MSC-05161
SUPPLEMENT 4

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

APOLLO 15 MISSION REPORT

SUPPLEMENT 4

DESCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
SEPTEMBER 1972

MSC-05161

APOLLO 15 MISSION REPORT

SUPPLEMENT 4

DESCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

PREPARED BY

TRW Systems

APPROVED BY

Owen G. Morris
Manager, Apollo Spacecraft Program

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

20029-H084-R0-00

PROJECT TECHNICAL REPORT

APOLLO 15

LM-10

DESCENT PROPULSION SYSTEM
FINAL FLIGHT EVALUATION

NAS9-12330

APRIL 1972

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

TRW
SYSTEMS

20029-H084-R0-00

PROJECT TECHNICAL REPORT

APOLLO 15

LM-10

DESCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

NAS9-12330

APRIL 1972

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

Prepared by
A. T. Avvenire
S. C. Wood
Propulsion Systems Section
Applied Mechanics Department

NASA MSC

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

Concurred by: _____
E. C. Currie, Manager
Descent Propulsion Subsystem

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

TRW SYSTEMS

Approved by: _____
R. J. Smith, Manager
Task E-99

Approved by: _____
J. M. Richardson, Head
Propulsion Systems Section

Approved by: _____
R. G. Payne, Manager
Applied Mechanics Dept.

TRW SYSTEMS

CONTENTS

TABLES

ILLUSTRATIONS

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 15 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 15 Mission report. In addition to further analysis of the DPS, this report brings together information from other reports and memorandums analyzing the performance in order to present a comprehensive description of the DPS operation during the Apollo 15 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 (PQGS) 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-10 Descent Propulsion System during the Apollo 15 Mission was evaluated and found to be satisfactory. The average engine effective specific impulse was 0.2 second lower than predicted, but well within the predicted 1σ uncertainty. The engine performance corrected to standard inlet conditions for the FTP portion of the burn at 33 seconds after ignition was as follows: thrust, 9807 lbf; specific impulse, 305.8 sec; and propellant mixture ratio, 1.594. These values are +0.16, 0.0 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) The chamber pressure transducer had a large drift, exhibiting a maximum error of about 5 psi at approximately 130 sec after engine ignition. This drift is due to thermal effects. Apparently, as the transducer temperature increases, its calibration "wanders." A similarly large error occurred during the Apollo 14 DPS descent burn. Previous flights have also had transducer drifts of smaller magnitude (less than 1 psi). 2) The fuel and oxidizer interface pressure measurements appeared to be low during the entire flight. The discrepancy is assumed to be a measurement bias (-0.77 and -2.48 psi for oxidizer and fuel, respectively). 3) The propellant quantity gaging system did not perform within expected accuracies during the first 150 sec of the burn when the fuel 1 and 2 probes were biased low by as much as 4%. The fuel 1 probe shows a bias (seen as a residual error in Figure 8) of about 3.5% for approximately 100 sec. into the analyses.

The low level sensor actuation time was about 9 seconds later than expected. This discrepancy is discussed in detail in Section 7.

3. INTRODUCTION

The Apollo 15 Mission was the eighth flight and the seventh manned flight, of the Lunar Module (LM). The mission was the fourth successful lunar landing.

A primary detailed test objective (DTO) of Apollo 15 with respect to the LM descent stage was to determine the performance of the modified LM descent engine. This new version of the descent engine was equipped with a quartz chamber and lengthened nozzle. Its performance is discussed in detail in Section 4.

The space vehicle was launched from Kennedy Space Center (KSC) at 9:34:00.6 a.m. (EDT) on July 26, 1971. At 104:30:09 (G.E.T), the Descent Burn (PDI) was initiated and lasted about 739 sec. The burn was started at the minimum throttle setting and after approximately 26 sec., the thrust was increased to the fixed throttle position (FTP). An automatic descent was maintained to approximately 658 seconds after ignition, 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. Lunar landing occurred at 104:42:29.3 G.E.T. ending the DPS mission duty cycle. After a lunar stay of approximately 67 hours, the APS was ignited and the ascent stage of the LM was put into lunar orbit. Data from the DPS was terminated at ascent stage lift-off.

The actual ignition and shutdown times for the DPS firing are 104:30:09.4 G.E.T. and 104:42:28.1 G.E.T., respectively. The thrust profile for the DPS burn is shown in Figure 1.

The DPS burn was preceded by a two-jet +X LM Reaction Control System (RCS) ullage maneuver of 7 seconds to settle propellants.

