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SUPPLEMENT 5



NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

APOLLO 14 MISSION REPORT

SUPPLEMENT 5

DESCENT PROPULSION SYSTEM

FINAL FLIGHT EVALUATION

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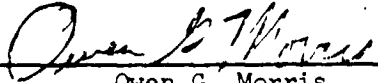
DESCENT PROPULSION SYSTEM

FINAL FLIGHT EVALUATION

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PROJECT TECHNICAL REPORT

APOLLO 14

LM-8

DESCENT PROPULSION SYSTEM  
FINAL FLIGHT EVALUATION

Follow-On Contract to NAS 9-8166

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## 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 14 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 14 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 14 Mission.

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

- (1) The performance values for the 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 is discussed in greater detail.
- (5) Engine transient performance and throttle response are discussed.
- (6) Estimated propellant consumption and residuals are revised.



## 2. SUMMARY

The performance of the LM-8 Descent Propulsion System during the Apollo 14 Mission was evaluated and found to be satisfactory. The average engine effective specific impulse was 0.1 second higher than predicted, but well within the predicted 1 $\sigma$  uncertainty. The engine performance corrected to standard inlet conditions for the FTP portion of the burn at 43 seconds after ignition was as follows: thrust, 9802 lbf; specific impulse, 304.1 sec; and propellant mixture ratio, 1.603. These values are +0.8, -0.06, and +0.3 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) The chamber pressure transducer had a large drift, exhibiting a maximum error of about 5 psi at approximately 150 sec after engine ignition. This problem has occurred during other flights but never to this magnitude. (Previous errors were less than 1 psi.)
- 2) The fuel and oxidizer interface pressure measurements appeared to be low during the entire flight. The discrepancy, which does not include the regulator outlet measurement bias, is assumed to be a measurement bias (-.92 and -2.44 psi for oxidizer and fuel, respectively).
- 3) The propellant quantity gaging system performance was not within specification. Testing at the White Sands facility prior to the flight showed that the design could not meet the specification performance; consequently, this out-of-specification performance was expected.

The low level sensor actuation time was as predicted, indicating that the new anti-slosh baffles installed for this flight performed well.

### 3. INTRODUCTION

The Apollo 14 Mission was the seventh flight and sixth manned flight of the Lunar Module (LM). The mission was the third successful lunar landing.

The space vehicle was launched from Kennedy Space Center (KSC) at 4:03:02 p.m. (EST) on January 31, 1971. The launch was delayed about 40 minutes due to unsatisfactory weather conditions. At 108:02:27 hours, the Descent Burn (PDI) was initiated and lasted about 765 sec. The burn was started at the minimum throttle setting and after approximately 27 sec, the thrust was manually increased to the fixed throttle position (FTP). The targeted time for throttle up was 26 sec. However, due to a faulty abort signal recorded prior to the PDI burn, a manual rather than an automatic throttle up was required. An automatic descent was maintained to approximately 643 seconds after throttle recovery, at which time the astronauts assumed semi-manual control of the final landing phase. The engine was commanded through a substantial number of throttle changes by the LM commander during this final landing phase. Lunar landing occurred at 108:15:09 AET ending the DPS mission duty cycle. Successful venting of the descent propellant tanks occurred about 1 min. after touchdown at about 108 hr. 16 min. AET. After a lunar stay of approximately 33-3/4 hours, the APS was ignited and the ascent stage of the LM lifted off and was inserted into lunar orbit about 12-1/2 minutes later. Descent data terminated at ascent liftoff.

The actual ignition and shutdown times for the DPS firing are 108:02:26 hours and 108:15:10.8 hours, respectively. The throttling profile for the DPS burn is shown in Figure 1.

The Apollo 14 Mission utilized LM-8 which was equipped with DPS engine S/N 1032. The engine and feed system characteristics are sent in Table 1.

The DPS burn was preceded by a two jet +X Reaction Control System (RCS) ullage maneuver of 7 seconds 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. A reconstruction of the throttled portion of the Descent Burn was attempted, however, due to the rapid changes in the engine thrust often experienced during this portion of the burn, a detailed analysis was not possible. The performance analysis of the FTP region was accomplished by use of the Apollo Propulsion Analysis Program which utilizes a minimum variance technique to "best" correlate the available flight data. The program embodies error models for the various flight data that are used as inputs, and by iterative methods, arrives at estimates of the system performance history and propellant weights which "best" (minimum - variance sense) reconcile the data.

The reconstruction of the throttled portion was made using a simulation technique and hand adjusting various initial parameters to achieve a reasonable fit to the data.

##### 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 initial vehicle weight was obtained from Reference 2. The initial estimates of the propellant onboard at the beginning of the analyzed time segment were calculated from the loaded propellant weights. The damp weight was also adjusted for consumables such as RCS propellant, water, etc., used between ignition and the start of the analyzed time segment. During the Descent Burn approximately 120 lbm of consumables other than the DPS propellant were used. Of the amount, 90.1 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 .08 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 33 seconds after DPS ignition (FS-1) and included the flight time between 108:02:59 hours and 108:08:39 hours ground elapsed time. Engine throttledown to 60 percent occurred 8 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 3 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 the difference can be attributed to the difference between predicted and actual regulator outlet pressure (1 psia at the end of the FTP burn).

The inflight throat erosion agreed well with predicted values. At the end of the FTP portion of the burn, the inflight throat erosion was within 1% of the predicted value of 9.07%.

### Critique of Analysis Results

Figures 3 through 10 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, and chamber pressure. The chamber pressure plot indicated how badly the chamber pressure measurement behaved during the burn. Because of this, chamber pressure was not used in the PAP program as a measurement.

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

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 GQ 3018P and GQ 3025P. The pressure indicated by GQ 3025P was about 2 psi lower than that from GQ 3018P. Based on earlier analyses and preflight tests, the data from GQ 3018P was used for the analysis. It should be noted that tests made at KSC several weeks prior to launch on the helium regulator

indicated about a 1 psi difference between the two measurements. Figure 11 shows the helium regulator pressure used to drive the program. Its shape was dictated by the interface pressure data which, although they were biased, agreed well with each other as to shape. Analysis indicated that the helium regulator measurement GQ 3018P was biased by about 1 psi.

The inflight chamber pressure (Figure 10) could not be used due to a large drift as previously stated.

The filtered value of the fuel interface pressure (GQ 4111P) was severely biased by -2.1 psi, although this is within the instrument accuracy. The oxidizer interface pressure was also biased by -.9 psi.

During the throttled portion of the burn, intermittent data dropout occurred from about 530 seconds to 580 seconds after ignition. As previously stated, the throttle region is driven by the actual commanded thrust (GH1331V). Since no attempt was made to smooth the data during the throttled portion of the burn, other than to eliminate "wild" points, a poor acceleration match during this time was obtained, with the calculated values higher than the measured values.

#### Comparison with Preflight Performance Predictions

Prior to the Apollo 14 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 11.

### 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	222.0
Fuel interface pressure, psia	222.0
Oxidizer interface temperature, °F	70.0
Fuel interface temperature, °F	70.0
Thrust acceleration, lbf/lbm	1.0
Throat area, in <sup>2</sup>	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	Ground Test	Flight
Data Source	Engine Prediction Characterization	Analysis Results
Thrust, lbf	9732	9802
Specific Impulse, sec	304.3	304.1
Mixture Ratio	1.598	1.603

data shows the flight data to be 0.8% more, 0.06% less, and 0.3% more for thrust, specific impulse and mixture ratio, respectively. These differences are 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 (GR 3018P) 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 a minimum variance match is not performed.

Based on the FTP analysis, it was determined that a -1 psia correction should be made to the regulator outlet pressure (GQ3018P). For the simulation, the initial values of throat erosion 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 a rate of 0.22 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 + 374 seconds) and continued for 388 seconds. This includes all of the powered descent burn after throttle down and includes the flight time between 108:08:48 hours to 108:15:09 hours.

Typical values of the simulation results are presented in Table 5. Since throttle recovery occurred 10 seconds earlier than predicted due to a high thrust at FTP, a comparison between the original preflight and flight data is not realistic. This is due to trajectory differences which would force the guidance system to command difference thrust levels than predicted.

However, another "preflight" was made assuming the high thrust level at FTP. This allows for similar commanded thrusts during the throttle region, and so these preflight values appear in Table 5 for comparison with the flight data.

Figures 12 through 14 present plots comparing the modified preflight predicted and the analysis program simulated values of throttle command percent, mixture ratio, and specific impulse. Also shown is the original preflight. The difference in the profiles between the original preflight and the analysis simulation is due primarily to the trajectory differences caused by the high thrust at FTP.

Figures 15 through 26 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). The vehicle effective specific impulse was computed based on spacecraft weight reduction due to both DPS propellant consumption and non-DPS consumables (approximately 0.08 lbm/sec during FTP and 0.22 lbm/sec during throttled operation). The engine effective specific impulse was calculated considering only weight reduction due to DPS propellant usage. Contributions from RCS activity are not included.

	Propellant Consumption (lbm)		Average MR (O/F)	Vehicle Effective Isp(sec)	Engine Effective Isp(sec)
	Oxidizer	Fuel			
Preflight Prediction	10758	6729	1.601	300.4	302.5
Analysis Program	10809	6794	1.591	299.8	302.6
Gaging System*	10742	6731	1.596	300.4	303.2

\*Based on gaging system and  $\Delta V$  measurements

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  $\pm 85$  lbm and  $\pm 53$  lbm (3 $\sigma$ ) for oxidizer and fuel, respectively. The uncertainties in mixture ratio and effective specific impulse resulting from these uncertainties are  $\pm 0.016$  and  $\pm 1.59$  respectively. Both the predicted and analysis program results are within these uncertainties.

<sup>1</sup> 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 7017 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, 6997 ft/sec, includes the contribution of both the DPS engine and RCS activity. The simulation was in error by 20 ft/sec due to the previously mentioned data dropout in the commanded thrust data (G41331V). The propellant consumed as calculated in the simulation, was corrected to account for the error induced by the data dropout. The uncertainty in effective specific impulse due to measured propellant usage and velocity gain uncertainties is  $\pm 1.2$  seconds. The engine effective specific impulse for both the prediction and analysis are within this uncertainty. 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, rather than from gaging system data above.

Both the analysis results and measured results are within the prediction 3- $\sigma$  uncertainties of  $\pm 5.91$  sec and  $\pm 0.0225$  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

At ignition of the descent engine, all propellant gages should be reading off scale. However, the fuel tank 2 (FU2) gage never did read off scale, while the FU1 gage intermittently dropped slightly below the maximum reading of 95%. The oxidizer gages initially performed as expected.

All gages were indicating propellant consumption by about 50 seconds after ignition. At that time the gages were reading 93.5%, 95.0%, 91.5%, and 90.5% for OX1, OX2, FU1 and FU2, respectively. The reconstruction analysis indicates that the fuel gages were out of specification limits during the first 140 seconds of the analysis but were within the expected range as determined by the WSTF tests. Therefore, they were not included in the analysis until 140 seconds when they were respectively 0.6% and -0.8% for FU1 and FU2 between the calculated and measured values. The oxidizer difference was +0.4% and +0.4% for OX1 and OX2, respectively. Throughout the burn, a difference of from 1 to 3% exists in the liquid level measurement between the two oxidizer tanks. 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. At the end of the FTP portion, a predicted difference of 4.0% versus a measured difference of 3.4% was observed. At touchdown, agreement between the average reading of both oxidizer and both fuel tanks were within 0.1% for oxidizer and 0.3% for the fuel. The oxidizer gages indicated a difference of 2.7% between the two tanks as compared to a predicted value of 3%. At the end of the burn the propellant gages were reading approximately 6.1%, 3.3%, 5.1%, and 3.4% for OX1, OX2, FU1 and FU2, respectively.

The expected accuracies for the gaging system, based on tests conducted at WSTF (Reference 4) 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*
100-50%	2.7%	100-60%	3.5%
50-25%	1.0%	60-20%	2.0%
25-8 %	0.5%	20-0 %	1.0%
8-0 %	1.0%	-	-

\* Percent of Full Tank

The specification limit of the PQGS is  $\pm 1\%$  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  $\pm 0.5\%$  of full tank capacity. However, the WSTF tests indicate that these specifications cannot be met.

In the analysis of FTP, propellant was transferred from OX2 to OX1 at a rate of 0.67 lbm/sec with an initial unbalance of 11.6 lbm. No fuel transfer was simulated. Table 5 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 probe based on WSTF results. At engine shutdown, the quantities of propellants remaining in the tanks were computed to be 5.6%, 2.6%, 3.4%, and 3.4% for OX1, OX2, FU1 and FU2, respectively. This is equivalent to remaining tanked quantities of 477 lbm of oxidizer and 244 lbm of fuel. Of these quantities, 374 lbm of oxidizer and 221 lbm of fuel are usable to depletion (including burning usable

propellants in feedlines). Using the computed propellant flowrates and the computed propellant residuals at engine shutdown, 60 seconds of hover time remained to fuel depletion. The measured propellant quantities remaining at engine shutdown, 439 lbm of usable oxidizer and 294 lbm of usable fuel, indicated a hover time of 77 seconds to oxidizer depletion.

The propellant low level sensor was triggered at 711 seconds into the burn. Gaging system data indicates that the sensor was triggered by the OX2 probe. At the time of signal, the measured readings were 8.4%, 5.5%, 7.5%, and 5.6% for OX1, OX2, FU1 and FU2 gages, respectively. Based on predicted times, the low level sensor was triggered on time. This indicated the new anti-slosh baffles installed in the descent propellant tanks operated as expected, and premature triggering of the low level sensor was prevented (was approximately 20 seconds premature on Apollo 12).

#### 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  $90.33 \text{ lbm/ft}^3$  and an average fuel density of  $56.52 \text{ lbm/ft}^3$  at a pressure of 240 psia and a temperature of 70° were determined from the samples. The tanked propellants were 7037.7 lbm of fuel and 11283.9 lbm of oxidizer. The total tanked propellants were 18321.6  $\pm 20$  lbm.

## 8. PRESSURIZATION SYSTEM EVALUATION

The DPS Supercritical Helium (SHe) system performed essentially as anticipated during the Apollo 14 Mission. The predicted and flight measured values for the pressure rise rates are shown below for comparison.

<u>Parameter</u>	<u>Preflight Prediction</u>	<u>Apollo 14 Mission</u>
On-Ground Pressure Rise Rate, psi/hr	8.15	8.0
Coast Pressure Rise Rate, psi/hr	6.65	6.2
Lunar Stay Pressure Rise Rate, psi/hr	4.9*	6.0

\* Apollo 12 Mission value during a 30-hour lunar stay.

Both the lunar rise rate and ground-to-coast shift in rise rate experienced during the Apollo 14 Mission were higher than predicted. The ground-to-coast shift in rise rate of 1.8 psi/hr, although higher than predicted, is close to the 1.5 psi/hr value. The lunar rise rate difference can be explained with the following information. Existing SHe system bottles are either doubly-evacuated or singly-evacuated types. The pressure rise rates of SHe bottles in which the annulus has been single-evacuated are higher than the doubly-evacuated annulus bottles and considerably higher when the bottles are less than full. Because Apollo 14 was the first successful flight that contained a single-evacuated type of SHe bottle and the preflight lunar rise rate shown above was obtained from Apollo 12 flight data for a doubly-evacuated bottle, the higher value of lunar rise rate for Apollo 14 appears reasonable.

An Apollo 14 postflight simulation of the SHe system generated with the latest version of the SHe system computer program is presented in Figure 27. The comparison with flight data shows that a close match to the SHe bottle pressure during the DPS engine burn was obtained.



## 9. ENGINE TRANSIENT ANALYSIS

The mission duty cycle of the Descent Propulsion System for Apollo 14 included one start at the minimum throttle setting, and one shutdown at approximately 29% throttle. Much throttling occurred during the Descent Burn, all of which was commanded by the LGC.

### Start and Shutdown Transients

Reference 5 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 0.55 seconds. The delay time compared favorably with the first burn delay observed during Apollo 13. Comparing the delay time for the PDI burn of Apollo 13 with other PDI burn delay times can not be done since all other PDI burns were second burns; that is, they all followed a DOI burn.

The start transient from FS-1 to 90% of the minimum steady-state throttle setting required 2.14 seconds with a start impulse of 710 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 0.13 seconds with an impulse of 160 lbf-sec.

The shutdown transient required 1.07 seconds from FS-2 to 10% of the steady-state throttle setting with an impulse of 976 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.

#### 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 (14% of full thrust) to FTP, which was executed 27 seconds into the burn, required approximately 1 second. The engine then remained at FTP for 352 seconds. The second command, from FTP to 59%, occurred 380 seconds after ignition and required approximately 0.5 second. This value compares favorably with similar maneuvers on previous flights. Little throttling was performed during the next 145 seconds. The LM Guidance Computer then commanded a ramping decrease in the throttle setting from 60% to 33% over 118 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 123 seconds of the burn, the LGC commanded approximately 60 throttle changes in the 28% to 35% range. The command time from one throttle setting to the next was generally less than 0.30 seconds. 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 14 Mission Report," May 1971.
2. SNA-8-D-027(III), Rev. 3., "CSM/LM Spacecraft Operational Data Book," Vol. III, Mass Properties, 31 April 1971.
3. TRW Report, "Apollo Mission H3/LM-8/DPS Preflight Performance Report," A. T. Avvenire, November 1970.
4. GAC LED-271-98, "PQGS Accuracy Study," I. Glick and S. Newman, 14 May 1969.
5. 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  
LM-8 DESCENT PROPULSION ENGINE AND  
FEED SYSTEM PHYSICAL CHARACTERISTICS

ENGINE

Engine Number	1032
Chamber Throat Area, in <sup>2</sup>	54.197 <sup>1</sup>
Nozzle Exit Area, in <sup>2</sup>	2569.7 <sup>3</sup>
Nozzle Expansion Ratio	47.4 <sup>3</sup>

FEED SYSTEM

Oxidizer Propellant Tanks, Total	
Ambient <sup>4</sup> Volume, Ft <sup>3</sup>	126.0 <sup>3</sup>
Fuel Propellant Tanks, Total	
Ambient Volume, Ft <sup>3</sup>	126.0 <sup>3</sup>
Oxidizer Tank to Interface	
Resistance, $\frac{\text{lbf-sec}^2}{\text{lbm-ft}^5}$	431.215 <sup>2</sup>
Fuel Tank to Interface	
Resistance, $\frac{\text{lbf-sec}^2}{\text{lbm-ft}^5}$	684.787 <sup>2</sup>

<sup>1</sup>TRW No. 01827-6260-R-00, TRW LEM Descent Engine Serial No. 1032 Acceptance Test Performance Report Paragraph 6.10, dated 7 October 1969.

<sup>2</sup>GAEC Cold Flow Tests.

<sup>3</sup>Approximate Values.

