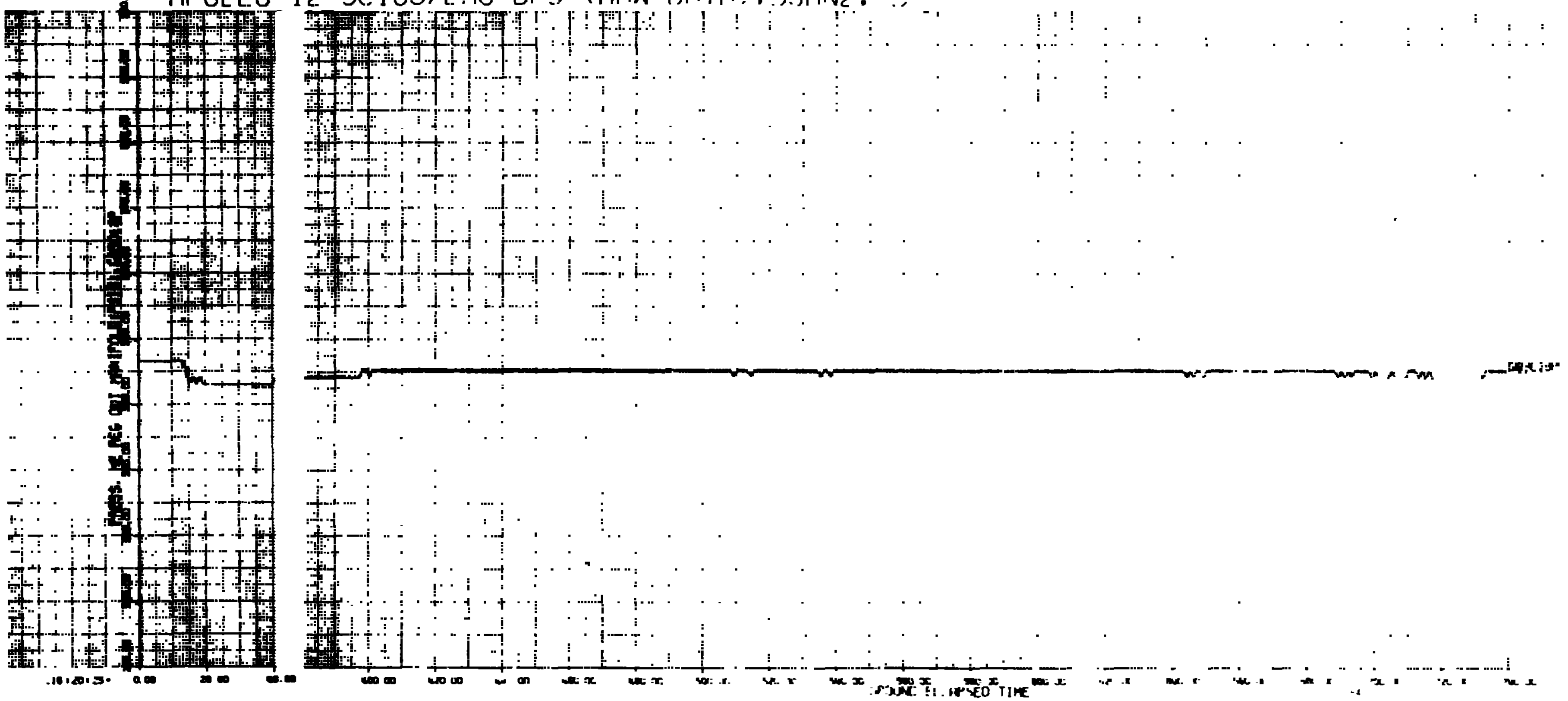


FIGURE 15. COMPARISON OF PREFLIGHT PREDICTED AND ANALYSIS PROGRAM SIMULATED THRUST (THROTTLING REGION)

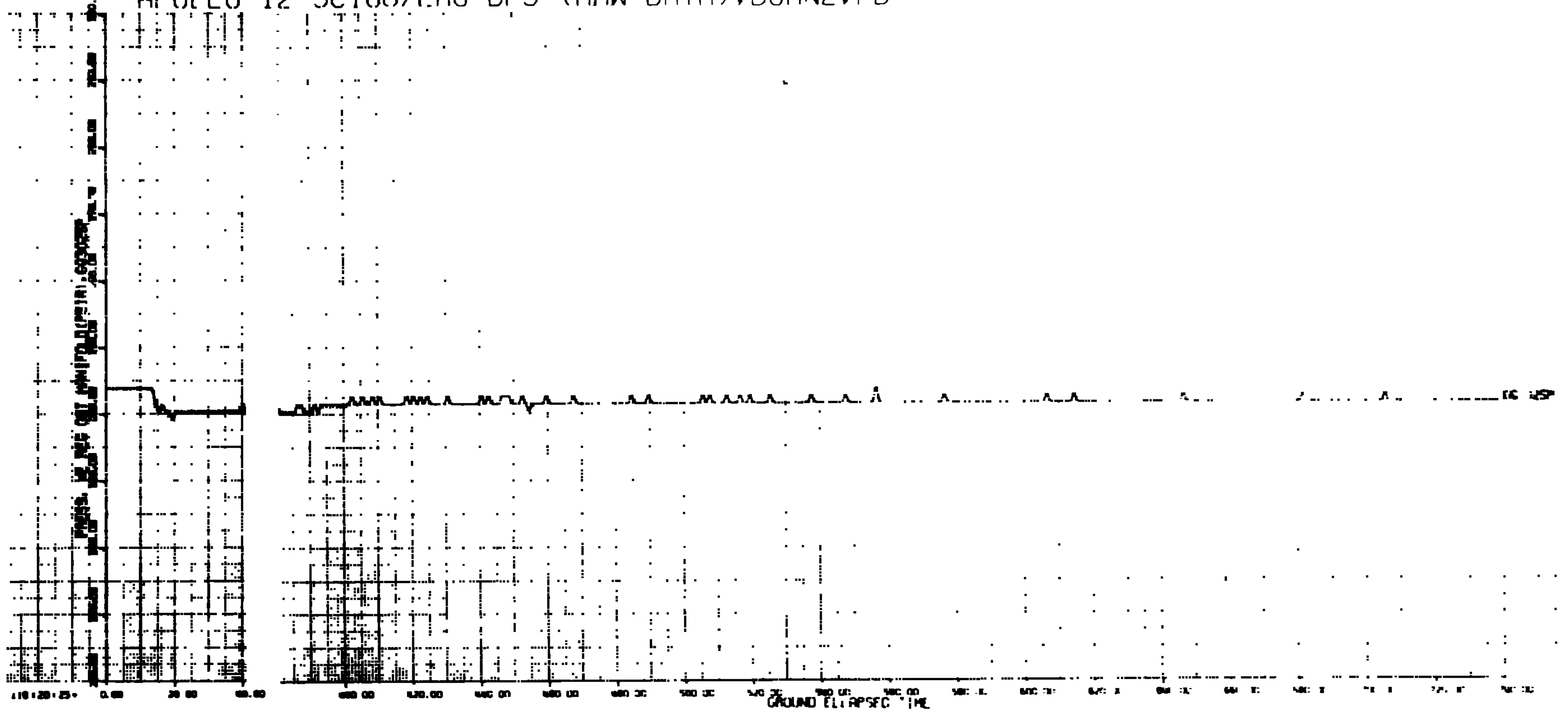
FIGURE . COMPARISON ... PREDICTED AND ANALYSIS PROGRAM ... ENGINE SPECIFIC IMPULSE (THROTTLING REGION)