The Apollo 15 Mission utilized LM-10 which was equipped with DPS engine S/N 1046. The engine and feed system characteristics are presented in Table 1.

4.0 STEADY-STATE PERFORMANCE ANALYSIS

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 embodied 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. One of the primary DTO's associated with the descent stage was the inflight performance of the modified LM descent engine. The engines inflight throat erosion characteristics were close to predicted being only .6% lower at the end of FTP than predicted (5.9% vs. 6.5%). This is well within the 3 sigma uncertainty of ±1.9%. The engine inflight specific impulse was 305.8 sec, as predicted. The 3 sigma uncertainty is

±.6 sec. The inflight thrust was 9807 lbf, 16 lbf higher than predicted but well within the ±48 lbf 3 sigma uncertainty. The inflight values of thrust and specific impulse are reduced to standard interface conditions.

The Apollo Propulsion Analysis Program (PAP) results presented in this report are based on reconstructions using data from the flight measurements listed in Table 2.

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 87 lbm of consumables other than the DPS propellant were used. Of that amount, 54 lbm were 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 .05 lbm/sec.

The DPS steady-state FTP performance was determined from the analysis of a 400 second segment of the burn. The segment of the burn analyzed commenced approximately 31 seconds after DPS ignition (FS-1) and included the flight time between 104:30:40 hours and 104:37:20 hours ground elasped time. Engine throttle down to 60 percent occurred 10 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.

The inflight throat erosion agreed well with predicted values. At the end of the FTP portion of the burn, the inflight throat erosion was 5.9% or within 1% of the predicted value of 6.5%. Figure 2 shows a comparison between the predicted throat erosion and the estimated inflight throat erosion.

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 Performance Analysis Program (PAP). 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. The PQGS system measurements also behaved poorly; any attempt at including them as measurements failed. Therefore, the flight analysis was accomplished without the direct benefit of the gaging system data. (See Section 7 for detailed explanation).

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) 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 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 1 psi lower than that from GQ 3018P. Based on earlier analyses and preflight tests, the data from GQ 3018P was used for the analysis. Also, GQ 3018P appeared to behave better, that is, was much smoother and therefore consistent with previous missions. It should be noted that tests made at KSC several weeks prior to launch on the helium regulator indicated that GQ 3025P should have been 1.0 psi higher than GQ 3018P. The helium regulator pressure determined by the program is approximately the average of GQ 3025P and GQ 3018P.

The inflight value of the fuel interface pressure (GQ 4111P) was biased by -2.77 psi, although this is within the instrument accuracy. The oxidizer interface pressure was also biased by -.77 psi.

The gaging system data (Figures 21-24) could not be used due to what appears to be a large scaling error. The oxidizer gages read high at 70 sec after ignition and gradually improved during the entire FTP burn. The fuel gages initially read low and once again gradually improved with time. However, it is felt that at no time was there sufficient confidence

in the gages to use them in the PAP analysis as a measurement variable. However, the readings of the gages at the latter part of the FTP region appear to be accurate enough to compare with the calculated values from the analysis program. Table 5 shows the close agreement between the measured and calculated gaging system data from about 270 sec. into the descent burn to touchdown. Therefore, although the gaging system data was not input into the PAP directly, they were used to help validate the results obtained from PAP. The gaging system data at the end of the burn were accurate enough to be useful to flight control personnel operating in real time support to the mission.

Comparison with Preflight Performance Predictions

Prior to the Apollo 15 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^2	54.4

The following table presents ground test data and flight test data adjusted to standard inlet conditons. Comparing the corrected engine flight performance at FTP during the Descent Burn to the corrected ground test data shows the flight data to be 0.16% more, 0.0% more, and 0.0% more for thrust, specific impulse and mixture ratio, respectively. These differences are within the engine repeatability uncertainties and within the performance specification ranges.

Parameter \ Data Source	Ground Test Engine Prediction Characterization	Flight Analysis Results	Performance Specification Range	Engine Repeatability Uncertainty 3σ
Thrust, lbf	9791	9807	9712 - 10027	9742 - 9840
Specific Impulse, sec	305.8	305.8	> 305.0	305.14 - 306.46
Mixture Ratio	1.594	1.594	1.586- 1.614	1.590 - 1.598

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 (GQ 3018P) drives the program and the measured value of throttle command voltage (GH 1331V) 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 values with flight measurements and a minimum variance match is not performed.