<sup>4</sup>14.7 PSIA and 70°F

C

TABLE 2

## FLIGHT DATA USED IN FTP STEADY-STATE ANALYSIS

<u>Measurement Number</u>	<u>Description</u>	<u>Range</u>	<u>Sample Rate Sample/Sec</u>
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
GQ3603Q	Quantity, Fuel Tank No. 1	0-95 percent	100
GQ3604Q	Quantity, Fuel Tank No. 2	0-95 percent	100
GQ4103Q	Quantity, Oxidizer Tank No. 1	0-95 percent	100
GQ4104Q	Quantity, Oxidizer Tank No. 2	0-95 percent	100
GQ3718T	Temperature, Fuel Bulk Tank No. 1	20-120°F	1
GQ3719T	Temperature, Fuel Bulk Tank No. 2	20-120°F	1
GQ4218T	Temperature, Oxidizer Bulk Tank No. 1	20-120°F	1
GQ4219T	Temperature, Oxidizer Bulk Tank No. 2	20-120°F	1
GQ4001X	PGNS Downlink Data	40 Bits	1/2

TABLE 3  
DESCENT PROPULSION SYSTEM STEADY-STATE FTP PERFORMANCE

PARAMETER	FS-1 + 43 SECONDS			FS-1 + 373 SECONDS		
	PREDICTED	MEASURED	CALCULATED	PREDICTED	MEASURED	CALCULATED
Regulator Outlet Pressure, psia	244.8	245.7	244.5	244.8	246.9	245.7
Oxidizer Interface Pressure, psia	223.9	223.1	223.9	222.8	223.1	223.9
Fuel Interface Pressure, psia	223.9	221.6	224.0	223.0	221.8	224.1
Engine Chamber Pressure, psia	104.5	105.8	104.9	99	100	97.3
Oxidizer Bulk Temperature, Tank No. 1, °F	68	68	---	68	68	---
Oxidizer Bulk Temperature, Tank No. 2, °F	68	68	---	68	68	---
Fuel Bulk Temperature, Tank No. 1, °F	68	68	---	68	68	---
Fuel Bulk Temperature, Tank No. 2, °F	68	68	---	68	68	---
DERIVED						
Oxidizer Flowrate, lbm/sec	19.81	---	19.9	20.2	---	20.4
Fuel Flowrate, lbm/sec	12.4	---	12.5	12.7	---	12.8
Propellant Mixture Ratio	1.597	---	1.603	1.596	---	1.60
Vacuum Specific Impulse, sec	304.4	---	304.2	301.1	---	301.2
Vacuum Thrust, lbf	9792	---	9861	9901	---	9990
Throat Erosion, %	-.61	---	-.63	9.07	---	8.23

TABLE 4

## DESCENT PROPULSION SYSTEM THROTTLED PERFORMANCE

PARAMETER	FS-1 + 396 Seconds		FS-1 + 606 Seconds			
	Predicted	Measured	Simulation	Predicted	Measured	Simulation
Regulator Outlet Pressure, psia	245	247	246	246	247	245
Oxidizer Interface Pressure, psia	236	234	236	240	239	239
Fuel Inter Interface Pressure, psia	236	232	236	240	239	239
Engine Chamber Pressure, psia	62	-	61	37	-	37
Oxidizer Bulk Temperature, Tank No. 1, °F	68	68	68	68	68	68
Oxidizer Bulk Temperature, Tank No. 2, °F	68	68	68	68	68	68
Fuel Bulk Temperature, Tank No. 1, °F	68	68	68	68	68	68
Fuel Bulk Temperature, Tank No. 2, °F	68	68	68	68	68	68
Throttle Command Voltage, VDC	0	8.40	8.40	68	5.57	5.57
DERIVED						
Oxidizer Flowrate, lbm/sec	12.5	-----	12.2	7.7	-----	7.6
Fuel Flowrate, lbm/sec	7.8	-----	7.6	4.8	-----	4.8
Propellant Mixture Ratio, O/F	1.603	-----	1.605	1.604	-----	1.598
Vacuum Specific Impulse, sec	303.4	-----	303.2	298.1	-----	298.2
Vacuum Thrust, lbm	6159	-----	6003	3726	-----	3697
Throat Erosion, %	9.25	-----	8.53	10.96	-----	10.46

TABLE 5  
DPS PROPELLANT QUANTITY GAGING SYSTEM PERFORMANCE

Parameter	Time (From Descent Burn Ignition) sec							
	43	173	243	343	445	541	645	745
Oxidizer Tank No. 1								
Measured Quantity, percent	94.6	73.3	61.5	43.3	29.9	19.6	12.4	6.8
Calculated Quantity, percent	95.0	72.9	61.0	43.8	30.2	19.7	12.1	6.5
Difference, percent	-0.4	0.4	0.5	-0.5	-0.3	0.2	0.3	0.3
Oxidizer Tank No. 2								
Measured Quantity, percent	-	72.3	59.4	40.7	27.7	16.6	9.2	4.1
Calculated Quantity, percent	95.3	71.9	59.2	41.1	27.2	16.7	9.1	3.5
Difference, percent	-	0.4	0.2	-0.4	0.5	-0.1	0.1	0.6
Fuel Tank No. 1								
Measured Quantity, percent	92.5	71.9	60.1	42.0	28.6	18.2	11.0	5.6
Calculated Quantity, percent	94.0	71.3	59.1	41.4	27.7	17.5	9.8	4.3
Difference, percent	-1.5	0.6	1.0	0.6	0.9	0.7	1.2	1.3
Fuel Tank No. 2								
Measured Quantity, percent	91.3	70.6	58.3	40.6	28.0	17.5	9.7	3.8
Calculated Quantity, percent	94.0	71.3	59.1	41.4	27.7	17.5	9.8	4.3
Difference, percent	-2.7	-0.7	-0.8	-0.8	0.3	0.0	-0.1	-0.5

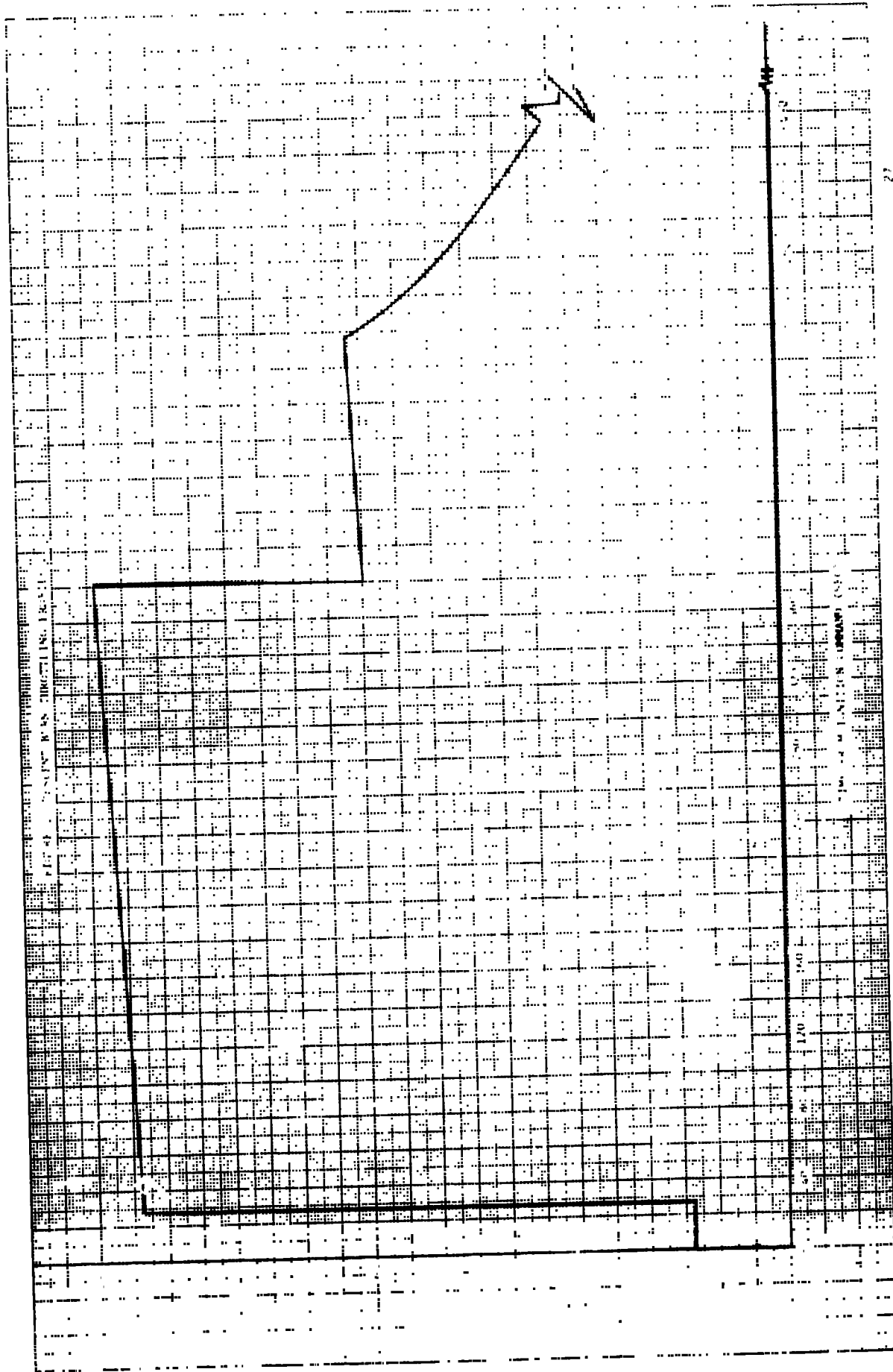


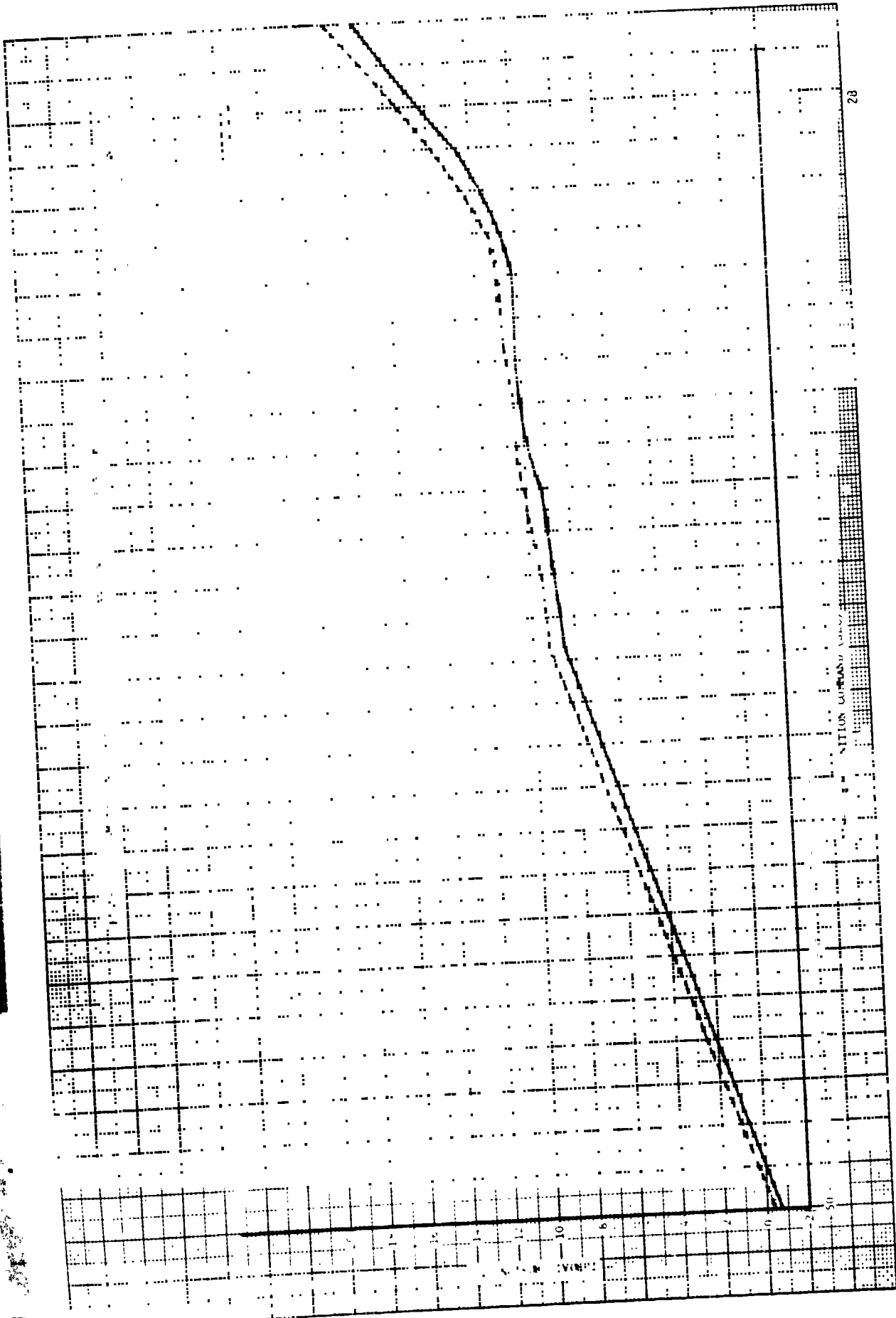
TABLE 6  
DPS START AND SHUTDOWN IMPULSE SUMMARY

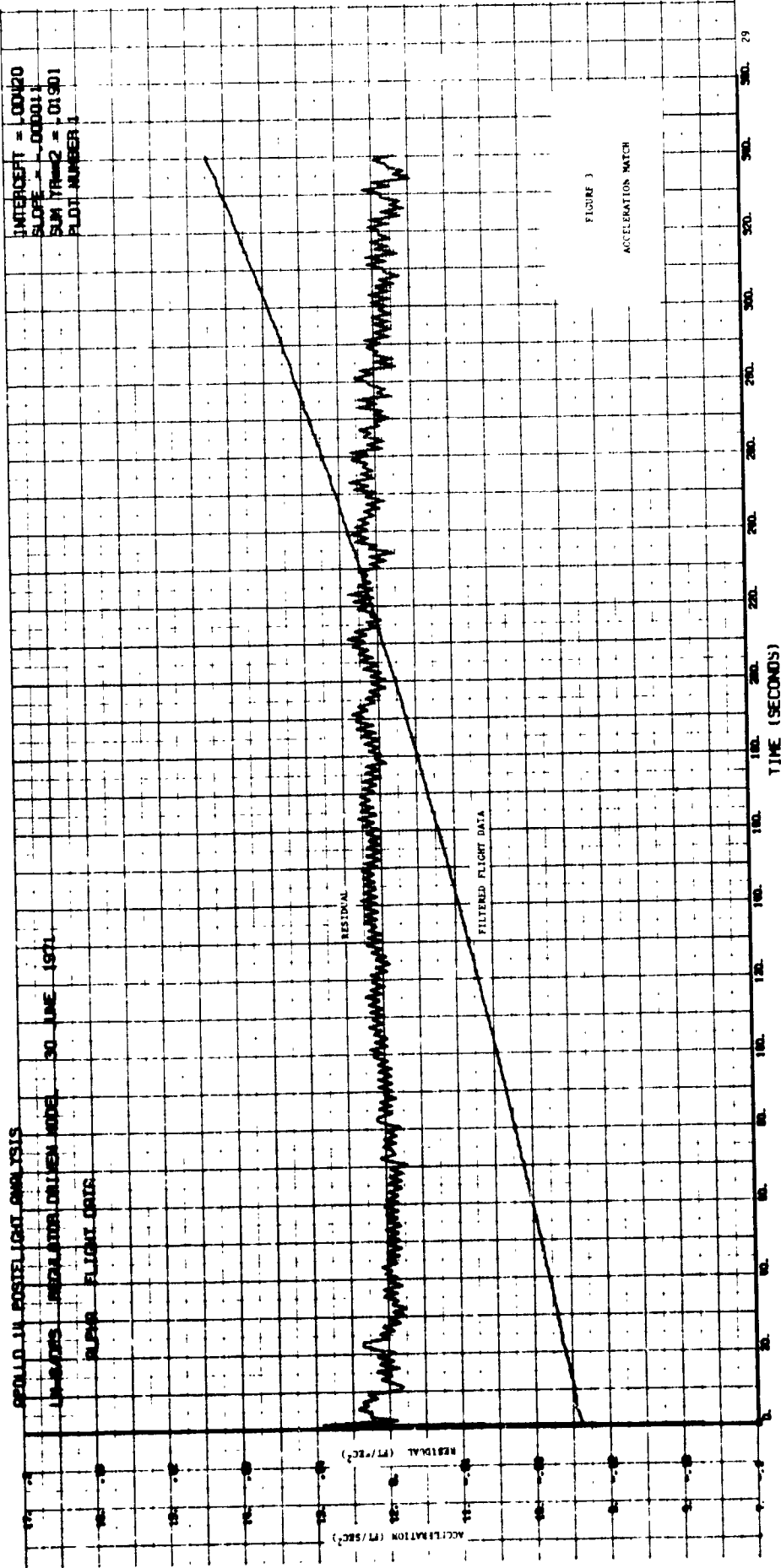
	Apollo 14 LM-8/DPS-1	Apollo 12 LM-6/DPS-2	Apollo 10 LM-4/DPS-2	Apollo 9 LM-3/DPS-1	Apollo 9 LM-3/DPS-2	Apollo 9 LM-3/DPS-3	Apollo 5 LM-1/DPS-2	Apollo 5 LM-1/DPS-3	SPECIFICATION LIMITS
STARTS									
Steady-State Throttle Position, Percent	13.1	16.2	13.1	12.7	12.7	12.7	12.4	12.4	
Total Vacuum Start Impulse (FS-1 to 90% steady-state), lbf-sec	710	591	728	805	1,295	950	834	574	
Start Time (FS-1 to 90% steady-state), sec	2.14	1.77	2.13	2.5 <sup>3</sup>	2.0	2.0 <sup>3</sup>	2.56	2.10	4.0
Coast Time from Prior Burn, Minutes	From Launch	55	72	From Launch	2640	111	131	0.5	
SHUTDOWNS									
Steady-State Throttle Position, Percent	27.0	23.4	FTP	40	40	12.7	FTP	FTP	
Total Vacuum Shutdown Impulse (FS-2 to 10% Steady-State), lbf-sec	976	1540	2041	--- <sup>2</sup>	1730	---	1727	1713	
Shutdown Time (FS-2 to 10% Steady-State), sec	1.23	2.06	0.34	1.1 <sup>1</sup>	1.1	1.1	--- <sup>2</sup>	2.30	0.25 <sup>4</sup>
Repeatability, lbf-sec							1734±7	1734±7	±100 <sup>4</sup>
Total Vacuum Shutdown Impulse (FS-2 to Zero Thrust) from Velocity Gain Data, lbf-sec	--- <sup>5</sup>	--- <sup>5</sup>	2948	1777	--- <sup>2</sup>	1040	2493	--- <sup>3</sup>	

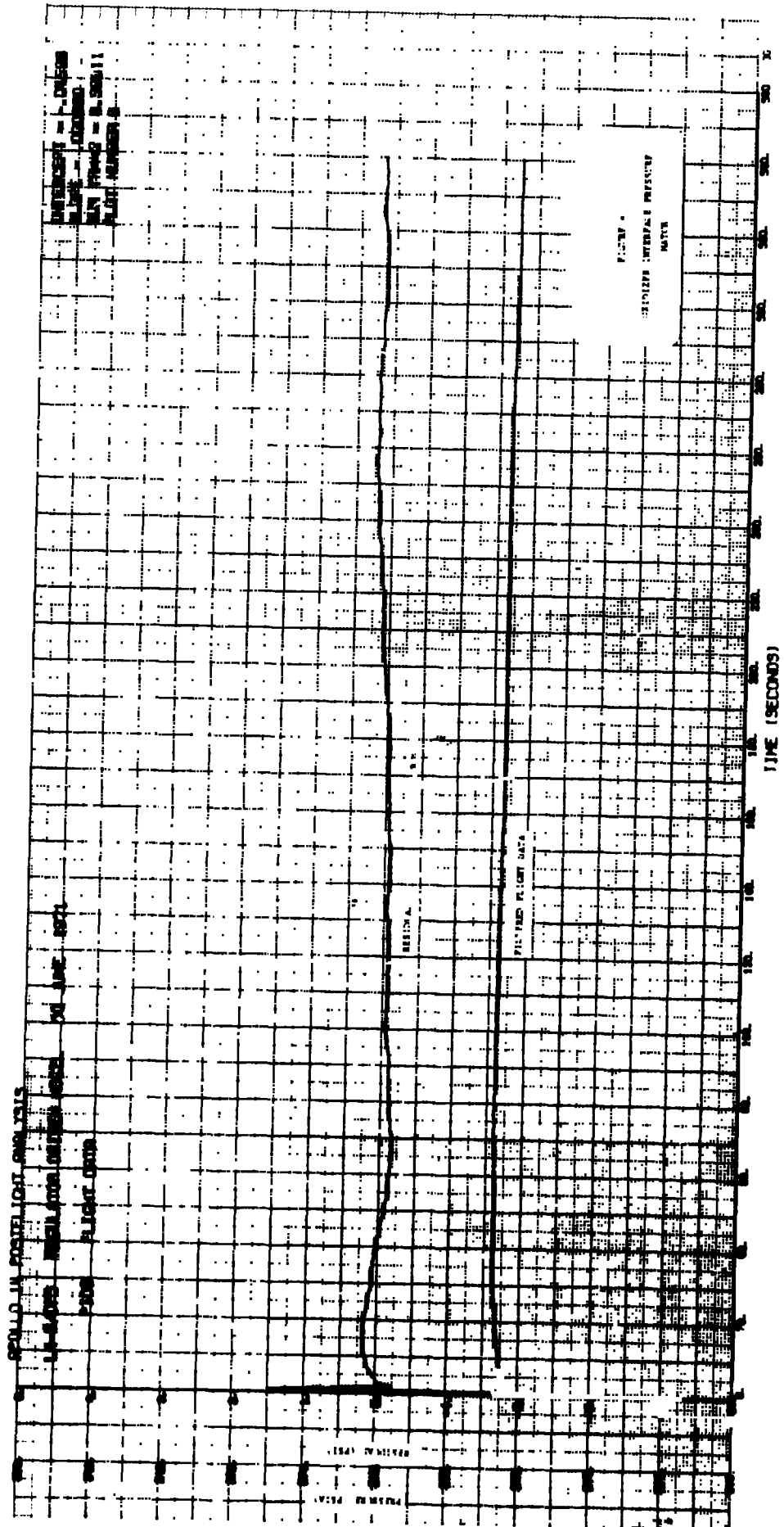
<sup>1</sup>Reference 5. <sup>3</sup>Unavailable due to APS "Fire-in-the-Hole" maneuver <sup>5</sup>Not applicable to lunar landing shutdown

<sup>2</sup>Data Unavailable. <sup>4</sup>Specification value for shutdowns performed from FTP only.