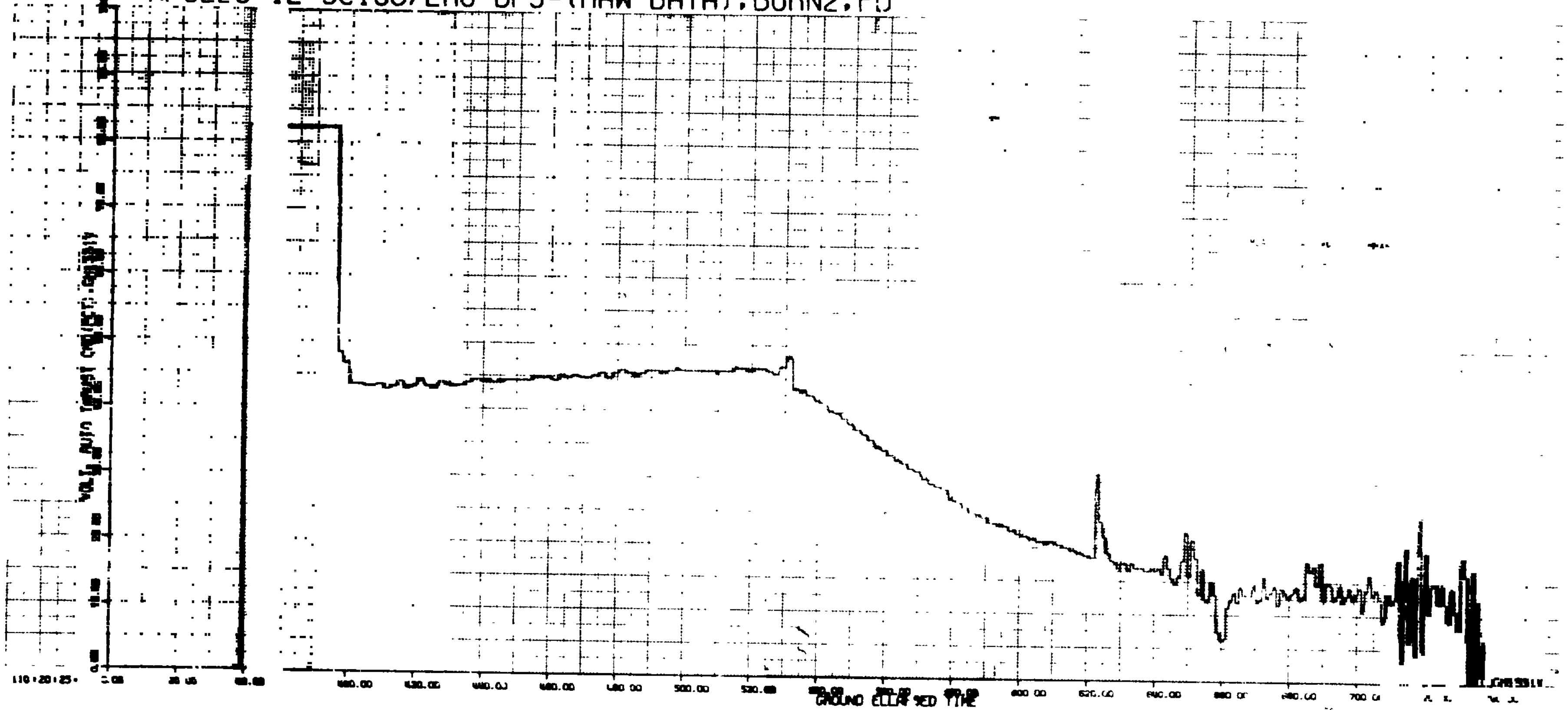
APOLLO 12 SC108/LM6 DPS (RAW DATA), BURN2, PC