Based on the FTP analysis, it was determined that a -.5 psia correction should be made to the regulator outlet pressure (GQ 3018P). For the simulation, the initial values of throat erosion, LM vehicle weight and propellant weights were obtained from the end point conditions of the FTP analysis. The damp weight was adjusted for non-DPS consumables during the throttle region 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 +431 seconds) and continued for 308 seconds. This includes all of the powered descent burn after throttle down and includes the flight time between 104:37:20 hours to 104:42:28 hours. Typical values of the simulation results are presented in Table 4.

Figures 12 through 14 present plots comparing the preflight predicted and the analysis program simulated values of throttle command precent, mixture ratio, and specific impulse.

Figures 15 through 25 presents the inflight values of 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.

6.0 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 ration (MR) and overall effective specific impulse (Isp). The vehicle effective specific impulse was computed based on spacecraft weight reduction due to DPS propellant consumption, and weight reduction due to non-DPS consumables usage. The non-DPS consumables usage is approximately 0.05 lbm/sec during FTP and 0.22 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.

	Propellant Consumption (lbm) Oxidizer	Fuel	Average MR (O/F)	Vehicle Effective Isp (sec)[1]	Engine Effective Isp (sec)[1]
Preflight Prediction	11249.6	7061.5	1.593	301.7	304.7
Analysis Program	11259.9	7064.8	1.594	302.4	304.5

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 6808.4 ft/sec. The total measured thrust velocity gain, 6813.0 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 the analysis is within this uncertainty.

The analysis results are within the predicted 3σ uncertainties of $\pm$ 1.8 sec and ± 0.012 for effective specific impulse and mixture ratio, respectively.

---

1 Calculated from FS-1 plus 31 seconds.

7. PQGS EVALUATION AND PROPELLANT LOADING

Propellant Quantity Gaging System

The PQGS measurements for Apollo 15 were not used in the PAP program as active measurement inputs. This was due to the poor performance of the gages during most of the FTP portion of the burn. Figures 28-31 show the residual errors (difference between the measured and calculated values) as well as the filtered measurement data. Note that both oxidizer tanks read high (a total of about 100 lbm) while both fuel tanks read low (a total of about 120 lbm) at 60 seconds. The poor performance of the fuel probes is due primarily to late activation. The probes were activated only minutes prior to the descent engine ignition. Tests on similar fuel probes indicate that the probes should be activated 30 minutes prior to ignition in order to allow the probe to generate the full 5 volts associated with a full fuel tank. The late activation of the PQGS system made the accuracy of the fuel probes questionable during most of the entire FTP burn. On the other hand, the higher than usual initial readings of the oxidizer probes and subsequent gradual improvement during the FTP portion of the burn cannot be explained.

In an attempt to improve the PQGS data, a full calibration of the probes was tried using data supplied by Grumman. Figures 6-9 show the results as the difference between measured and calculated data. The step like data apparent in the raw data are gone, yielding a somewhat smoother curve. However, the overall inaccuracies of the gages could not be reconciled. It was, therefore, apparent that a better analysis could be made by eliminating them as measurement variables in PAP. However, as was stated in Section 4, use was made of the PQGS data as a comparison against PAP results for the better part of the FTP burn and, in particular, the cavitation portion of the burn.

At the end of the analyzed portion of the FTP burn, the difference between the measured and calculated propellant liquid levels were 0.0, 0.4, -0.3, 0.5 % for the Ox1, Ox2, Fu 1 and Fu 2 respectively. At the end of descent burn, the differences were -0.3., 0.2, -0.2, and 0.9%, 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%	1.5%	20-0%	1.0%
8-0%	1.0%	-	-

*Percent of Full Tank

These expected accuracies are used in lieu of the specification accuracies which White Sands Test Facility(WSTF) tests indicate should not be met.

Table 5 presents a comparison of the measured data and the best estimate of the actual values at various time points. While the differences 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 714.9 lbm and 455.6 lbm for oxidizer and fuel, respectively. Of these quantities, 691.4 lbm of oxidizer and 445.1 lbm of fuel are usable to depletion (including

burning usable propellants in the feed lines). Applying the propellant flow-rates at engine shutdown, 112 seconds of hover time remained based on computed residual propellants. The measured quantities indicate 103 seconds of hover time, that is, about 622 lbm of usable oxidizer and 433 lbm of usable fuel. Both measured and calculated data indicated an oxidizer depletion.

The propellant low level sensor was activated at about the time touch-down occurred. Based on the predicted time of 731 seconds, the low level sensor was triggered about 9 seconds late at 740 seconds. This is believed to be caused by the removal of the balance line between the two oxidizer and two fuel tanks. The removal of these lines causes different flow patterns out of the tanks. This phenomena, explained in Reference 7, was not realized prior to the flight, but will be taken into account on future missions.