INSTRUMENT - CHROM  
 MODEL - 1000  
 MANUFACTURED - B. SMITH  
 PLANT NUMBER - 2

PULSE RATE  
 PULSE WIDTH

PULSED LIGHT DATA

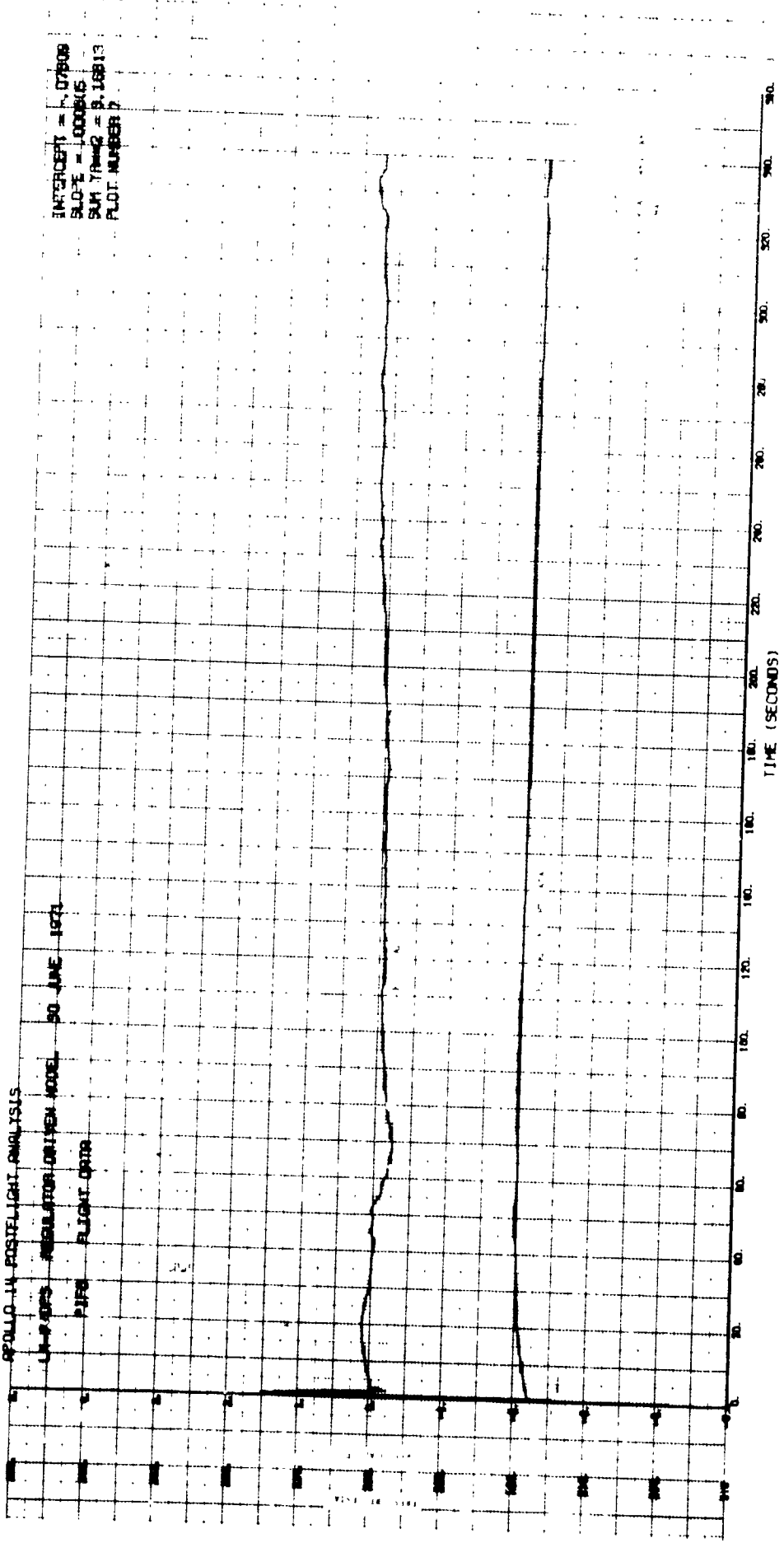
PULSE RATE  
 PULSE WIDTH

OPOLLO IN POSTFLIGHT ANALYSIS

LM-1000S REGULATORY DRIVEN MODEL 30 JUNE 1971

PIFB FLIGHT DATA

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SUM Y^2 = 9.18813  
PLOT NUMBER 7

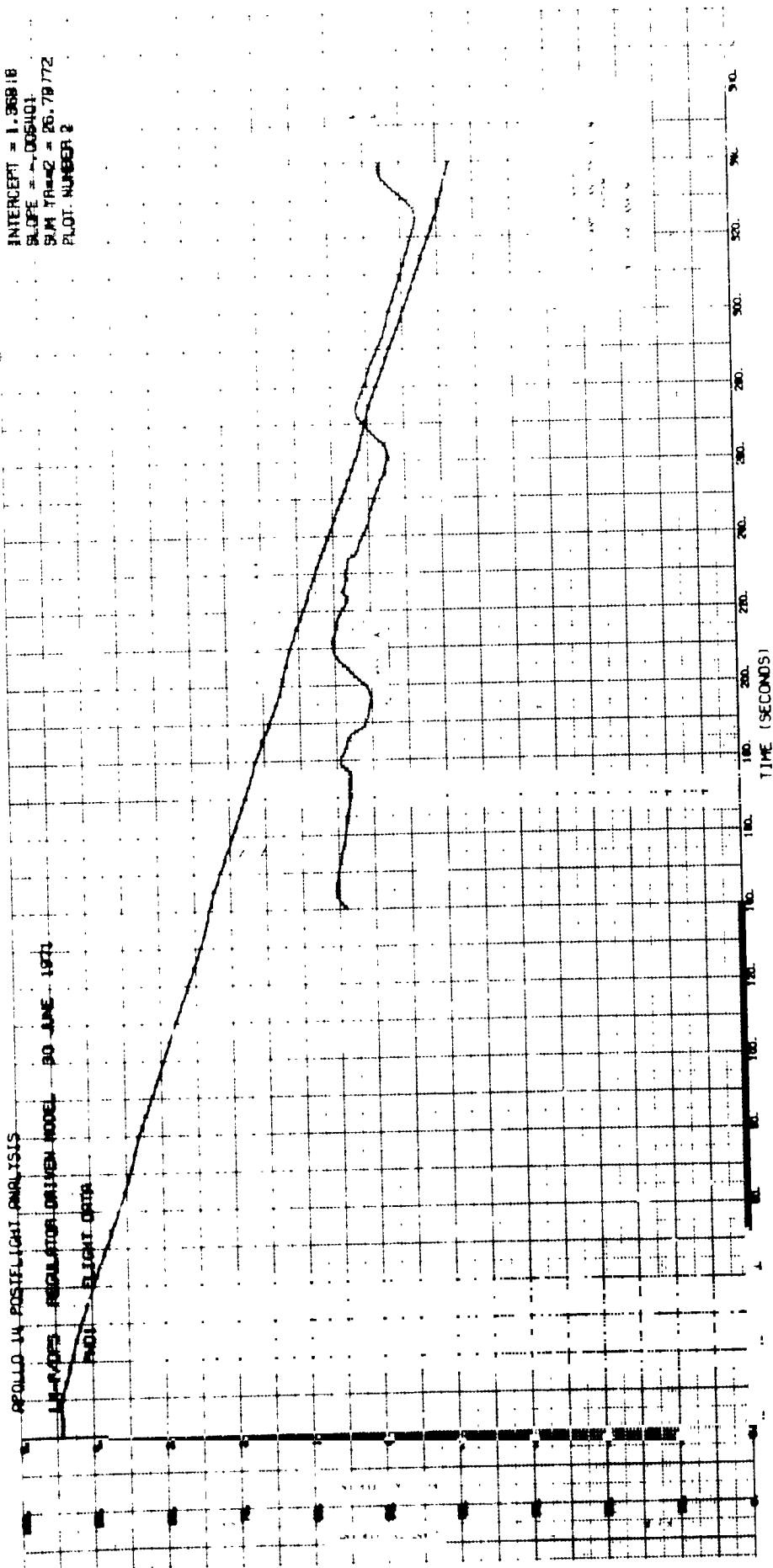


APOLLO 14 POSTFLIGHT ANALYSIS

LM-RV OPS REGULATOR DRIVEN MODEL 30 JUNE 1971

FM01 FLIGHT DATA

INTERCEPT = 1.369 18  
SLOPE = -.006401  
SUM TRM=2 = 26.79772  
PLOT NUMBER 2

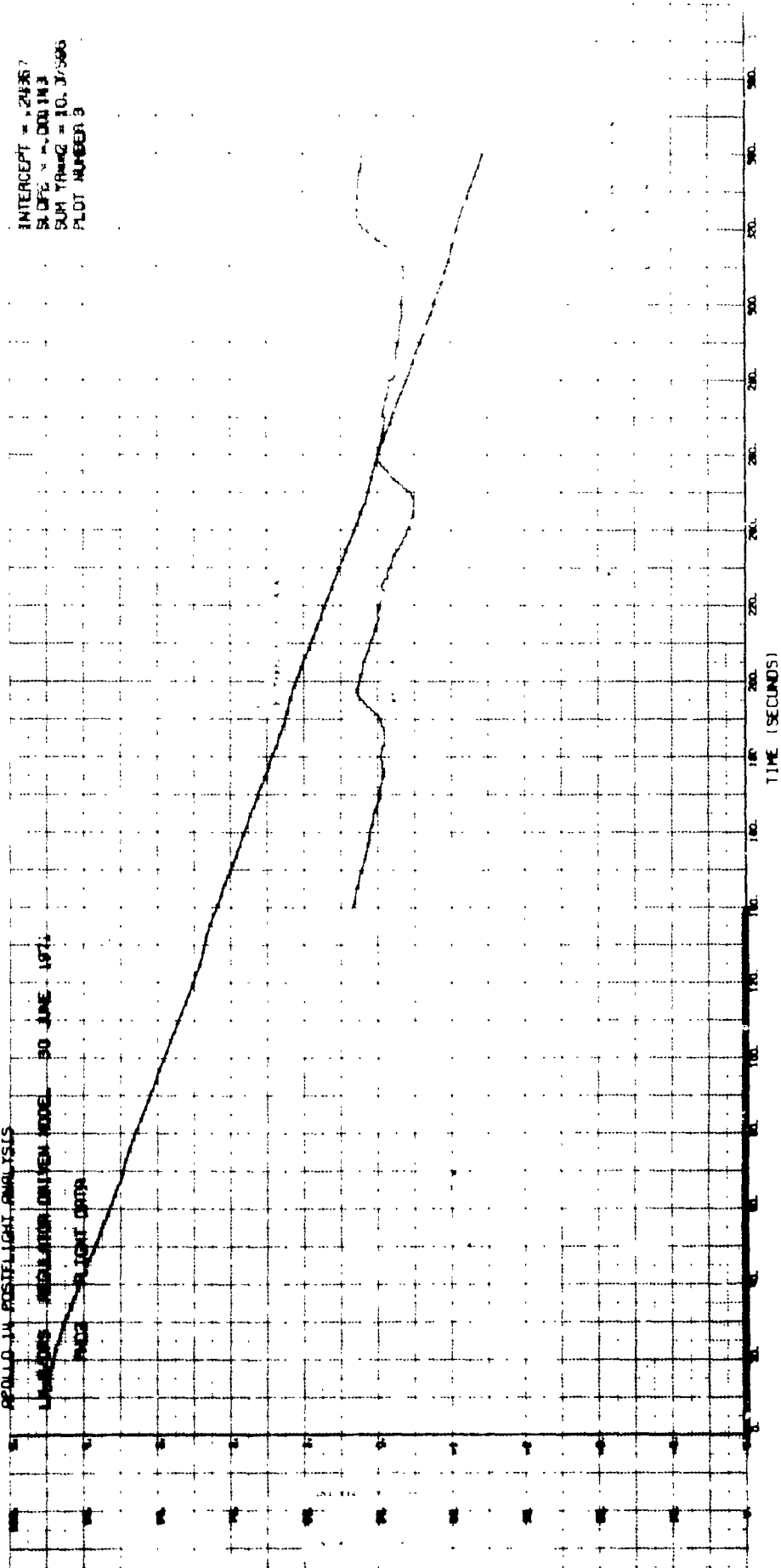


PELLO III POSTFLIGHT ANALYSIS

LAUNCHING - MISSILE DRIVEN MODEL 30 JUNE 1971

PLAC - FLIGHT DATA

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SL OFF = 000143  
SUM YRANG = 10.37566  
PLOT NUMBER 3



TIME (SECONDS)

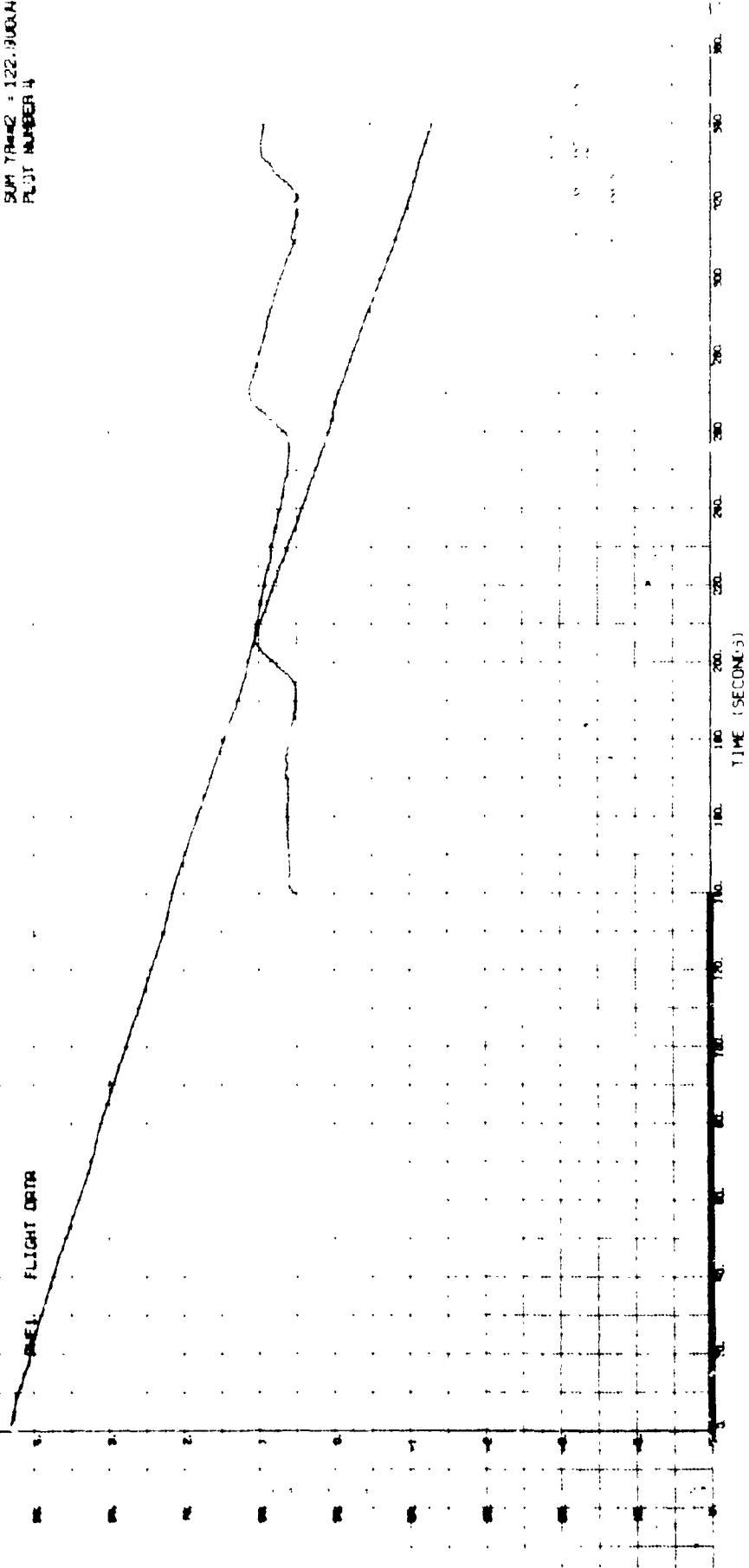


MODEL 14 FLIGHT ANALYSIS

UN-ADPS REGULATOR-DRIVEN MODEL 30 JUNE 1971

MODEL FLIGHT DATA

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SQUARED = 122.9064  
PLOT NUMBER 4