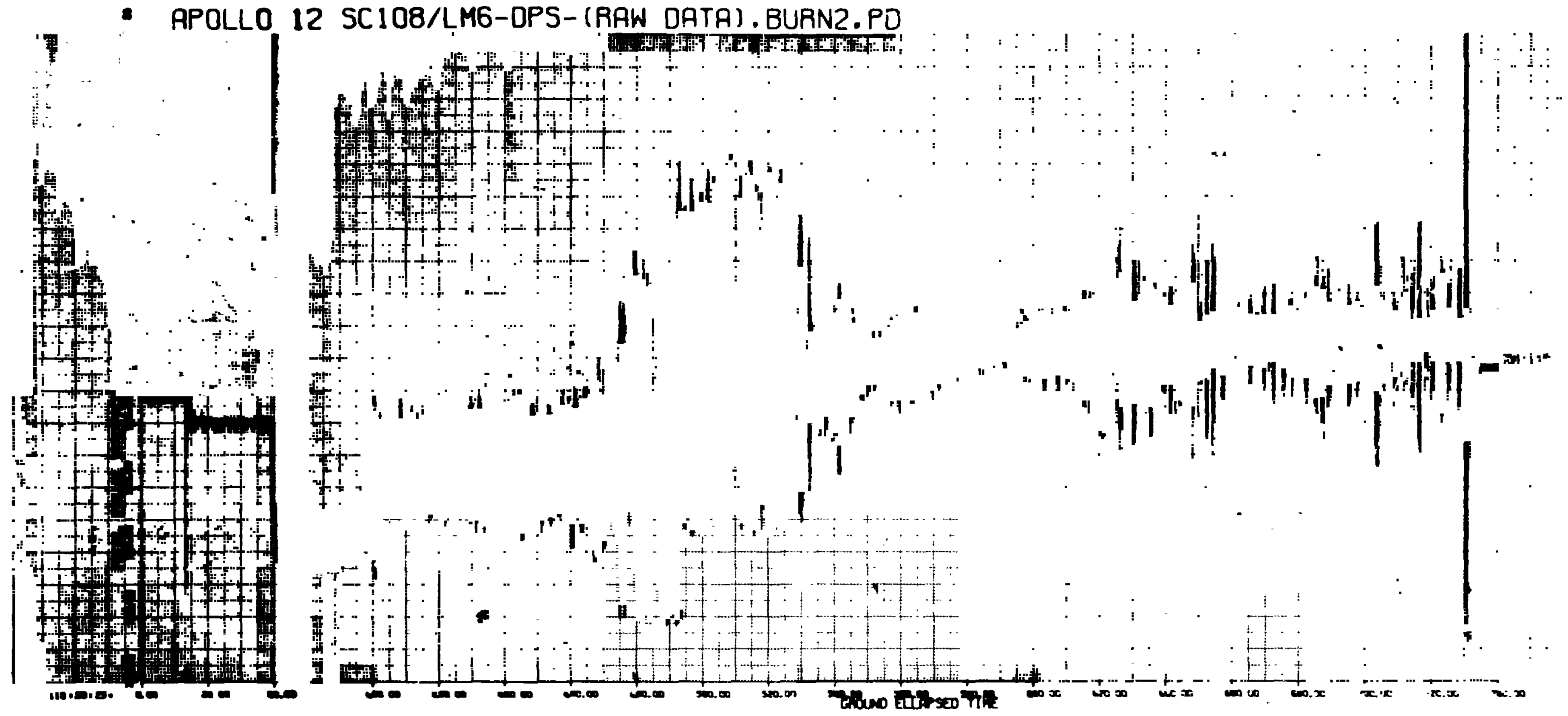


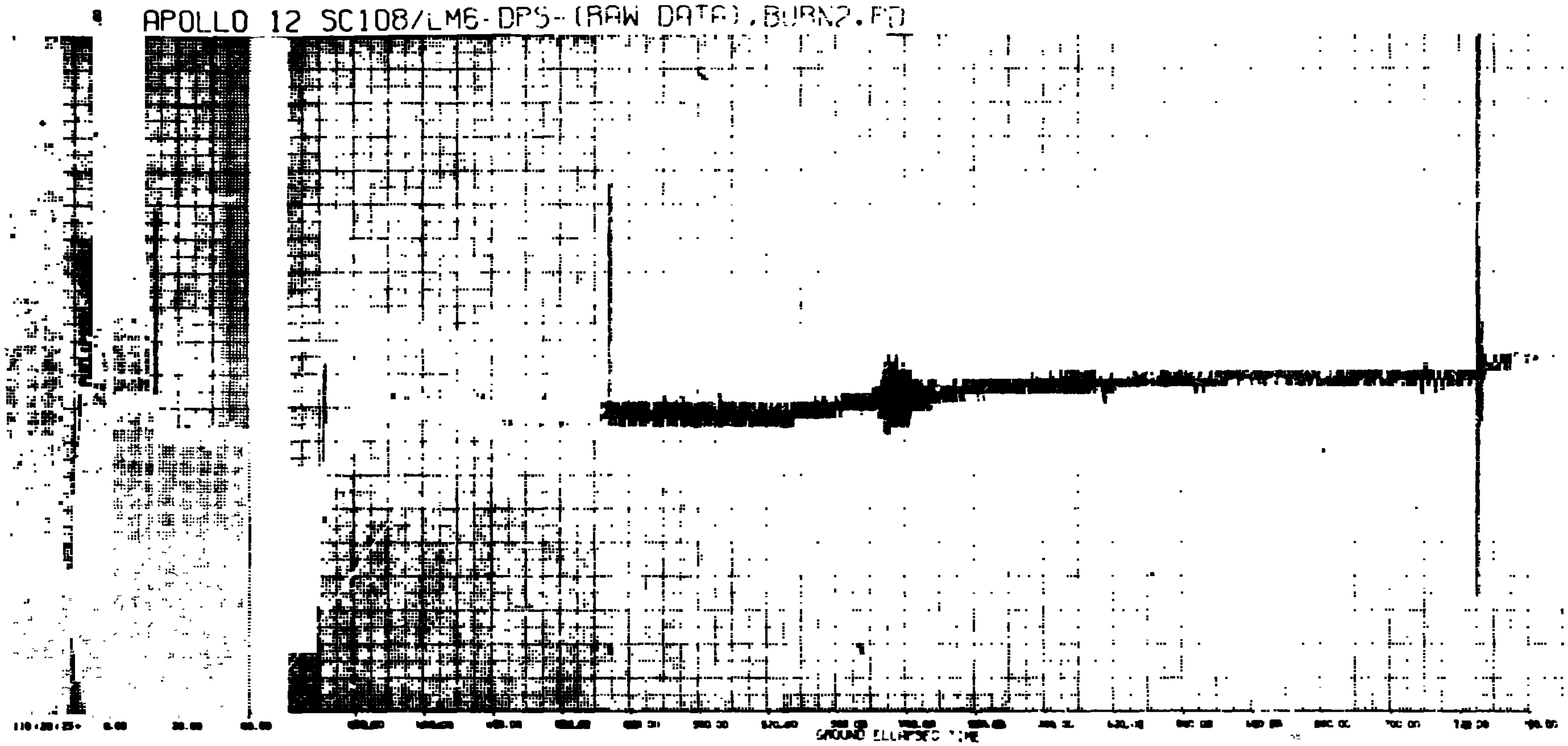
APOLLO 12 SC108/LM6-DPS-(RAW DATA),BURN2,PD