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 90.41 lbm/ft^3 and an average fuel density of 56.52 lbm/ft^3 at a pressure of 240 psia and a temperature of 70°F were determined from the samples. The propellant loads were 7537.6 lbm of fuel and 12023.9 lbm of oxidizer. The total DPS propellant onboard was 19561.5 lbm.

8. PRESSURIZATION SYSTEM EVALUATION

The DPS Supercritical Helium (SHe) Pressurization System performed satisfactorily during the Apollo 15 mission. The data plotted in Figure 26 shows that the flight data falls within the predicted performance (nominal $\pm 3\sigma$).

A postflight simulation for the SHe system generated with the SHe program with flight data as input, is presented in Figure 27. The flight data used as input include: 1.) SHe bottle pressure at PDI, 2.) DPS engine duty cycle (throttle setting versus burntime Figure 1), 3.) The average ullage pressure for the propellant tanks at PDI.

The most significant variation between the preflight and postflight data was found in the actual duty cycle, which when used as input to the prediction program produced a better match to the flight data as shown below.

SHe Bottle Pressures, PSIA					
Comparison Point	Preflight Prediction	Postflight Simulation	Flight Data	Delta Preflight-Flight	Delta Postflight-Flight
Press. at PDI	1318.	1276.	1276.	+42	-
Max. Pressure	1453.	1415.	1410.	+43	+5
Press. at T/D	384.	384.	459.	-75	-75

Although the match during the first part of the DPS burn is good, the prediction indicates a low pressure during the last half of the burn. This could be indicative of a warmer helium load in the flight bottle than the assumed value used in the program. The pre-launch and coast pressure rise rates for the SHe were found to be 8.8 and 7.2 psia/hour, respectively. The remaining SHe system performance parameters remain the same as reported in the preliminary flight evaluation (Reference 6).

9. ENGINE TRANSIENT ANALYSIS

The mission duty cycle of the Descent Propulsion System for Apollo 15 included one start at the minimum throttle setting, and one shutdown at approximately 29% throttle. Considerable throttling occurred during the Descent Burn, all of which were commanded by the LGC.

Start and Shutdown Transients

Table 6 presents the start and shutdown times and total impulses for the Apollo 15 mission and, for comparison, similar parameters for the other Apollo missions incorporating the DPS. 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 lead time applied.

The ignition delay from FS-1 to first rise in chamber pressure was approximately 0.61 seconds. The delay time compared favorably with the first burn delays observed during Apollo 13 and 14. The delay time for the PDI burn of Apollo 14, the only other single burn DPS mission, was 0.55 seconds indicating close agreement with Apollo 15.

The start transient from FS-1 to 90% of the minimum steady-state throttle setting required 2.35 seconds with a start impulse of 440 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 0.08 seconds with an impulse of 71 lbf-sec.

The shutdown transient required 2.06 seconds from FS-2 to 10% of the steady-state throttle setting with an impulse of 1113 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 26 seconds into the burn, required approximately 1 second. The engine then remained at FTP for 416 seconds. The second command, from FTP to 59%, occurred 442 seconds after ignition and required approximately 0.5 second. This value of 0.5 second compared favorably with similar maneuvers on previous flights. Little throttling was performed during the next 122 seconds. The LM Guidance Computer then commanded a ramping decrease in the throttle setting from 60% to 33% over 96 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 79 seconds of the burn, the LGC commanded approximately 60 throttle changes in the 28% to 45% 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-05161, "Apollo 15 Mission Report," December 1971.

2. SNA-8-D-027 (III), Rev. 3., "CSM/LM Spacecraft Operational Data Book," Vol. III, Mass Properties, 31 April 1971.

3. TRW Technical Report 17618-H169-R0-00, "Apollo Mission J1/LM-10/DPS Preflight Performance Report," A. T. Avvenire, May 1971.

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.

6. TRW Letter 71.4915.2-35, "Apollo 15 Primary Propulsion Systems Preliminary Flight Evaluation," R. L. Morris, 30 August 1971.