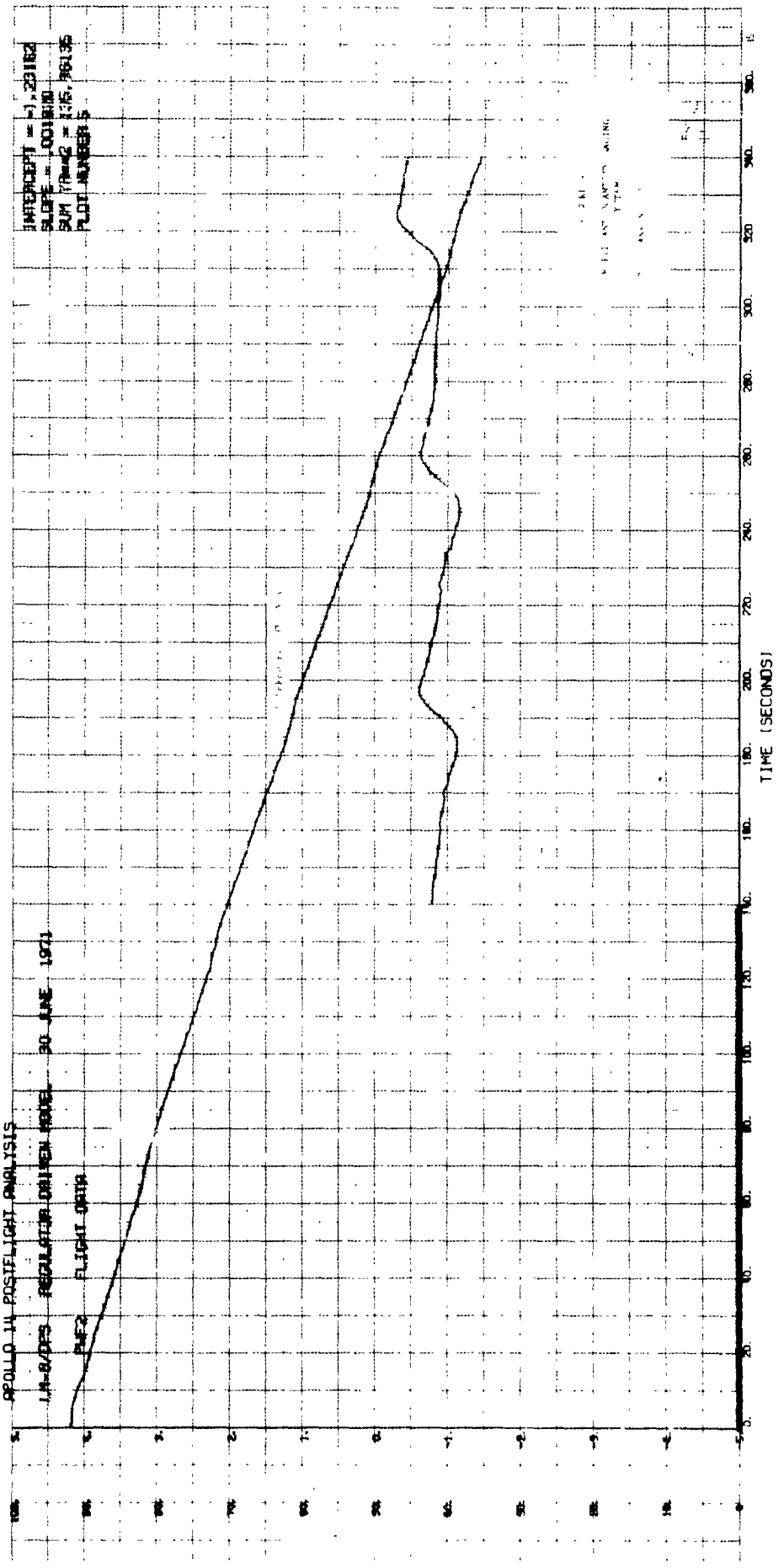


APOLLO 14 POSTFLIGHT ANALYSIS

14-0473 REGULATOR DRIVEN NOISE 30 JUNE 1971

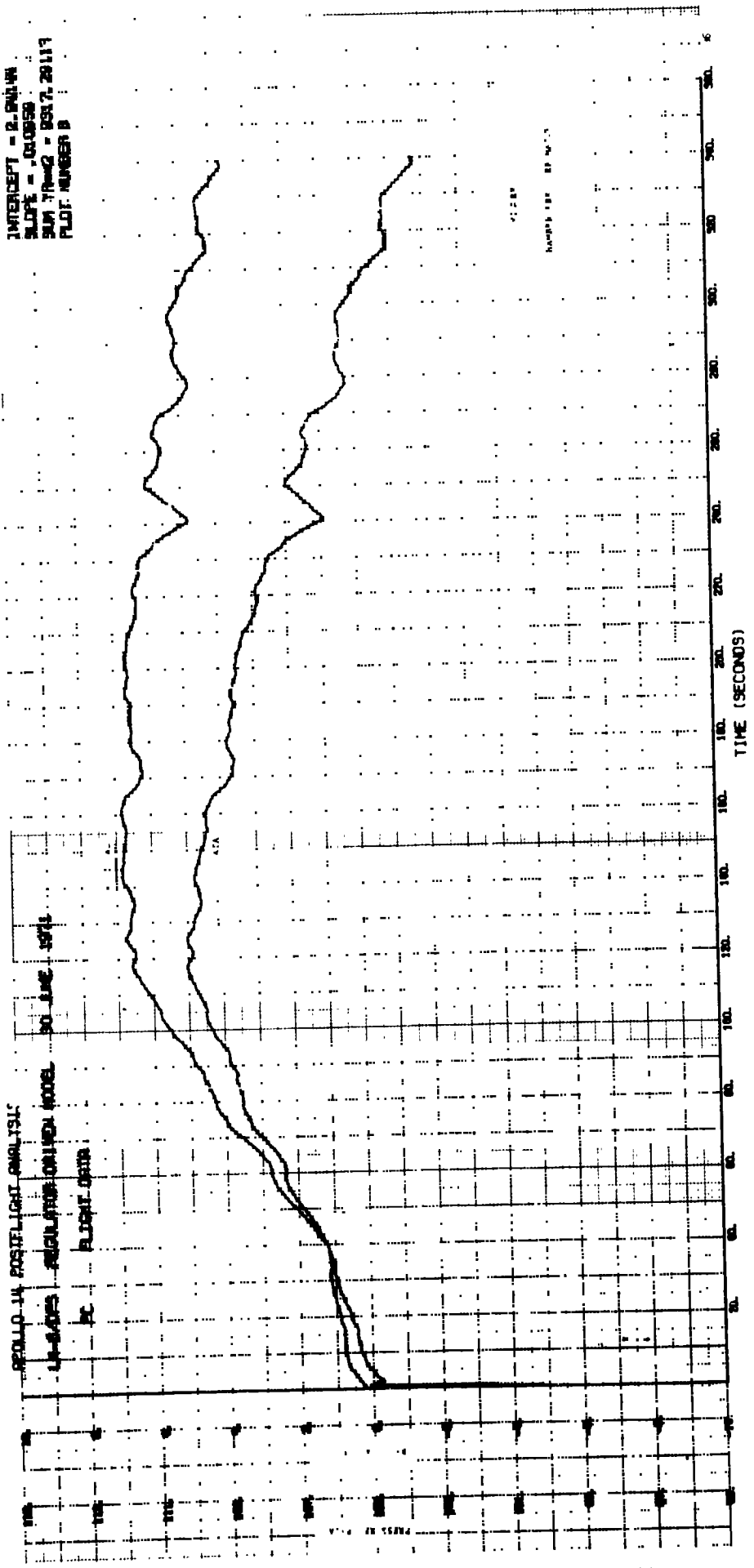
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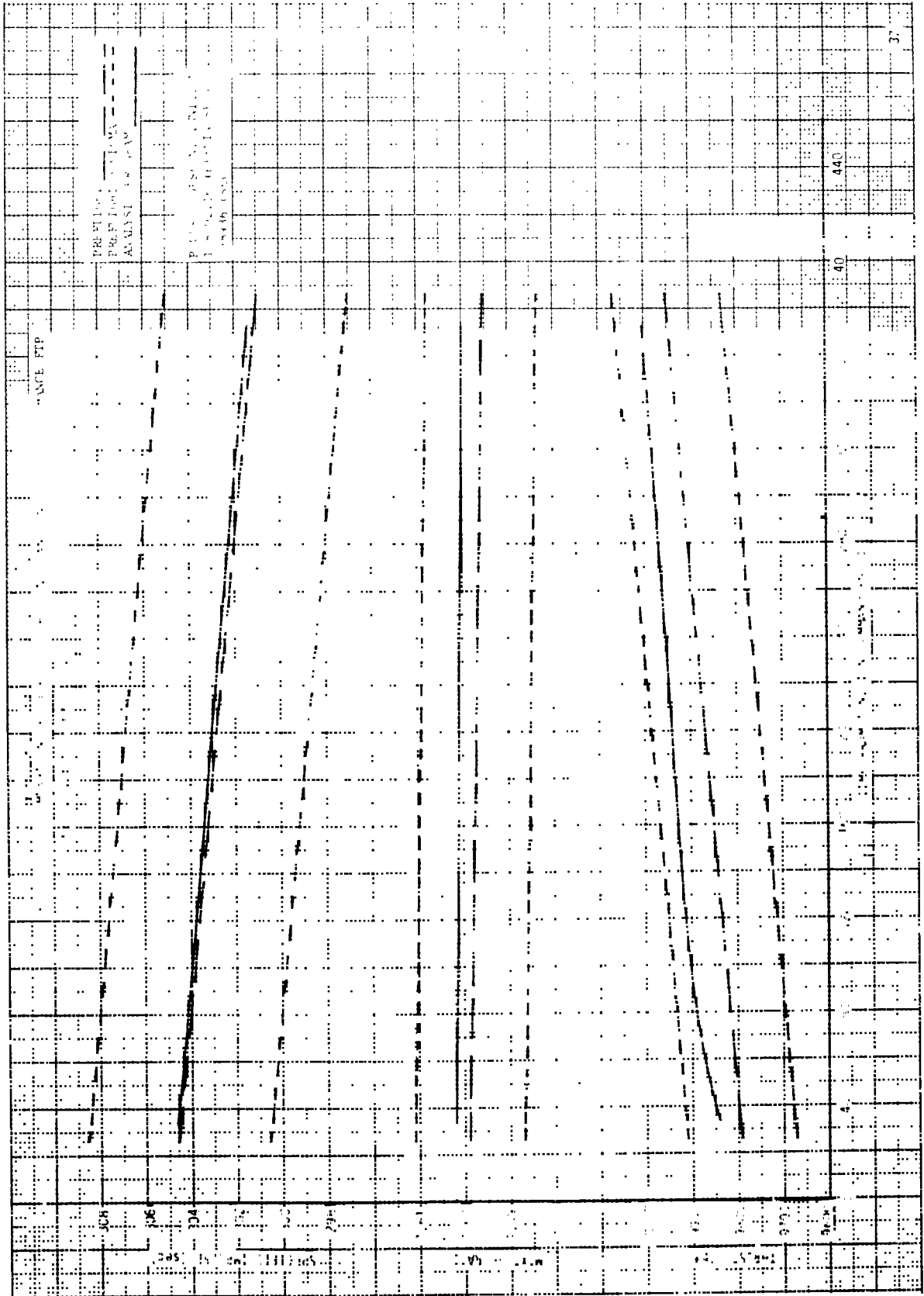
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PLOT NUMBER 5

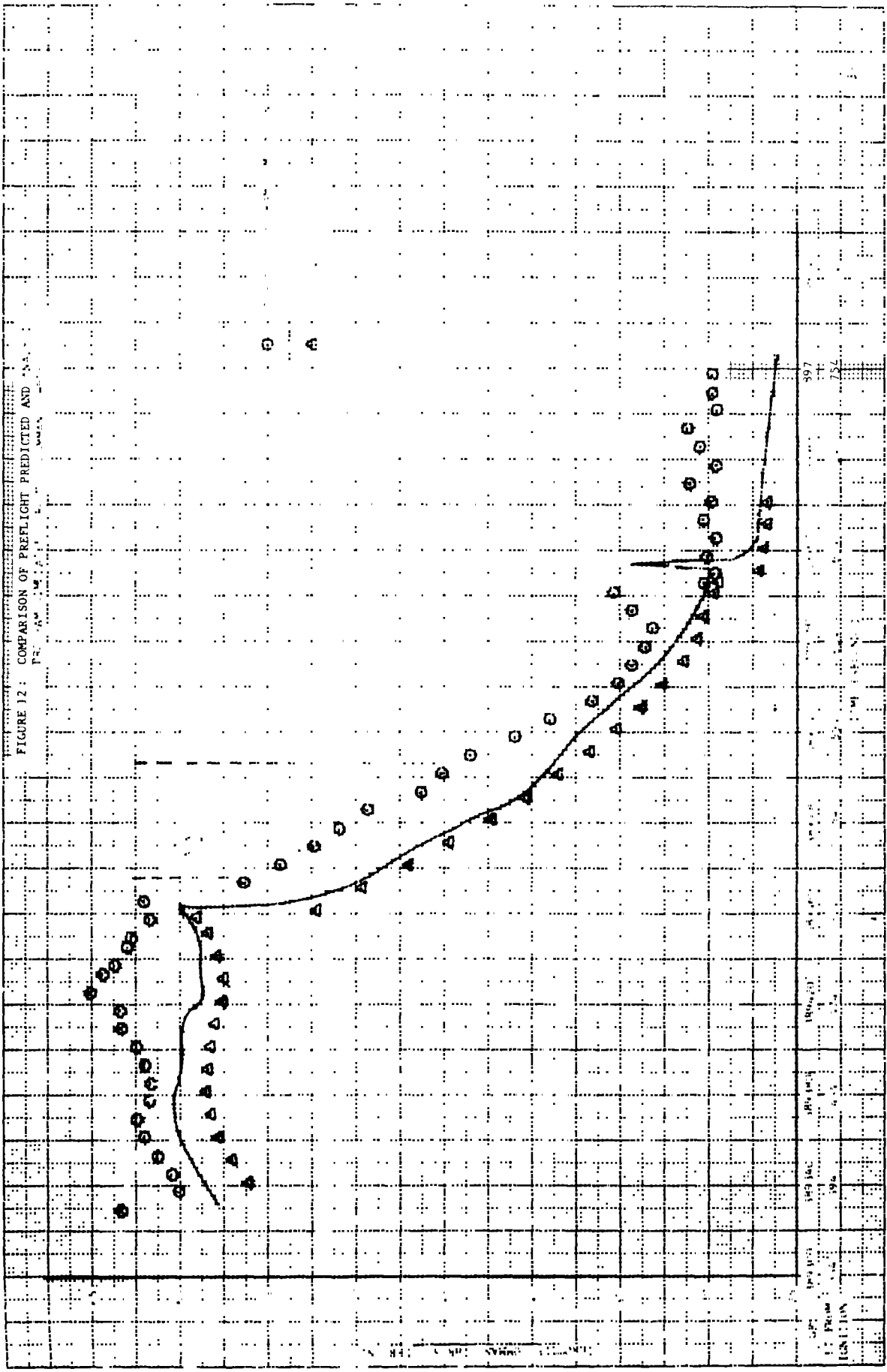


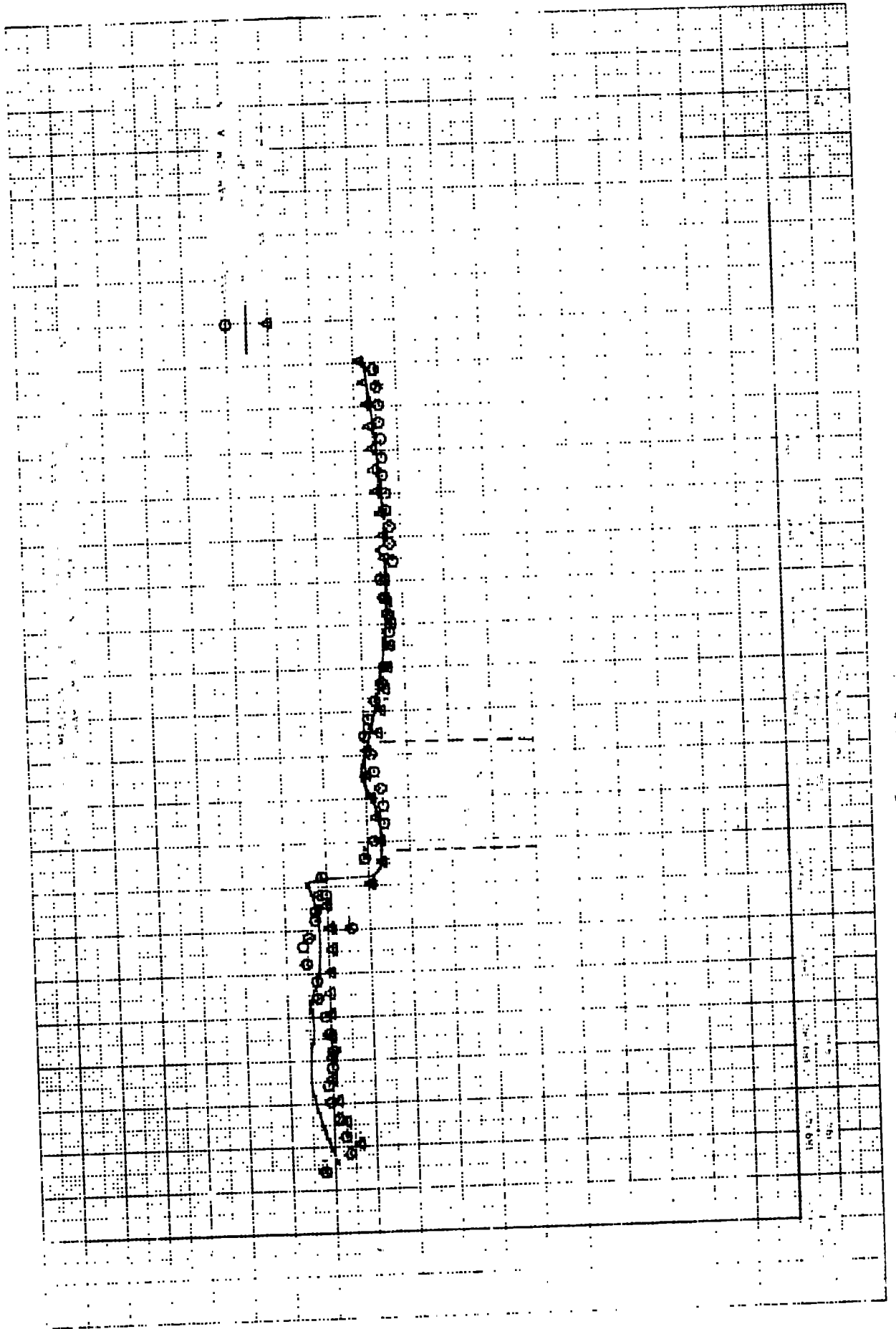
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PLOT NUMBER 8