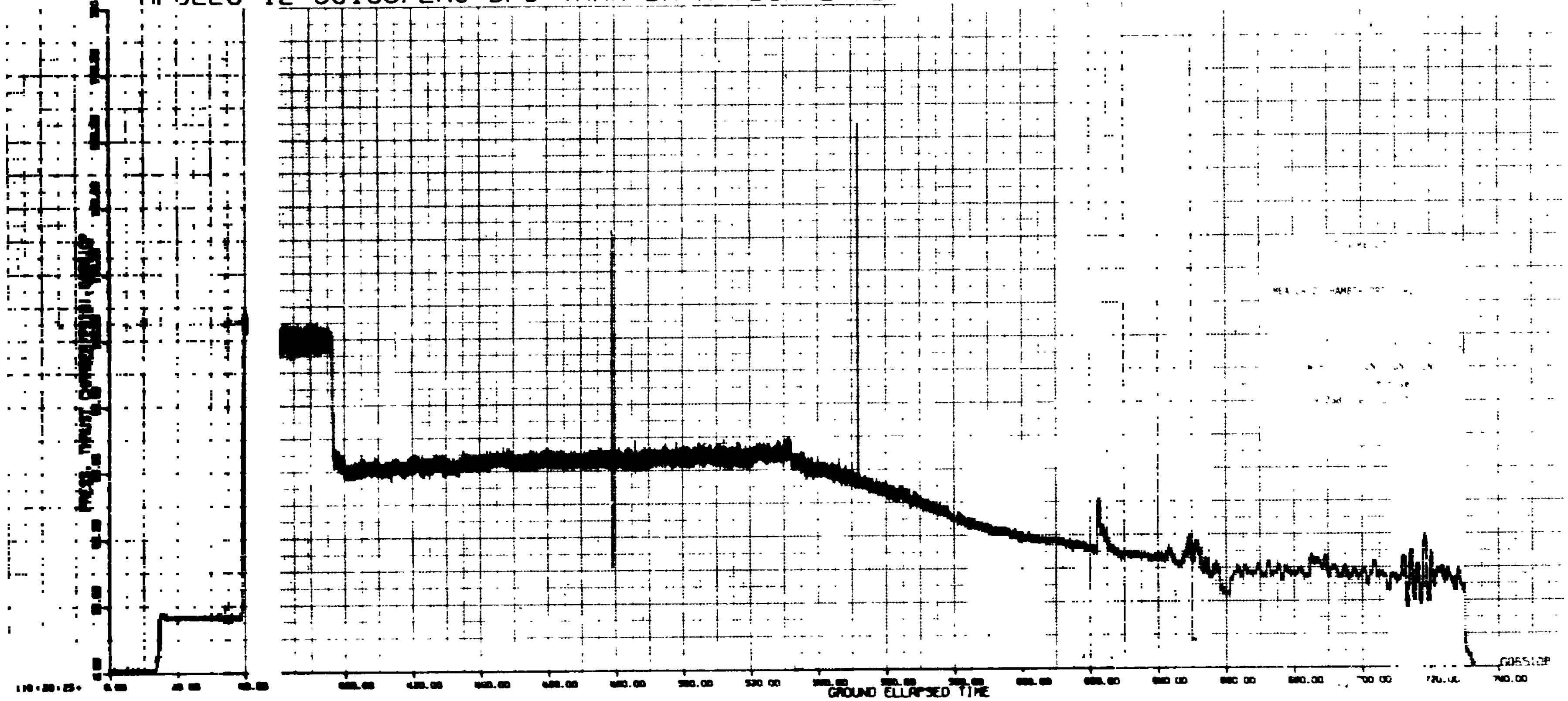
APOLLO 12 SC108/LM6-DPS-(RAW DATA),BURN2,PD