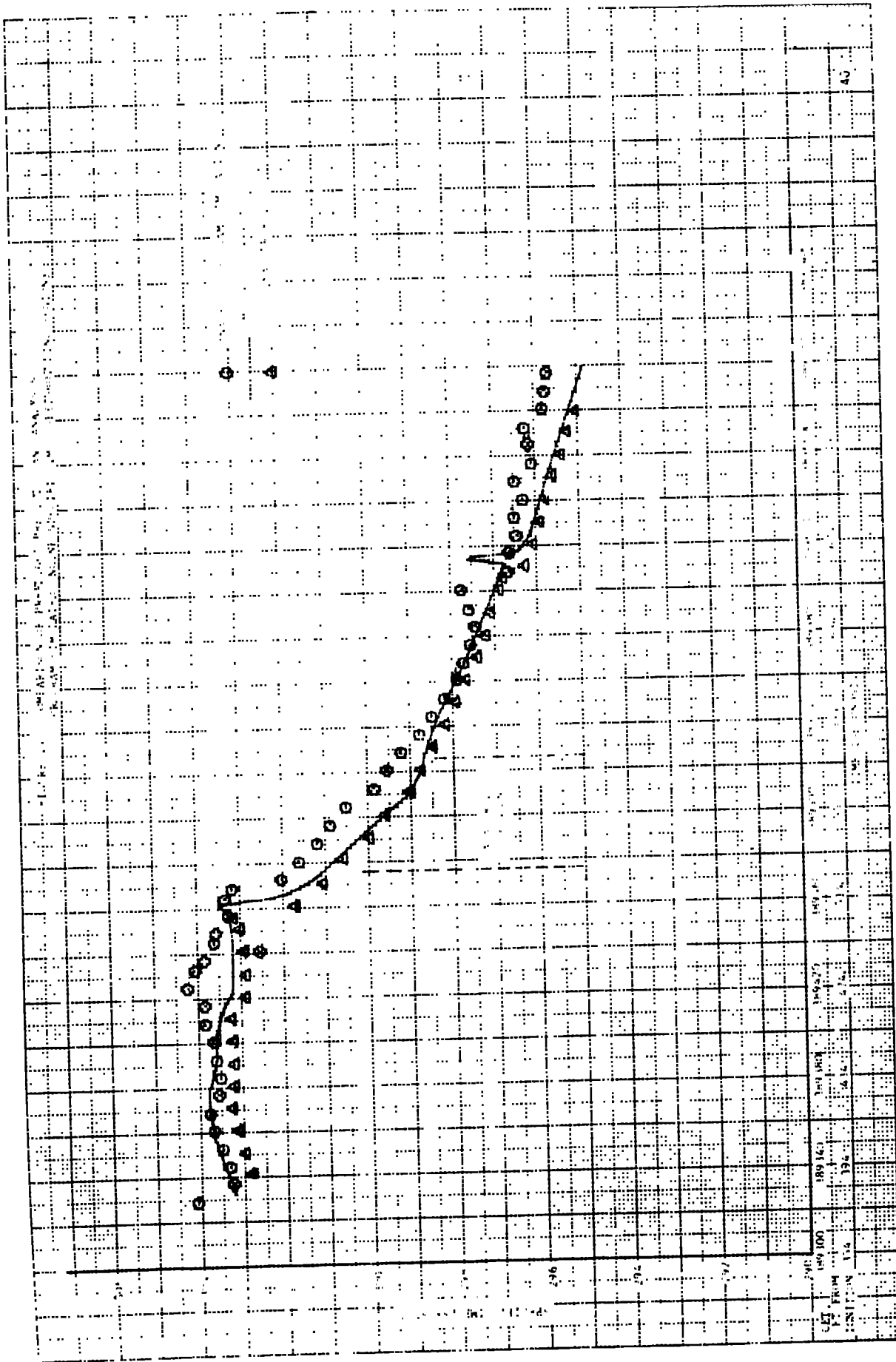
REGULATED POSITIVE ANALYSIS  
1-3-68 REGULATED POSITIVE MODEL 30 JUNE 1971  
PC FLIGHT DATA



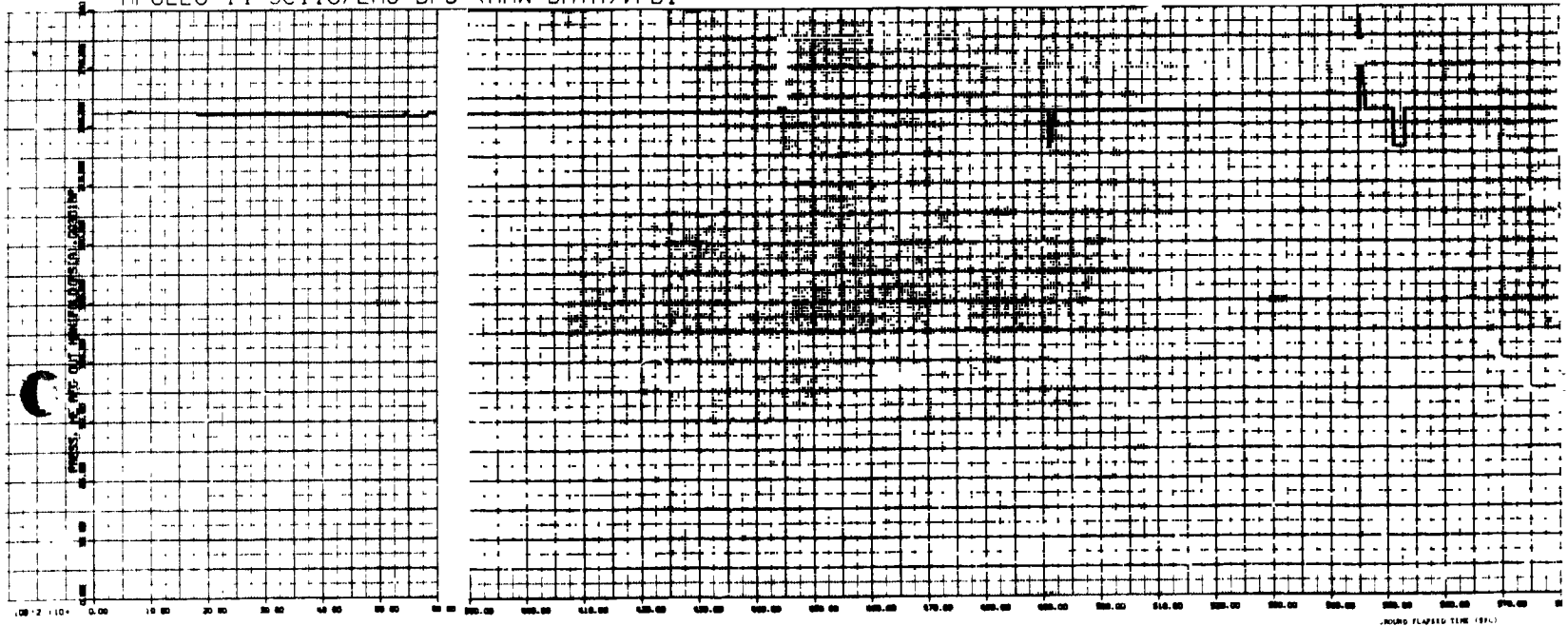








APOLLO 14 SC110/LM8-DPS-(RAW DATA).PD1



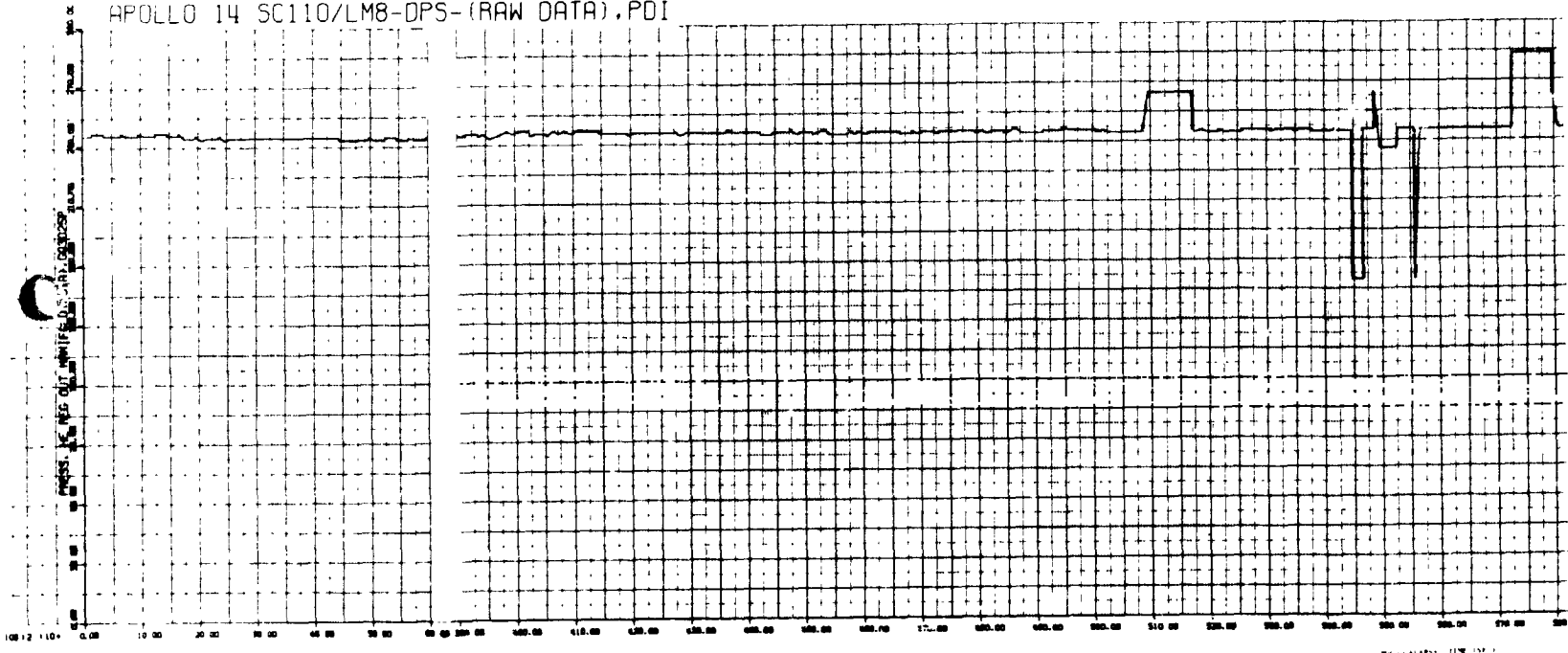
FOLDOUT FRAME

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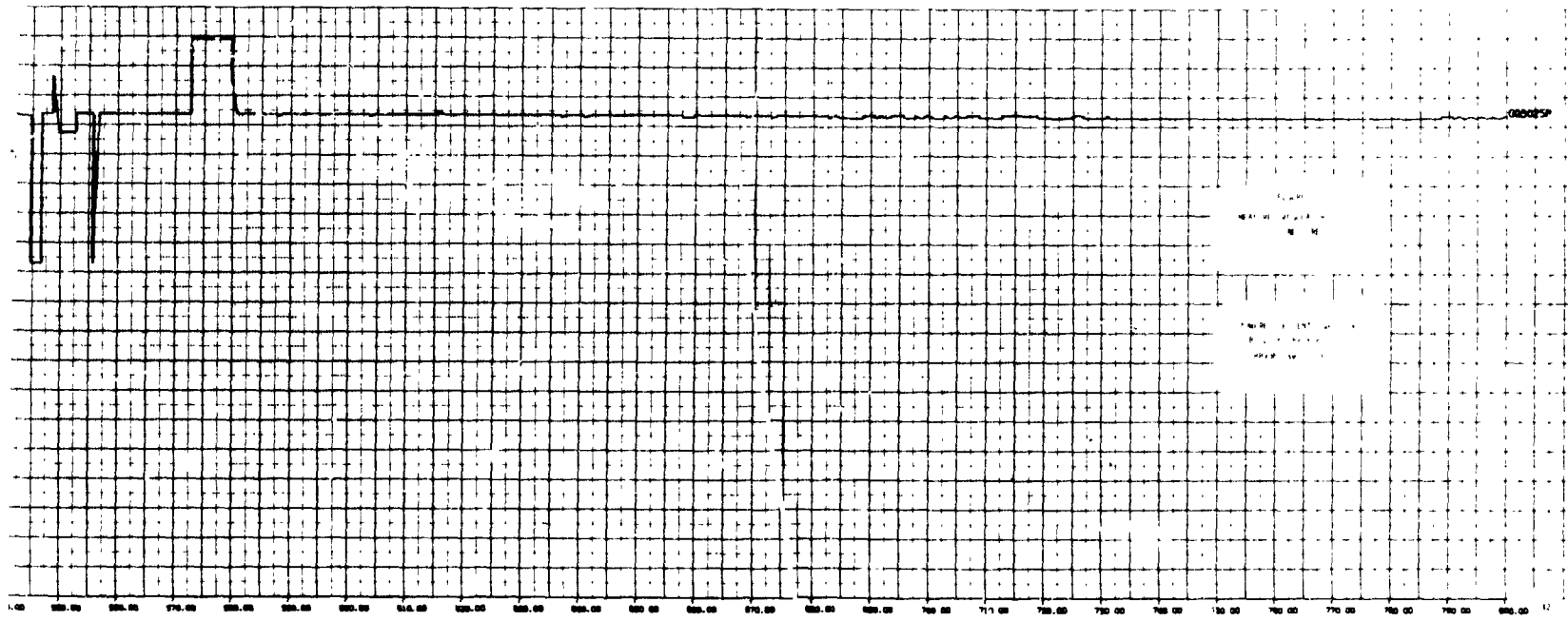




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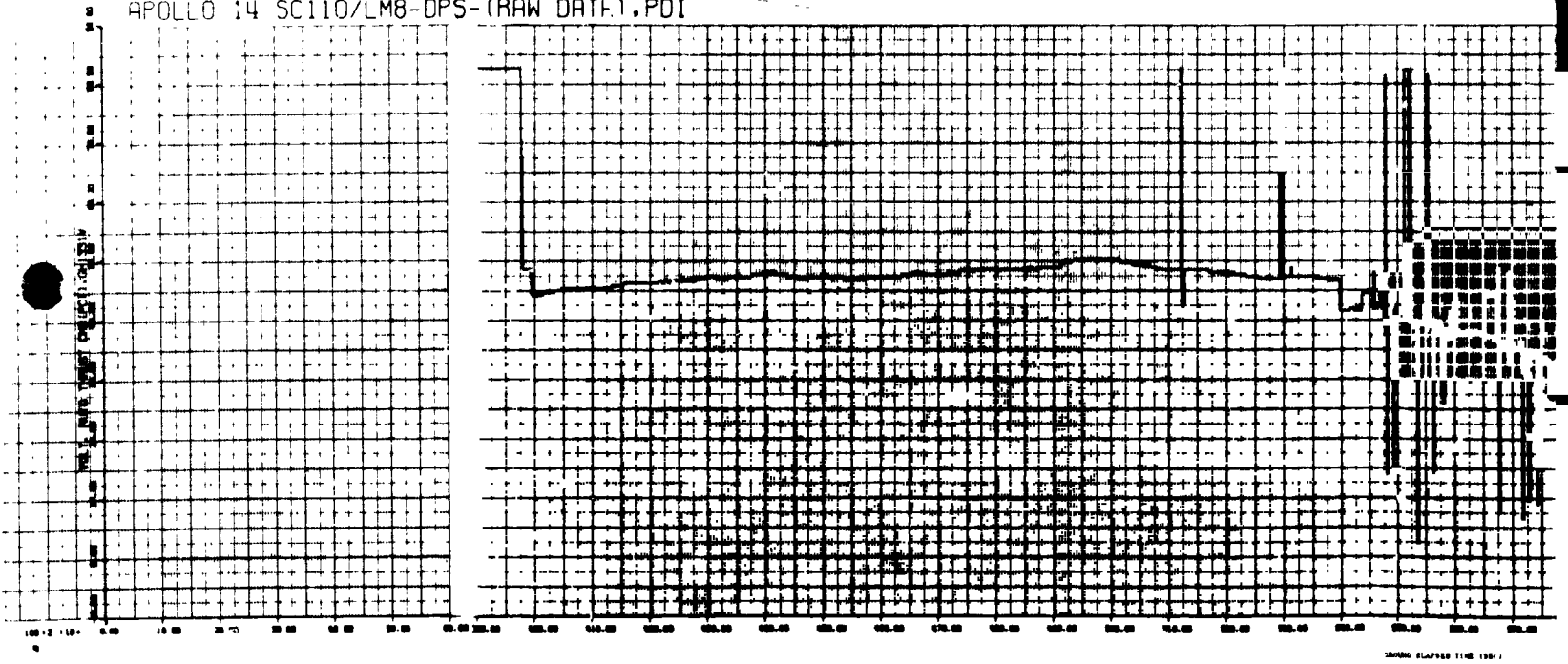


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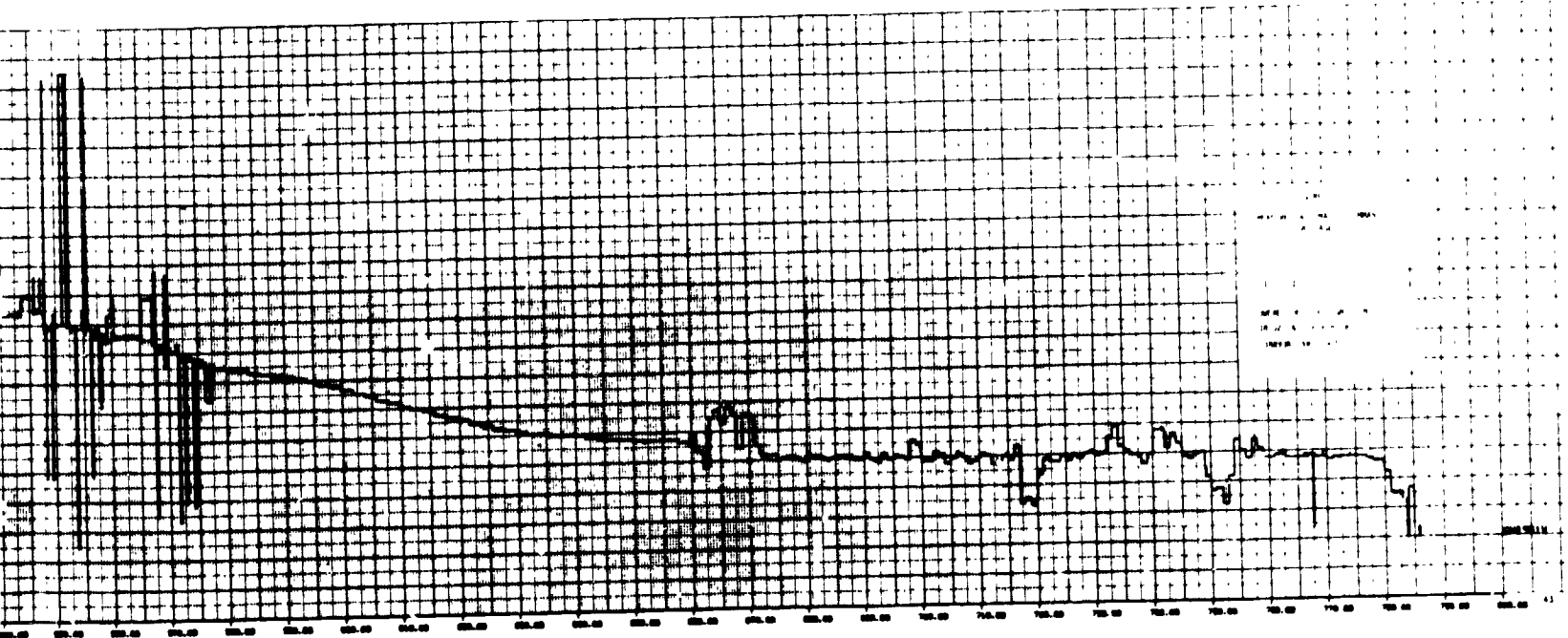


FOLDOUT FRAME  
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APOLLO 14 SC110/LM8-DPS-(RAW DATA).POI



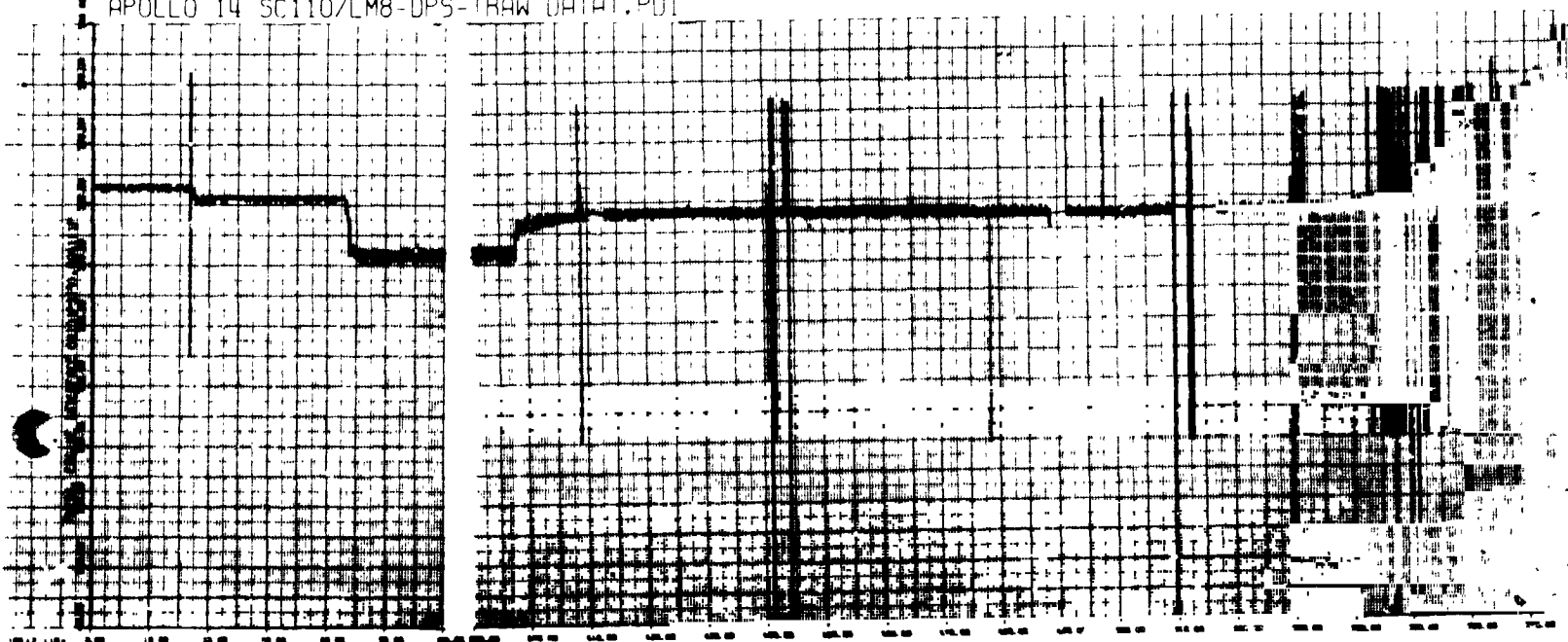
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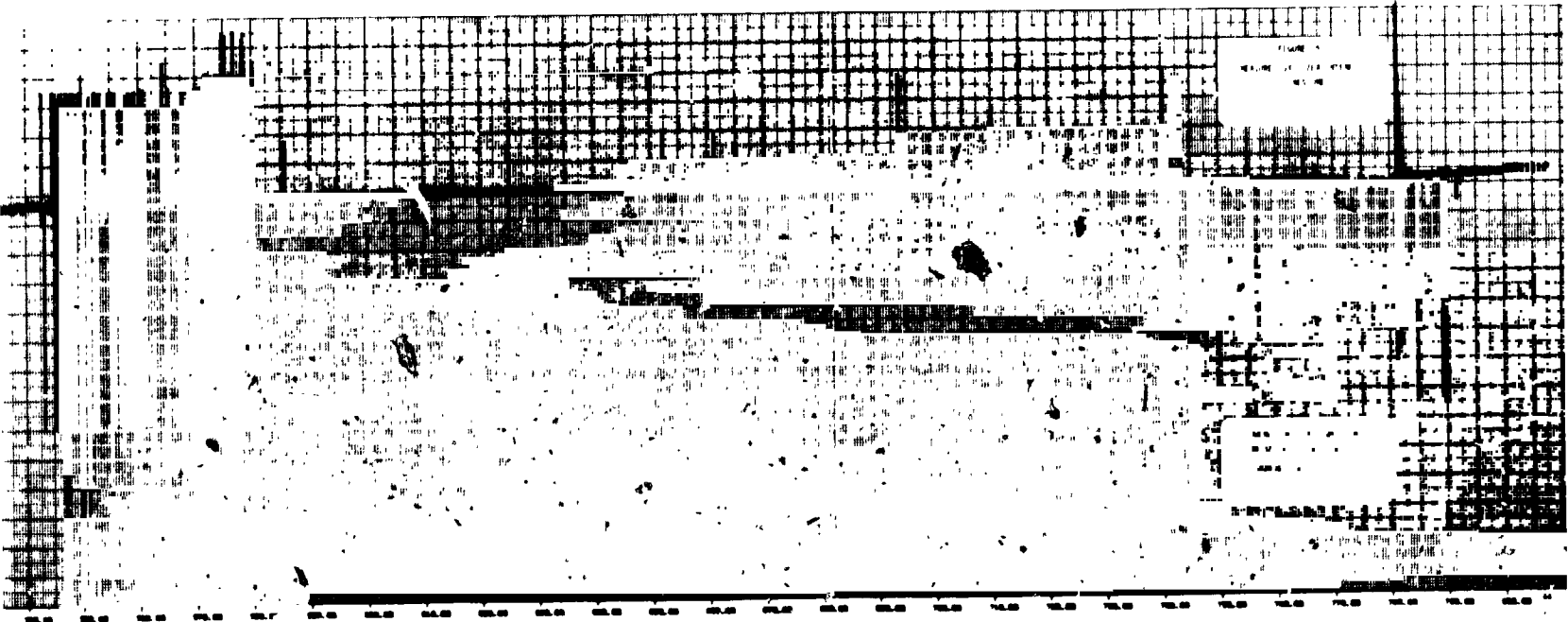
MEMO STAFFED TIME 1963

FOLDOUT FRAME  
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APOLLO 14 SC110/LM8-DPS-(RAW DATA).PDI



100001 FRAME

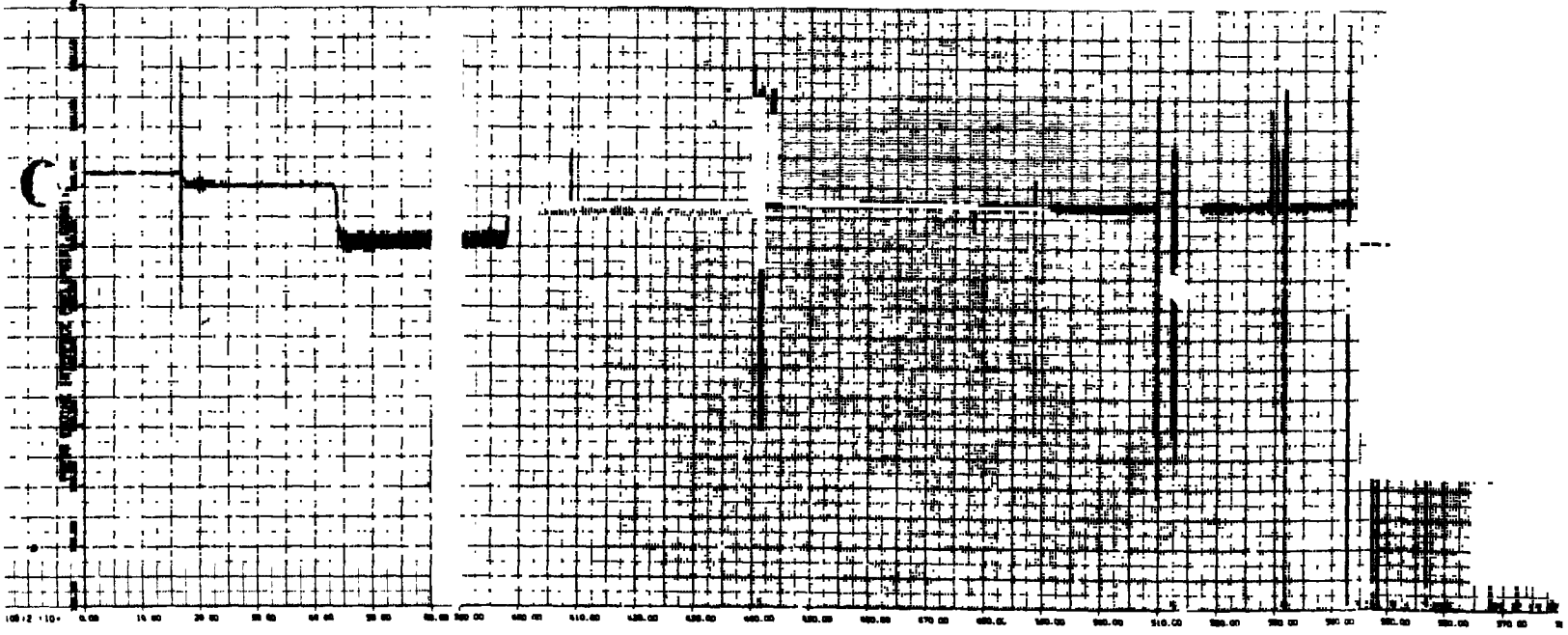


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C

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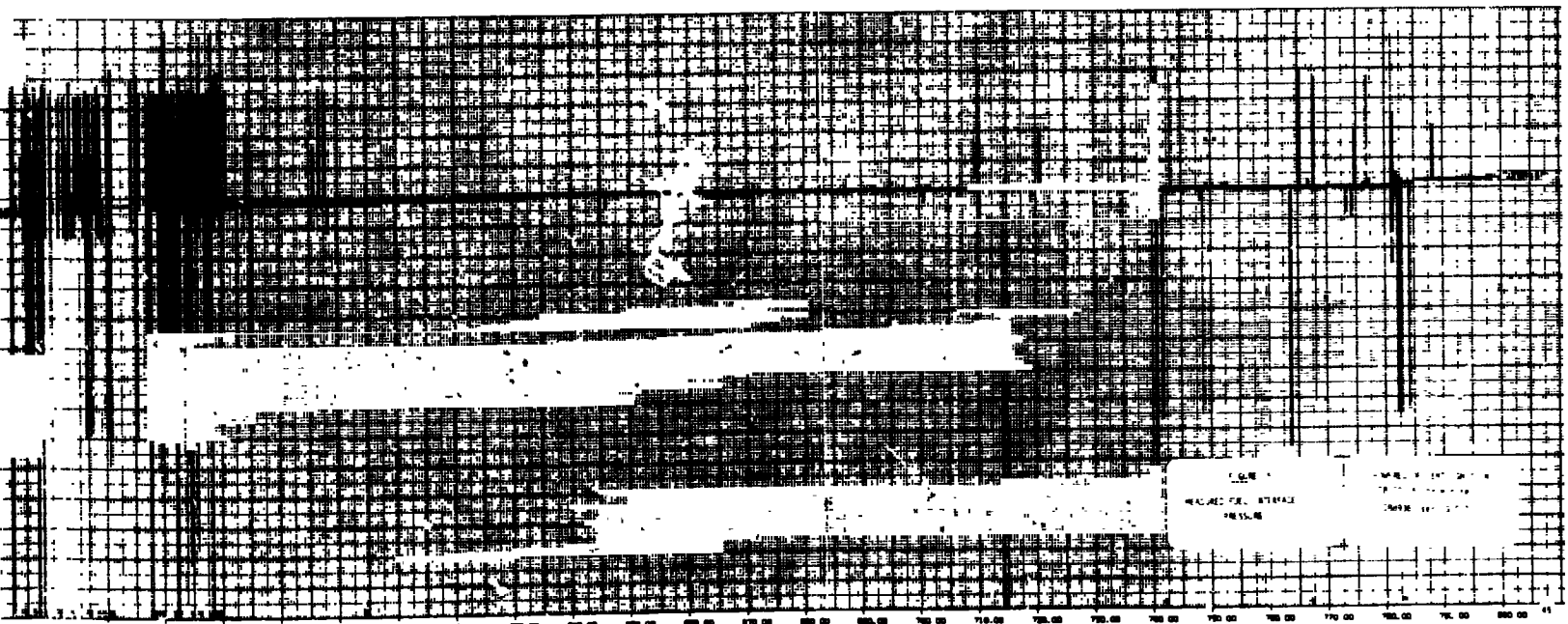


WORD ELAPSED TIME (SE)

C  
E

1200-11-17-1967



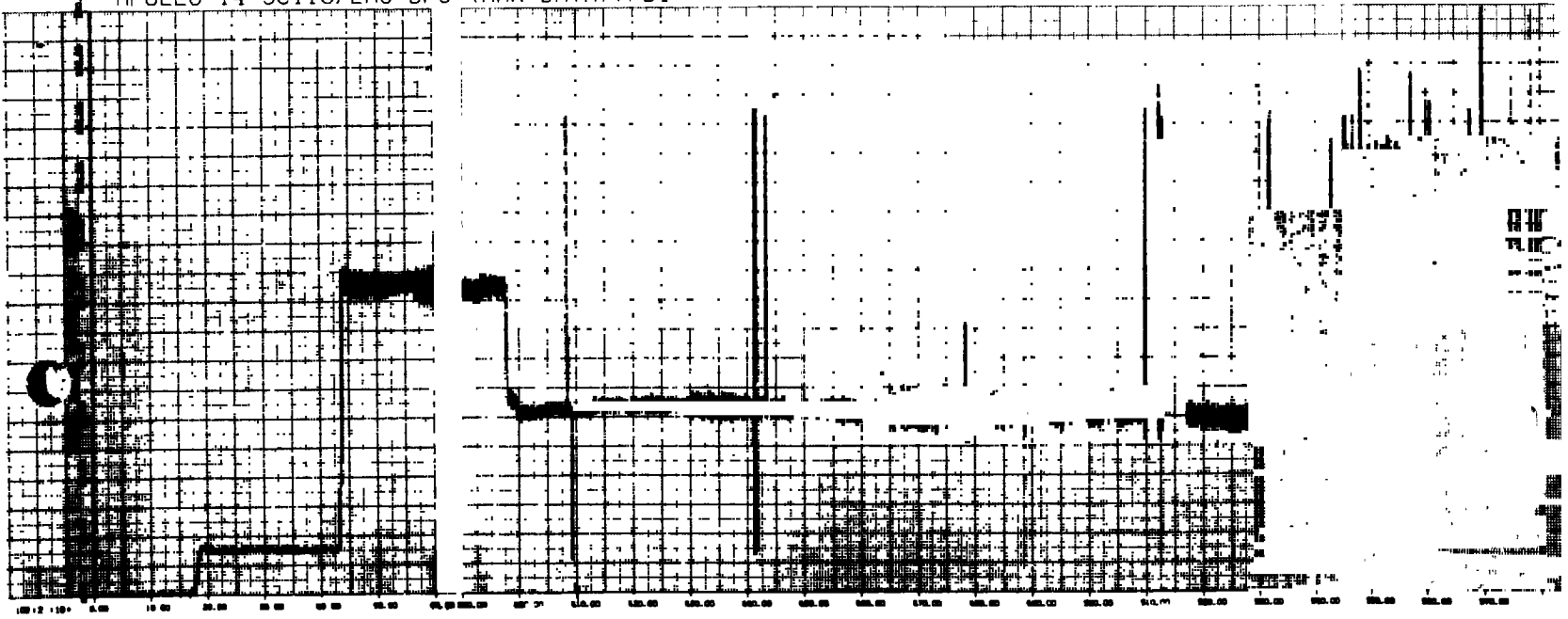


0 00 10 00 20 00 30 00 40 00 50 00 60 00 70 00 80 00 90 00 100 00 110 00 120 00 130 00 140 00 150 00 160 00 170 00 180 00 190 00 200 00 210 00 220 00 230 00 240 00 250 00 260 00 270 00 280 00 290 00 300 00 310 00 320 00 330 00 340 00 350 00 360 00 370 00 380 00 390 00 400 00 410 00 420 00 430 00 440 00 450 00 460 00 470 00 480 00 490 00 500 00 510 00 520 00 530 00 540 00 550 00 560 00 570 00 580 00 590 00 600 00 610 00 620 00 630 00 640 00 650 00 660 00 670 00 680 00 690 00 700 00 710 00 720 00 730 00 740 00 750 00 760 00 770 00 780 00 790 00 800 00 810 00 820 00 830 00 840 00 850 00 860 00 870 00 880 00 890 00 900 00 910 00 920 00 930 00 940 00 950 00 960 00 970 00 980 00 990 00 1000 00

2

C

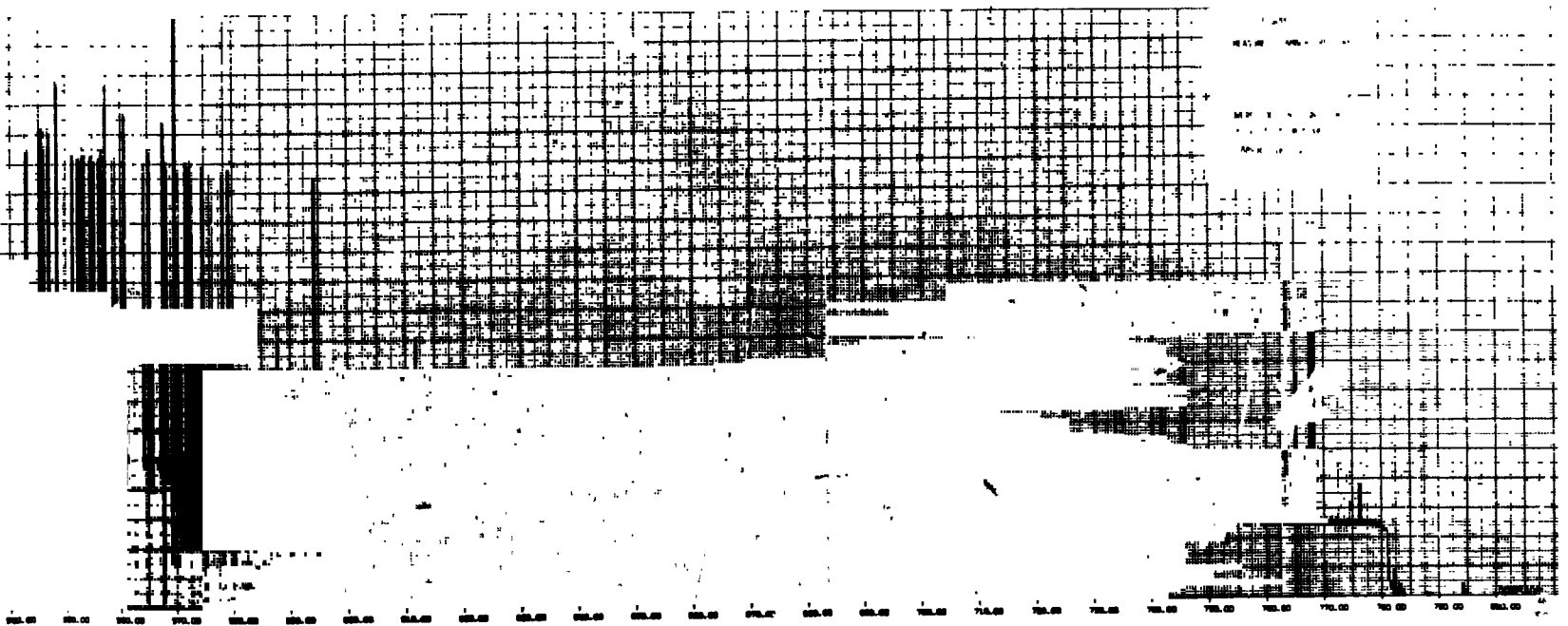
APOLLO 14 SC110/LM8-DPS-(RAW DATA).PDI



BOARD PLATED TIME (SEC)