APOLLO 12 SC108/LM6-DPS-(RAW DATA).BURN2.PD
GROUND ELAPSED TIME


APOLLO 12 SC108/LM6-DPS-(RAW DATA),BURN2,FD
GROUND ELLAPSED TIME

APOLLO 12 SC108/LM6-DPS-(RAW DATA).BURN2.PD



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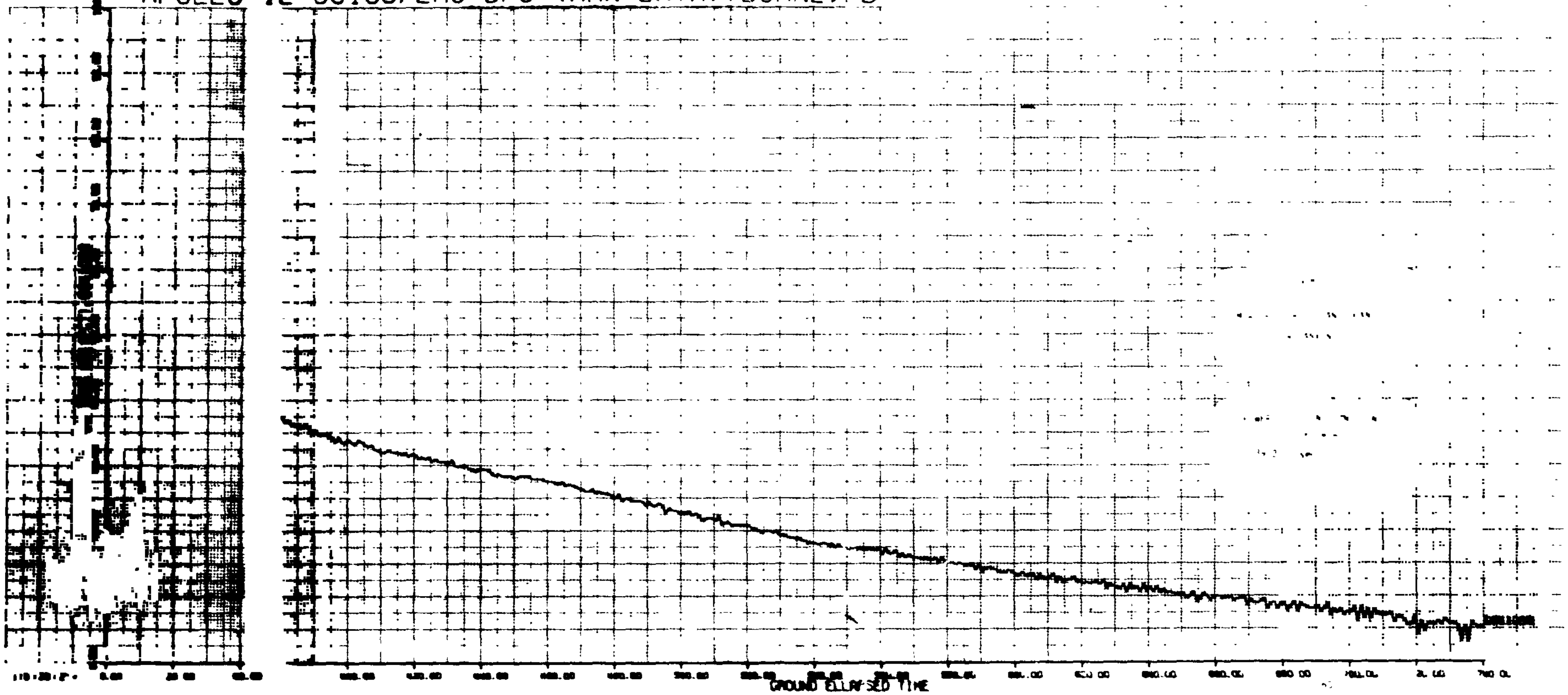
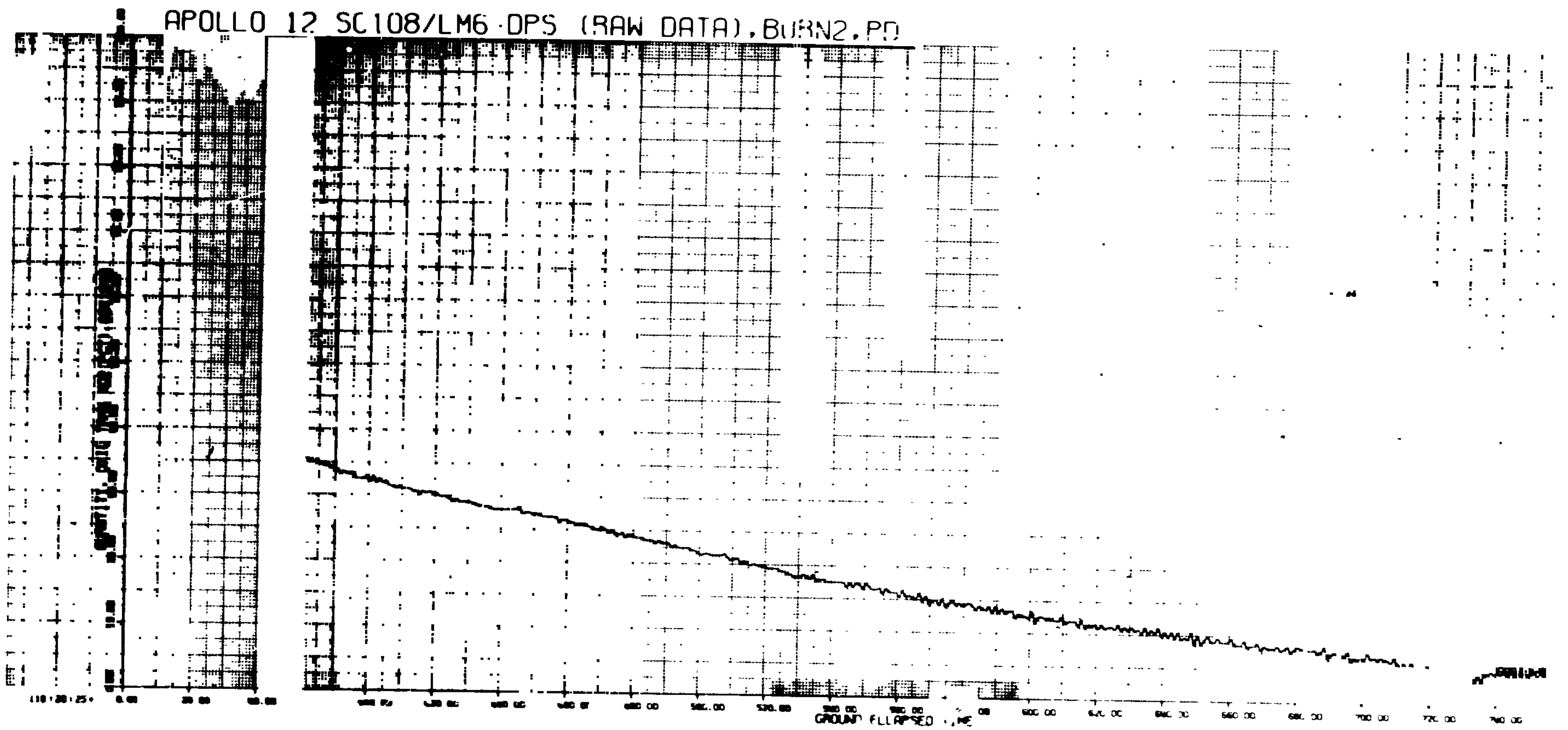
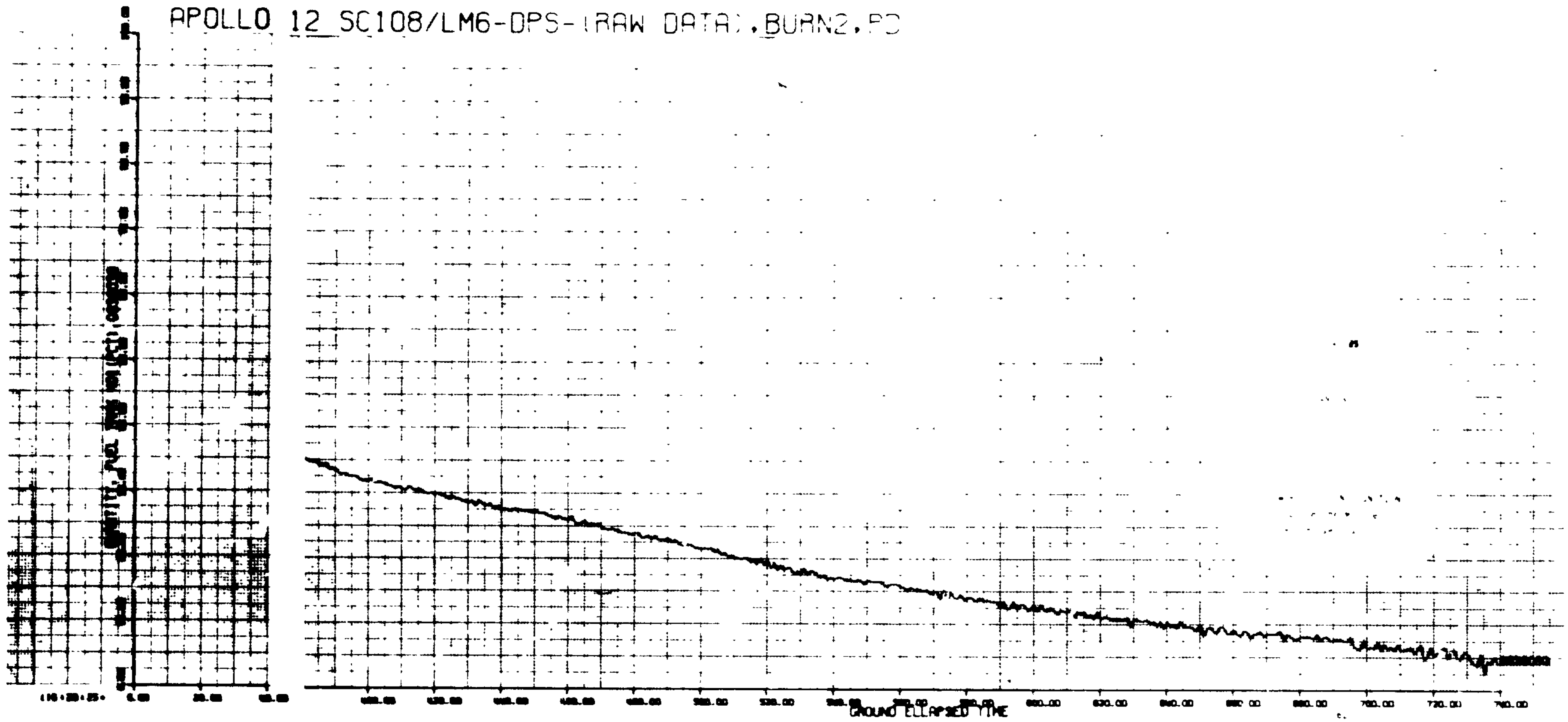
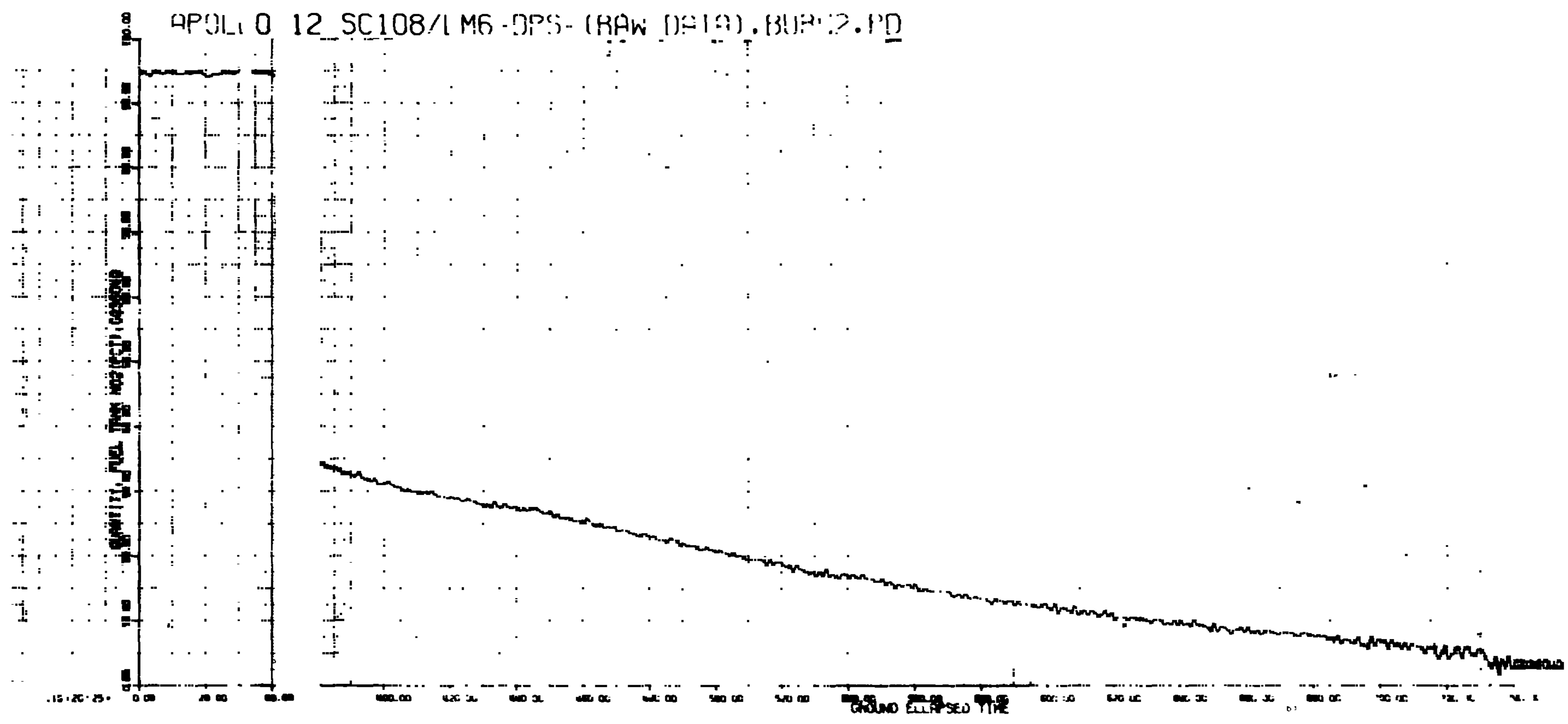