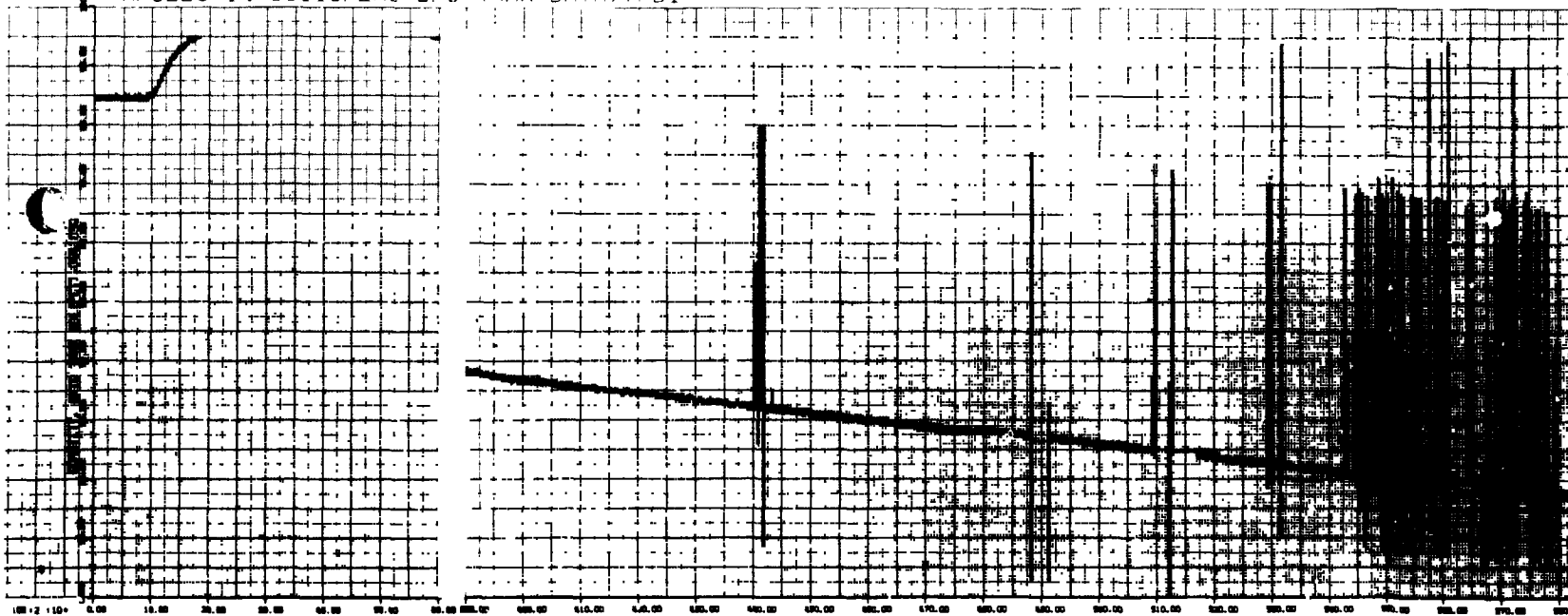
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PLAN OF THE BUILDING

APOLLO 14 SC110/LM8-DPS-(RAW DATA).PDI



SLIP SPEED TIME (SECS)

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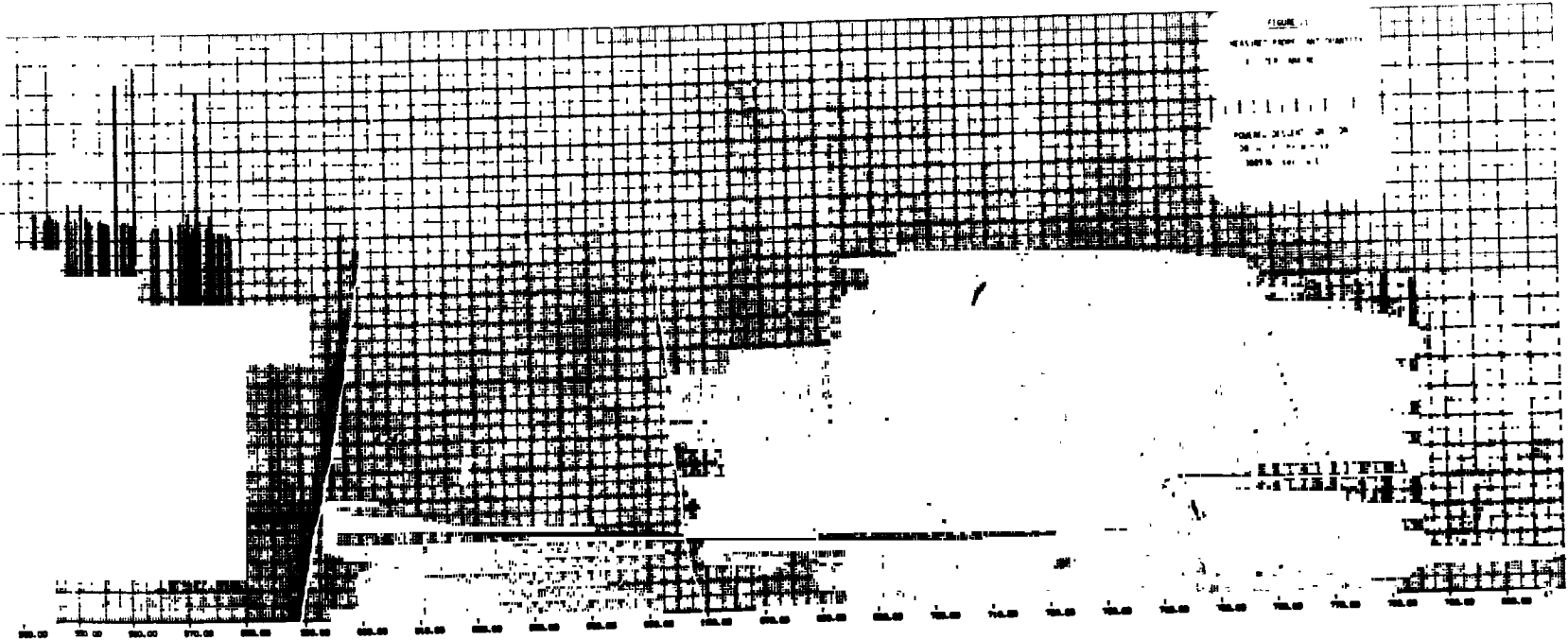
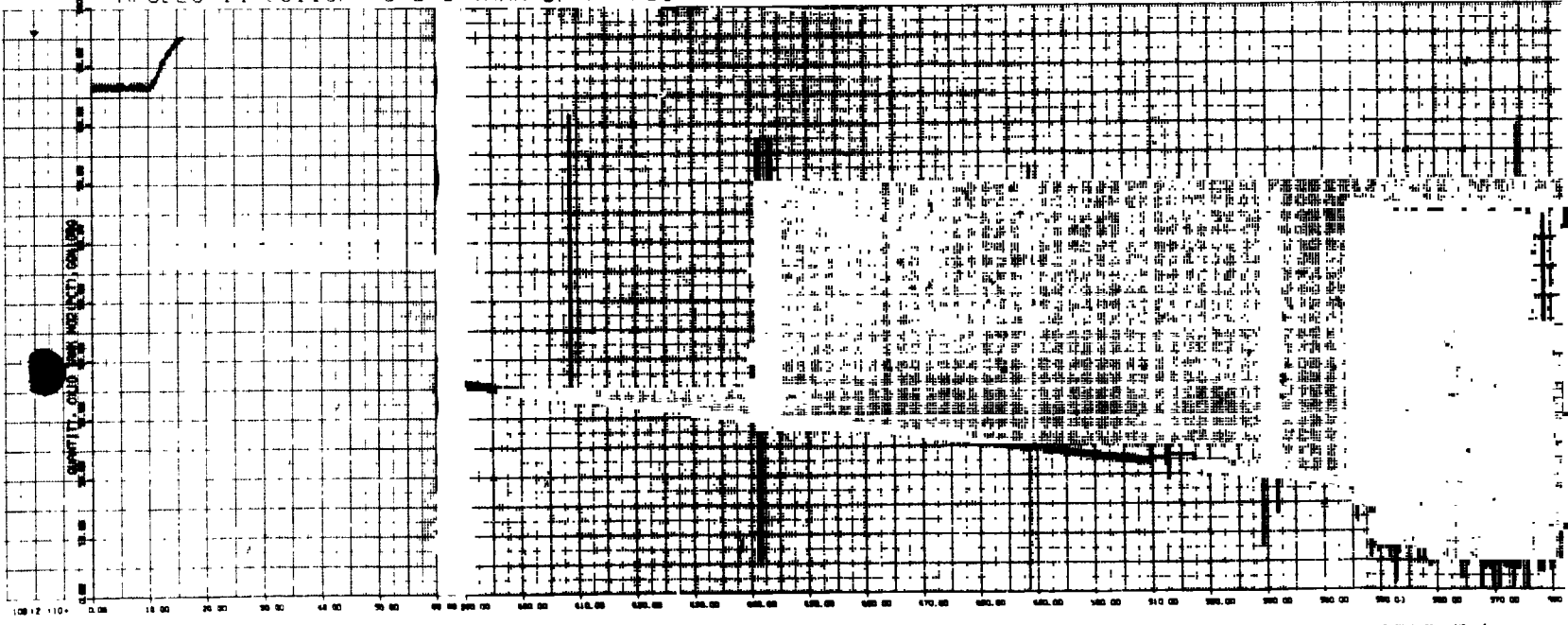


FIGURE 1  
MEASUREMENT OF THE  
...  
...  
...  
...  
...

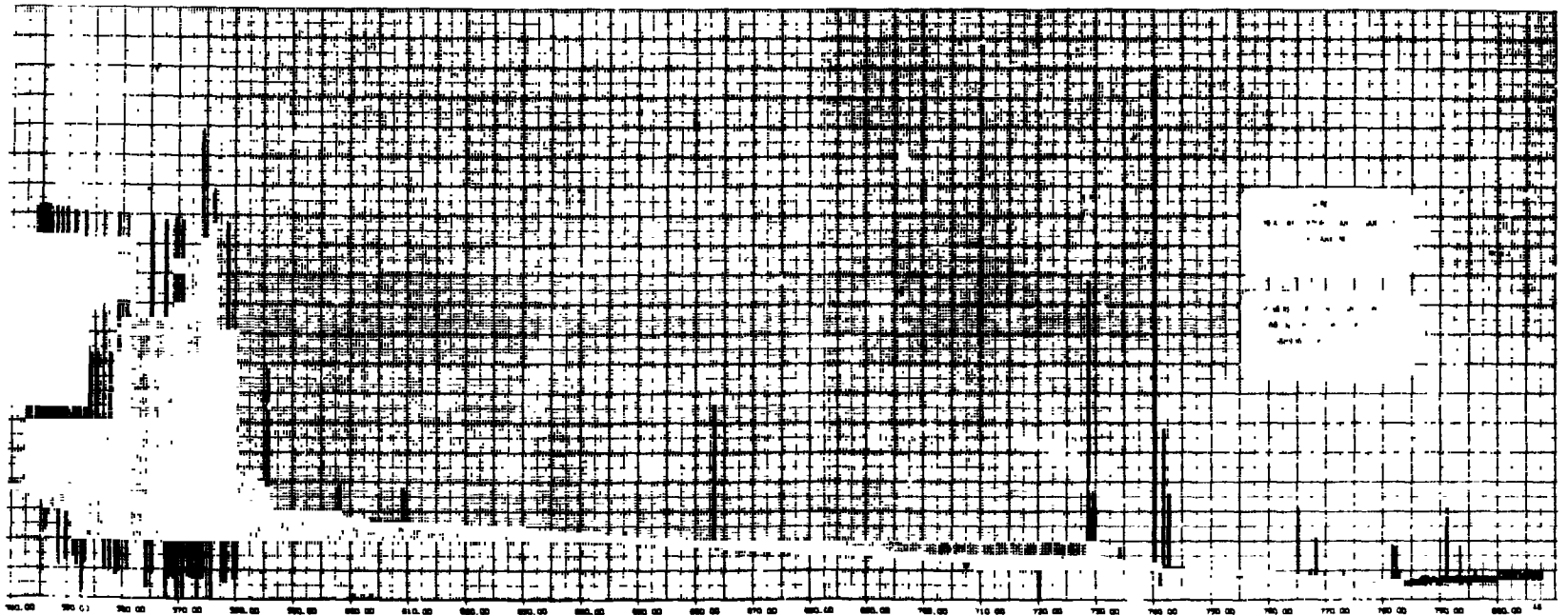
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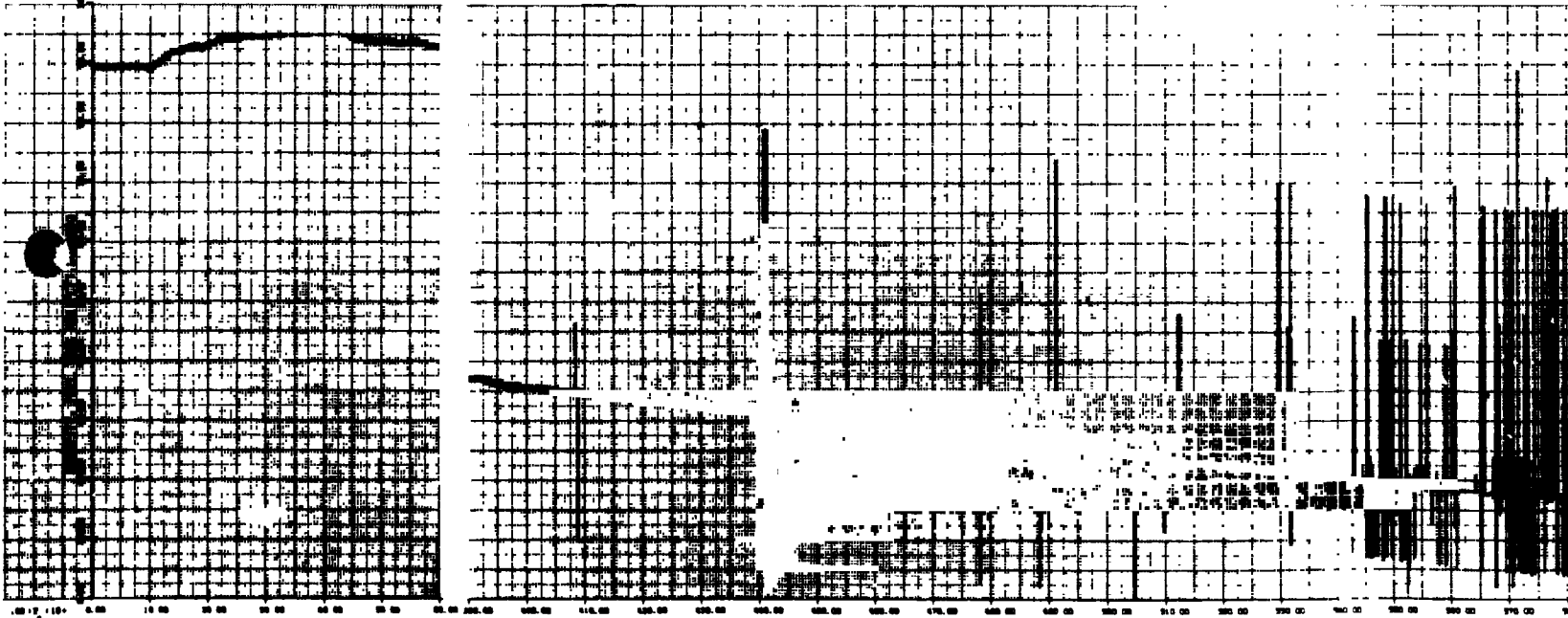
DON'T FRAME



FOLIUM 7

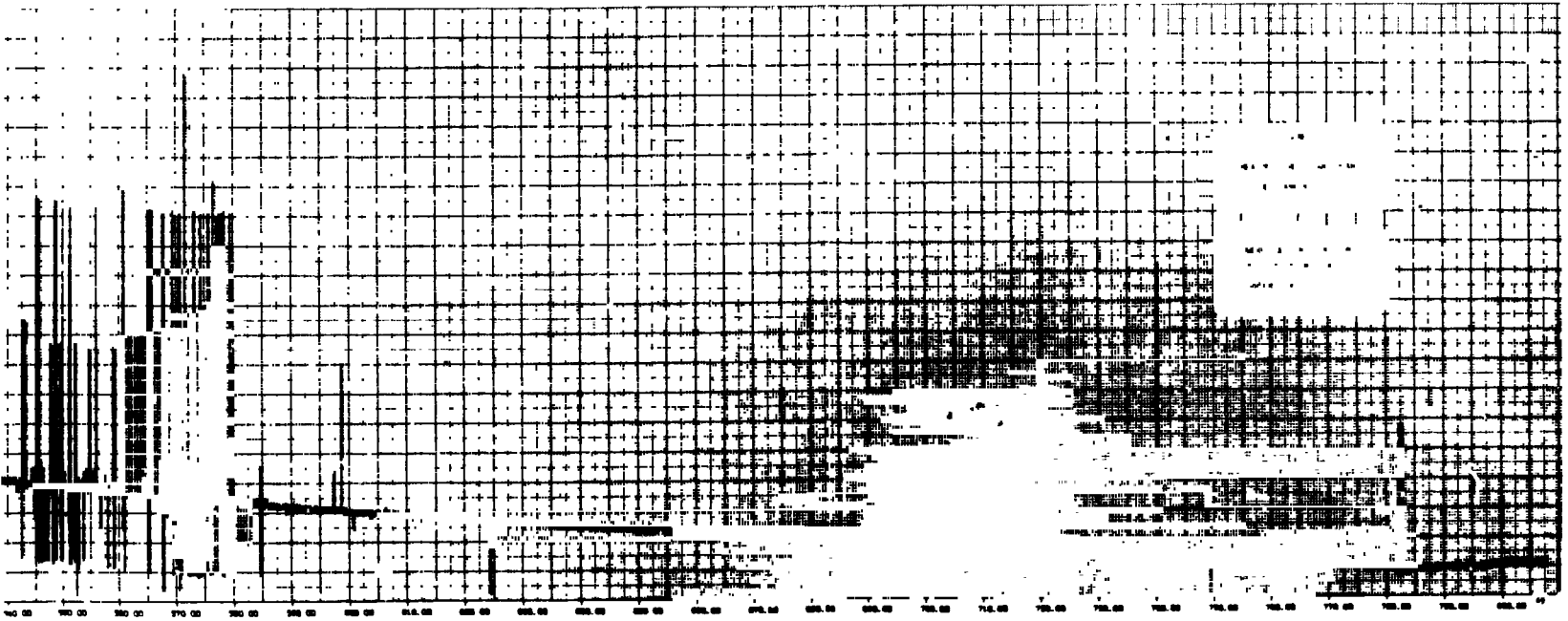
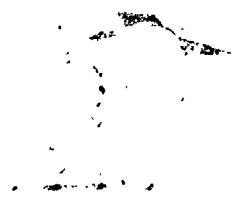
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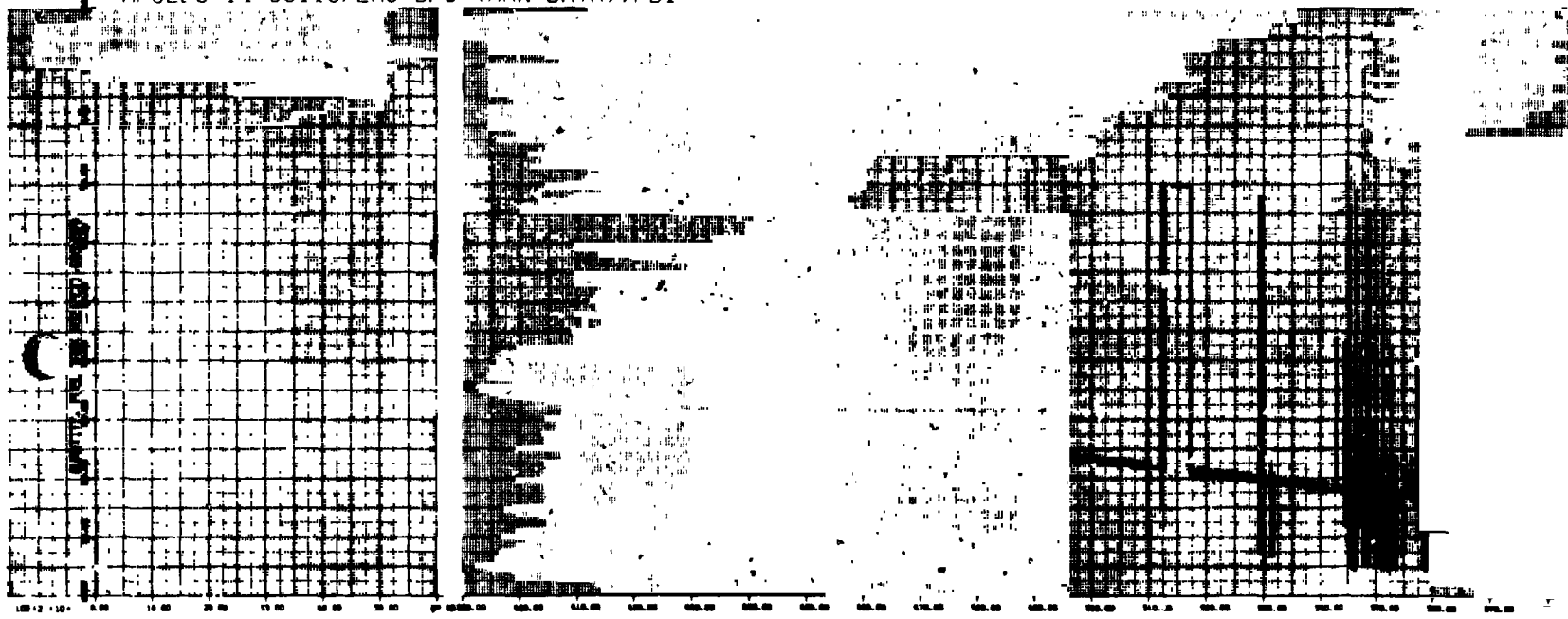