APOLLO 12 SC108/LM6 DPS (RAW DATA).BURN2.PD
GROUND ELLAPSED TIME

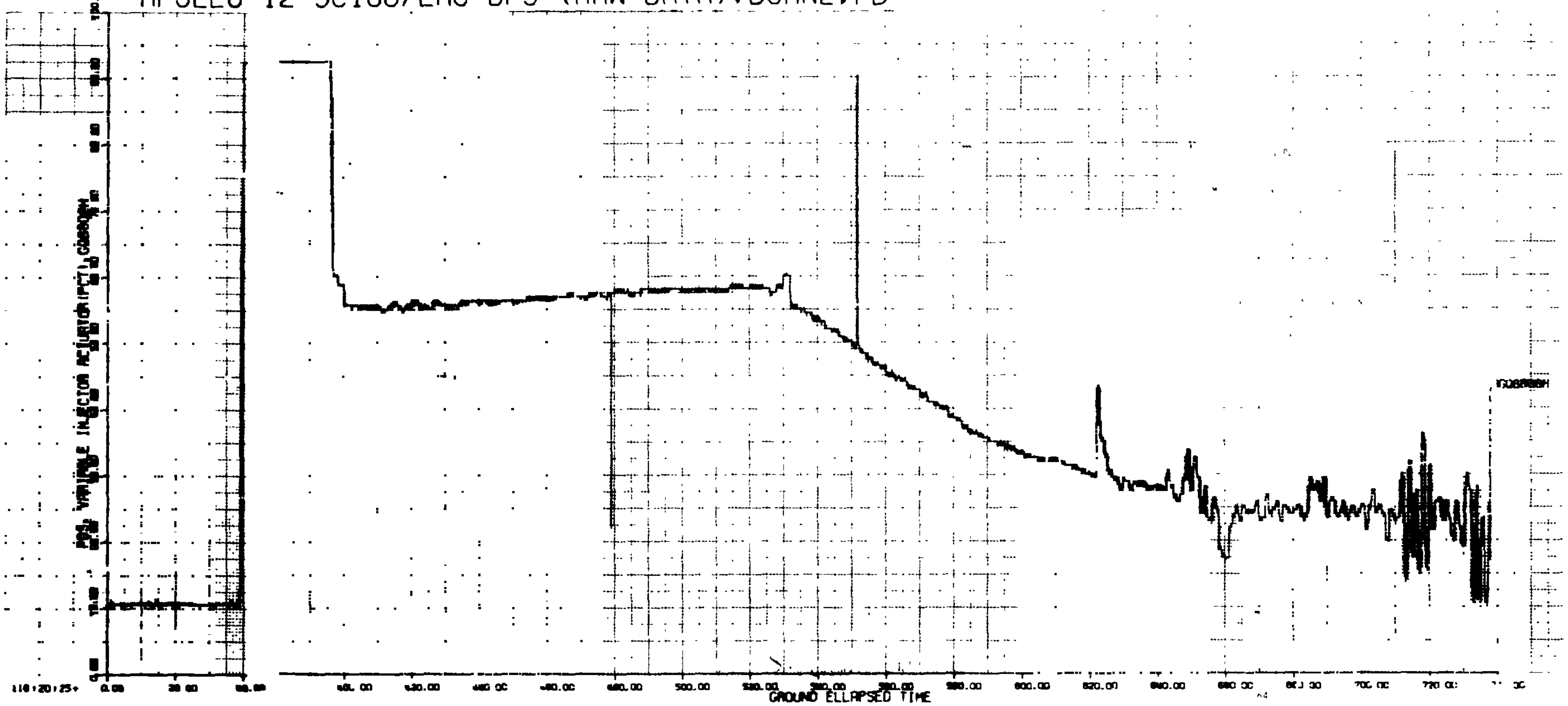
APOLLO 12 SC108/LM6-DPS-(RAW DATA).BURN2.PD