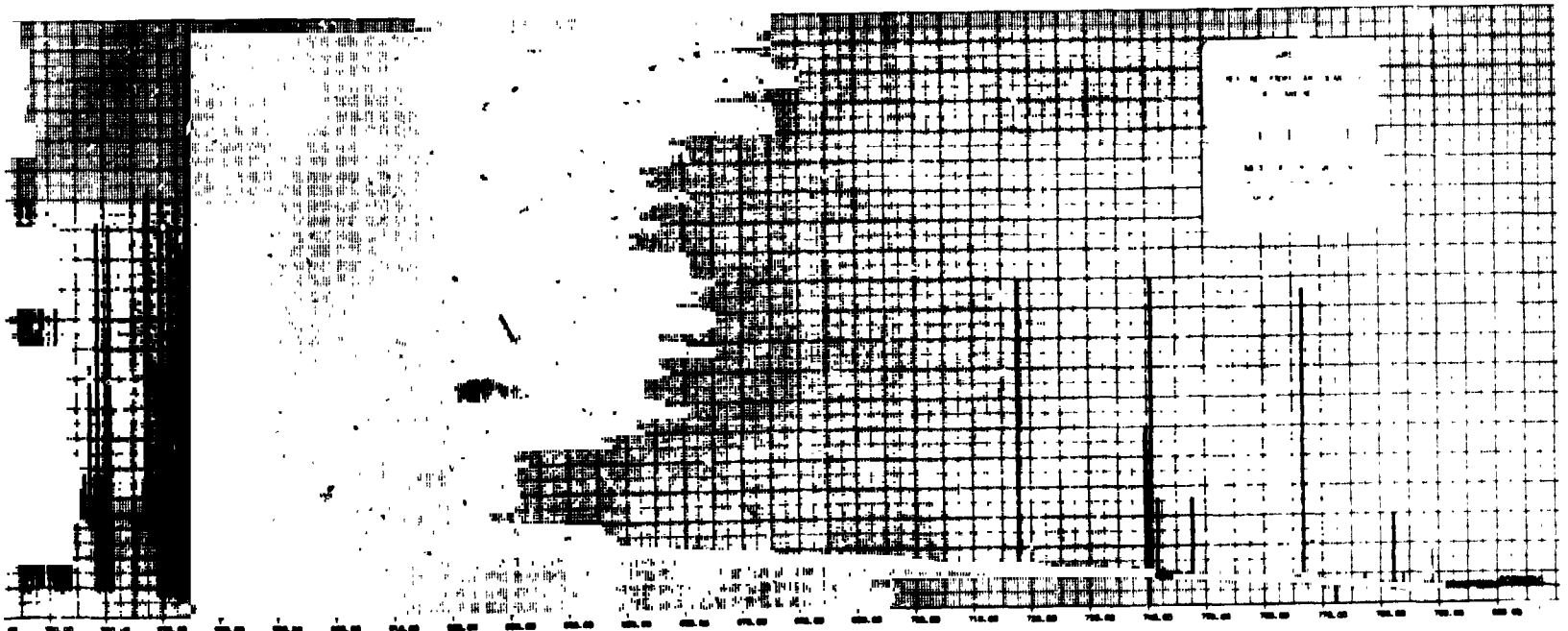
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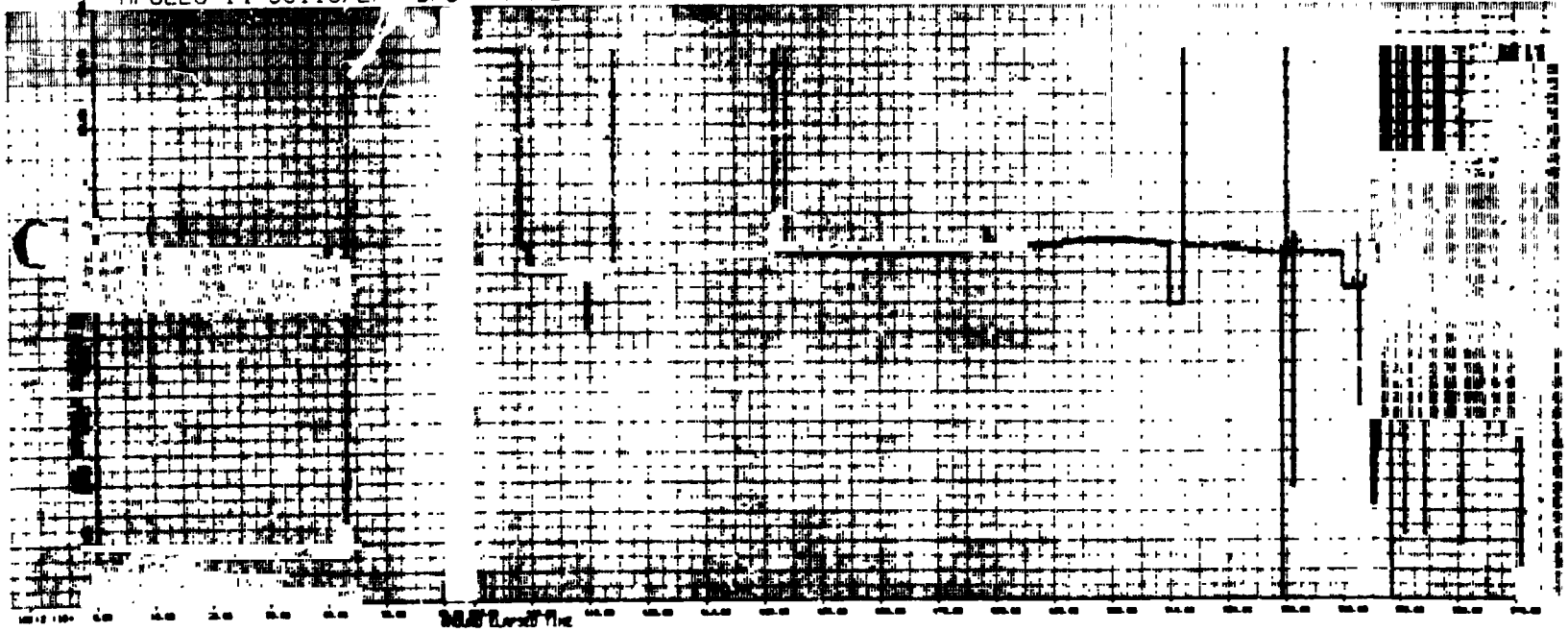


APOLLO 14 SC110/LM8-DPS-(RAW DATA).PDI





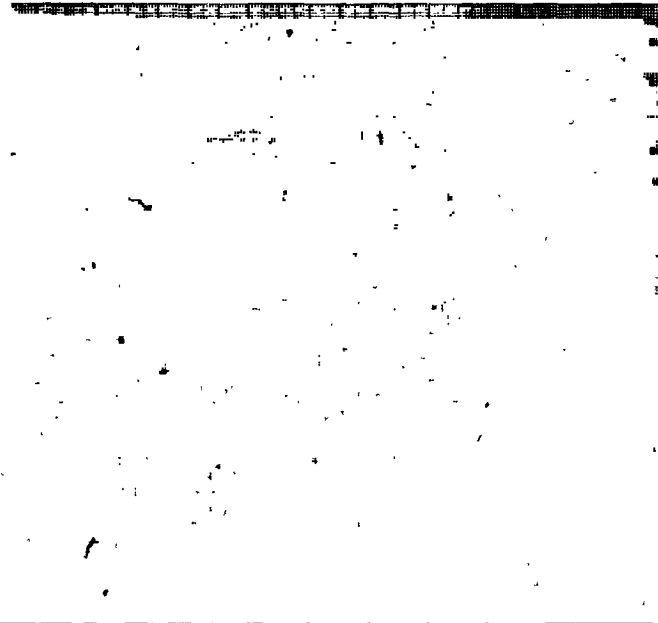
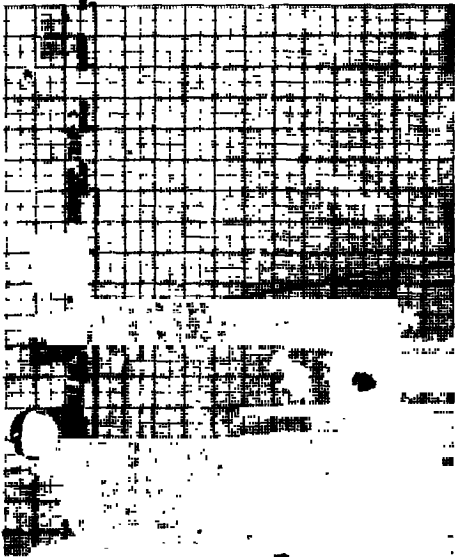
APOLLO 14 SC110/LM3-OPS-(RAW DATA).PDI



REPRODUCIBILITY OF THE ORIGINAL PAGE IS POOR.



APOLLO 14 SC110/LM8-DPS-(RAW DATA).PD1

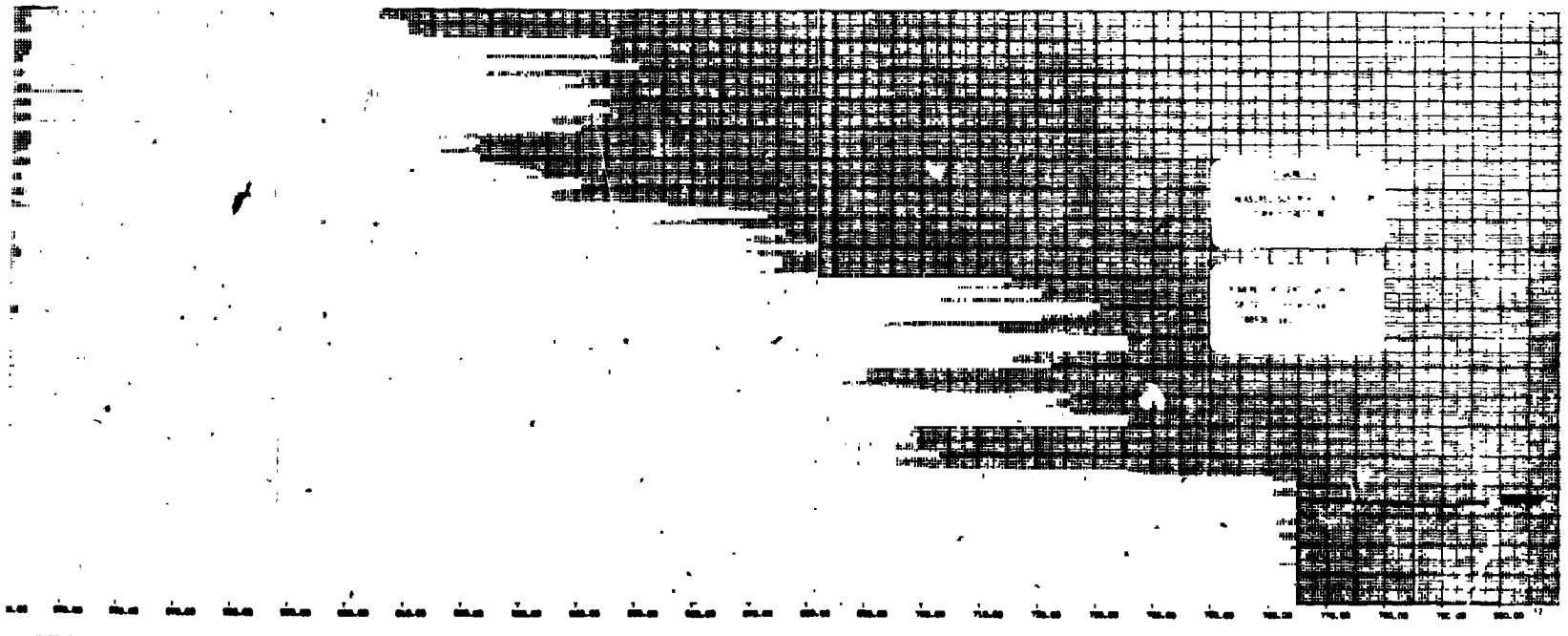


00:00 00:01 00:02 00:03 00:04 00:05 00:06 00:07 00:08 00:09 00:10 00:11 00:12 00:13 00:14 00:15 00:16 00:17 00:18 00:19 00:20 00:21 00:22 00:23 00:24 00:25 00:26 00:27 00:28 00:29 00:30 00:31 00:32 00:33 00:34 00:35 00:36 00:37 00:38 00:39 00:40 00:41 00:42 00:43 00:44 00:45 00:46 00:47 00:48 00:49 00:50 00:51 00:52 00:53 00:54 00:55 00:56 00:57 00:58 00:59 01:00

TIME ELAPSED (SEC)

CF

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00.00 01.00 02.00 03.00 04.00 05.00 06.00 07.00 08.00 09.00 10.00 11.00 12.00 13.00 14.00 15.00 16.00 17.00 18.00 19.00 20.00 21.00 22.00 23.00 24.00 25.00 26.00 27.00 28.00 29.00 30.00 31.00 32.00 33.00 34.00 35.00 36.00 37.00 38.00 39.00 40.00 41.00 42.00 43.00 44.00 45.00 46.00 47.00 48.00 49.00 50.00 51.00 52.00 53.00 54.00 55.00 56.00 57.00 58.00 59.00 60.00 61.00 62.00 63.00 64.00 65.00 66.00 67.00 68.00 69.00 70.00 71.00 72.00 73.00 74.00 75.00 76.00 77.00 78.00 79.00 80.00 81.00 82.00 83.00 84.00 85.00 86.00 87.00 88.00 89.00 90.00 91.00 92.00 93.00 94.00 95.00 96.00 97.00 98.00 99.00 100.00

TIME ELAPSED (SEC)

FIGURE 7

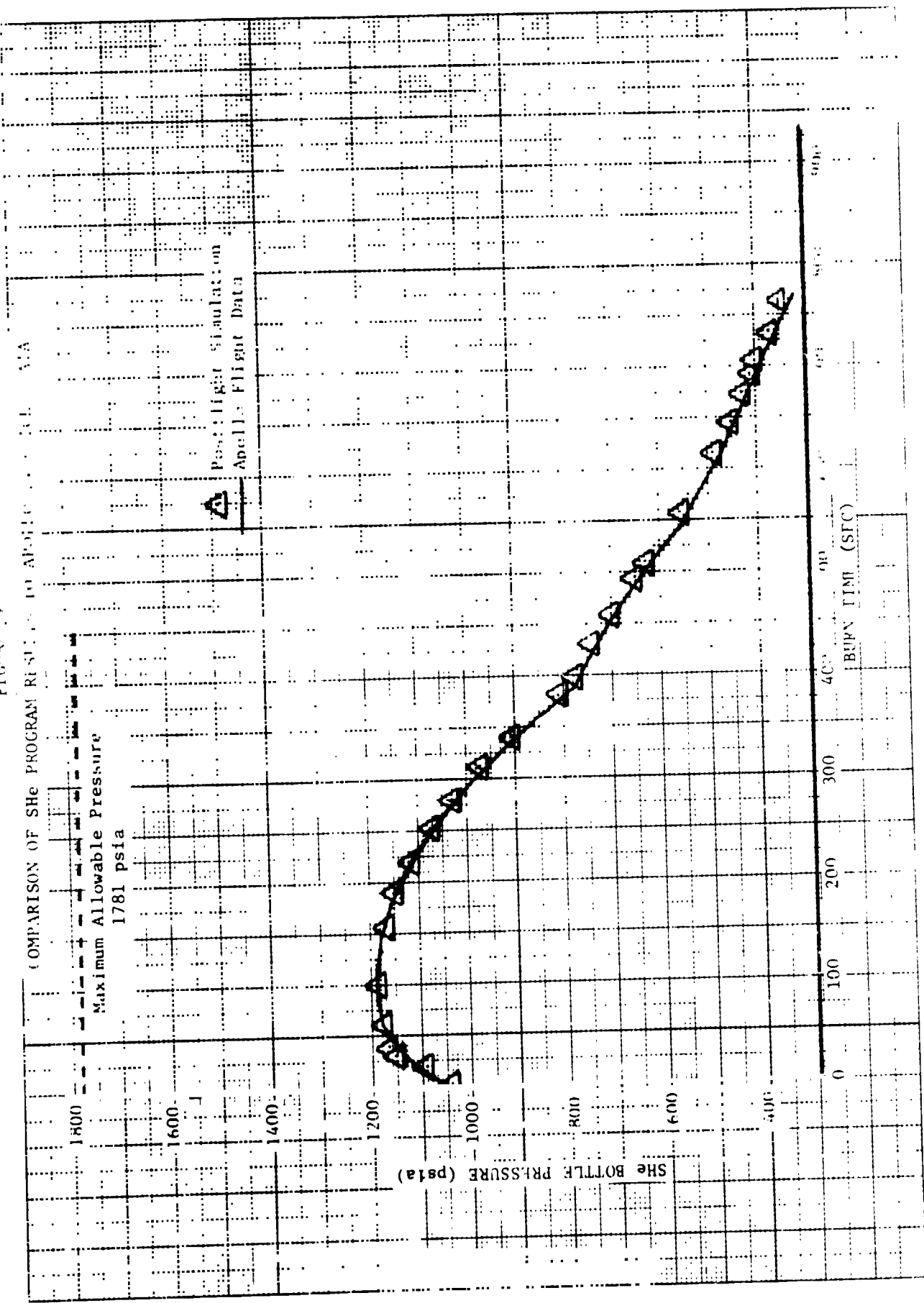




FIGURE 28

DESCENT STAGE SYSTEM SCHEMATIC