APOLLO 12 SC108/LM6-DPS-(RAW DATA).BURN2.PD
QUANTITY, FUEL TANK NO2(PCT),GQ3611Q
GROUND ELAPSED TIME

APOLLO 12 SC108/LM6-DPS-(RAW DATA),BURN2,PD



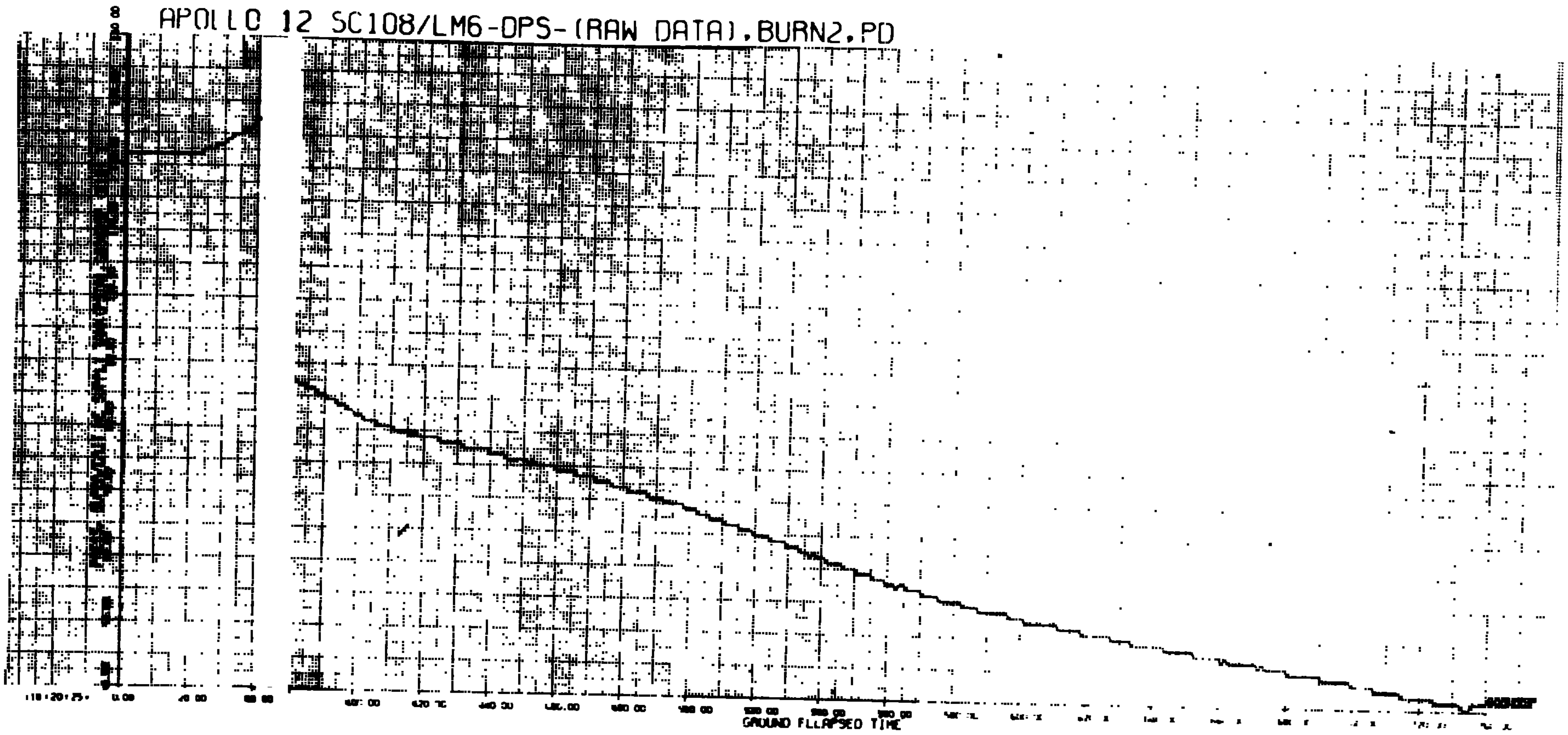
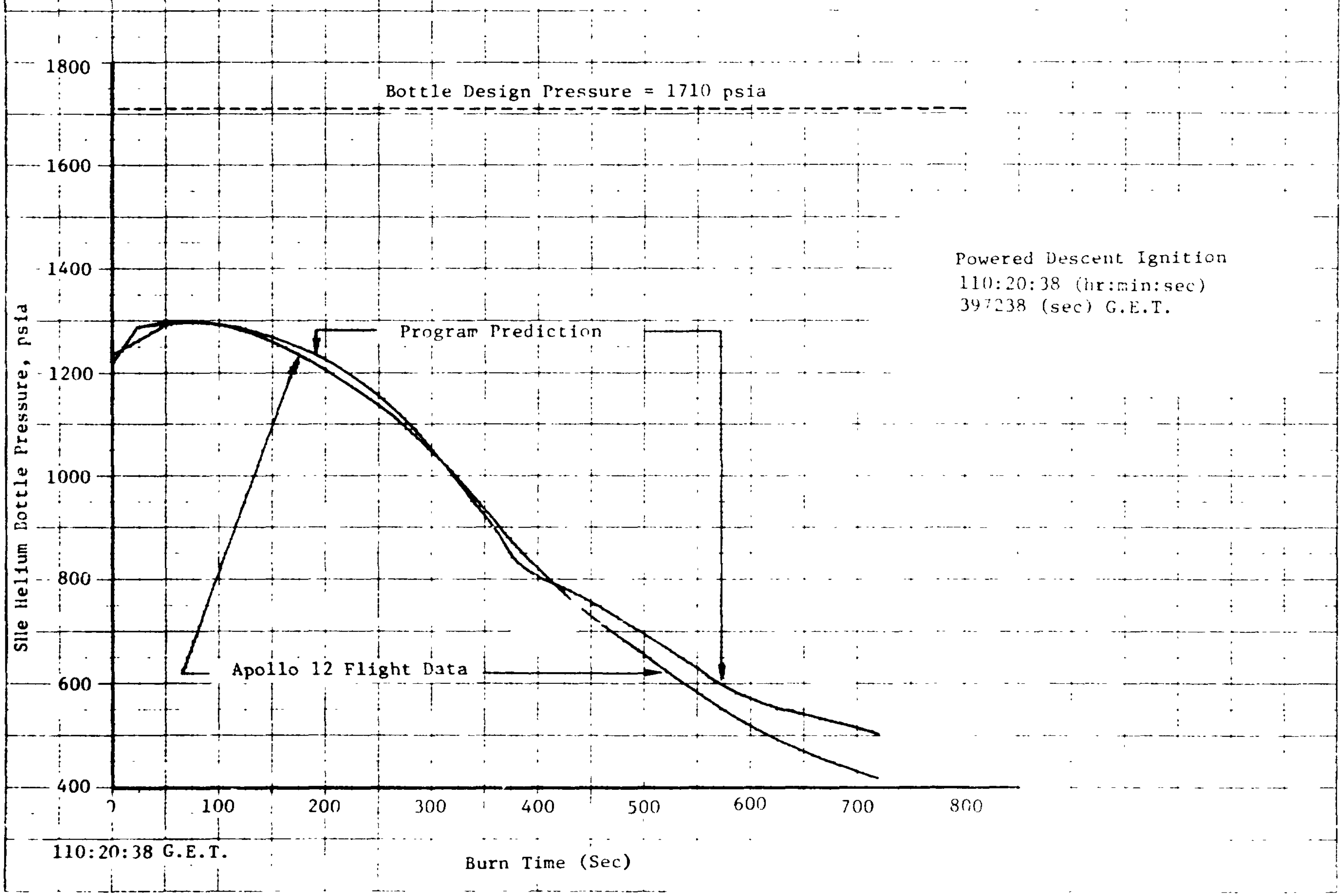
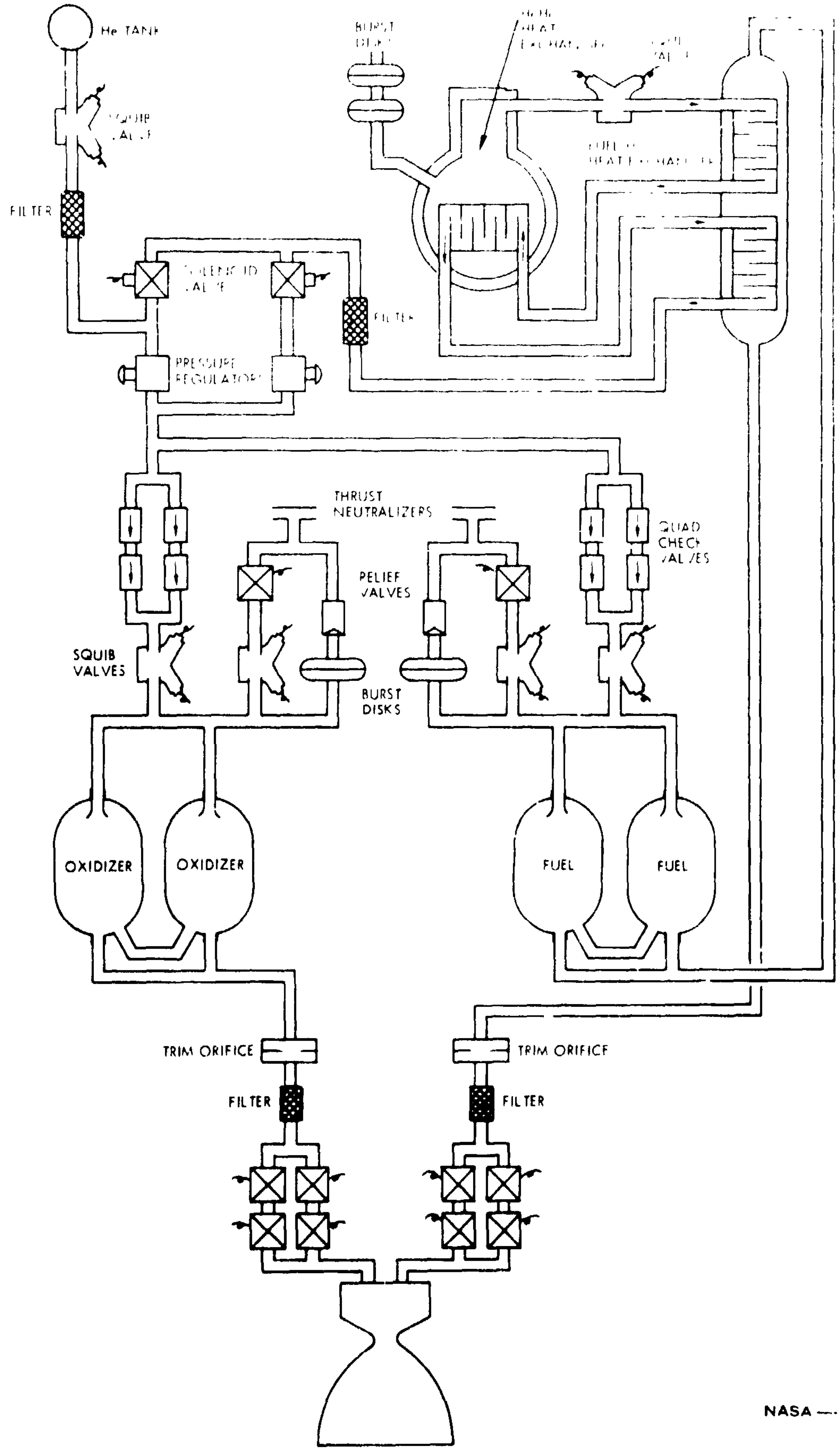

APOLLO 12 SC108/LM6-DPS-(RAW DATA).BURN2.PD
GROUND ELLAPSED TIME

Figure 29 SHe Tank Pressure (Powered Descent)



DESCENT STAGE SYSTEM SCHEMATIC



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