7. TRW IOC 72.4910.41-28, "Parallel Tank Flow Model For Use In LM Descent Prediction," C. R. Savino, 22 February 1972.

TABLE 1

LM-10 DESCENT PROPULSION ENGINE AND

FEED SYSTEM PHYSICAL CHARACTERISTICS

ENGINE	
Engine Number	1046
Chamber Throat Area, in^2	53.495[1]
Nozzle Exit Area, in^2	2937.6[3]
Nozzle Expansion Ratio	54.0[3]
FEED SYSTEM	
Oxidizer Propellant Tanks, Total	
Ambient[4] Volume, Ft^3	135.4[3]
Fuel Propellant Tanks, Total	
Ambient Volume, Ft^3	135.4[3]
Oxidizer Tank to Interface	
Resistance, $\frac{lbf\text{-}sec^2}{lbm\text{-}ft^5}$	413.194[2]
Fuel Tank to Interface	
Resistance, $\frac{lbf\text{-}sec^2}{lbm\text{-}ft^5}$	672.674[2]

[1]TRW No. 01827-6281-T0-00, TRW LEM Descent Engine Serial No. 1046 Acceptance Test Performance Report Paragraph 6.10, dated 5 March 1970.

[2]GAEC Cold Flow Tests.

[3]Approximate Values

[4]14.7 PSIA and 70°F

TABLE 2

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
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
GG0001X	PGNS Downlink Data	Digital Code	50

TABLE 3

DESCENT PROPULSION SYSTEM STEADY-STATE FTP PERFORMANCE

PARAMETER	FS-1 + 51 SECONDS			FS-1 + 431 SECONDS		
INSTRUMENTED	PREDICTED	MEASURED	CALCULATED	PREDICTED	MEASURED	CALCULATED
Regulator Outlet Pressure, psia	245.9	247.0	246.5	245.5	246.6	246.1
Oxidizer Interface Pressure, psia	226.2	225.3	226.8	225.1	225.1	225.8
Fuel Interface Pressure, psia	225.8	223.0	226.4	224.9	223.2	225.6
Engine Chamber Pressure, psia	105.0	105.6	105.3		104.1	100.6
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	19.89	---	19.91	20.20	---	20.29
Fuel Flowrate, lbm/sec	12.45	---	12.47	12.67	---	12.73
Propellant Mixture Ratio	1.597	---	1.597	1.594	---	1.594
Vacuum Specific Impulse, sec	306.3	---	306.3	303.5	---	303.8
Vacuum Thrust, lbf	9905	---	9921	9984	---	10032
Throat Erosion, %	-1.21	---	-1.35	6.27	---	5.95

TABLE 4

LM-10 DESCENT PROPULSION SYSTEM THROTTLED PERFORMANCE

PARAMETER	FS-1 + 466 Seconds			FS-1 + 606 Seconds		
INSTRUMENTED	Predicted	Measured	Simulation	Predicted	Measured	Simulation
Regulator Outlet Pressure, psia	246.0	246.7	246.2	246	245.7	245.2
Oxidizer Interface Pressure, psia	236.6	237.5	236.6	239.1	238.5	238.0
Fuel Interface Pressure, psia	236.5	235.5	236.6	239.1	237.0	238.0
Engine Chamber Pressure, psia	60.0	-	60.2	45.6	-	45.4
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	8.37	8.37	-	6.69	6.69
DERIVED						
Oxidizer Flowrate, lbm/sec	11.98	----	11.86	9.36	----	9.23
Fuel Flowrate, lbm/sec	7.56	----	7.49	5.91	----	5.83
Propellant Mixture Ratio, O/F	1.585	----	1.584	1.585	----	1.584
Vacuum Specific Impulse, sec	304.4	----	304.4	298.1	----	299.7
Vacuum Thrust, lbf	5948	----	5890	4581	----	4511.
Throat Erosion, %	7.09	----	6.30	9.22	----	8.02

TABLE 5

LM-10 DPS PROPELLANT QUANTITY GAGING SYSTEM PERFORMANCE

Parameter	Time (From Descent Burn Ignition) sec							
	71	171	271	371	431	531	631	739
Oxidizer Tank No. 1								
Measured Quantity, percent	96.7	79.4	61.4	43.4	32.8	21.6	12.4	5.9
Calculated Quantity, percent	96.0	78.5	61.0	43.4	32.8	21.8	12.9	6.2
Difference, percent	0.7	0.9	0.4	0.0	0.0	-0.2	-0.5	-0.3
Oxidizer Tank No. 2								
Measured Quantity, percent	97.0	79.6	61.4	43.6	33.2	22.2	13.1	6.4
Calculated Quantity, percent	96.0	78.5	61.0	43.4	32.8	21.8	12.9	6.2
Difference, percent	1.0	1.1	0.4	0.2	0.4	0.4	0.2	0.2
Fuel Tank No. 1								
Measured Quantity, percent	93.3	77.2	60.7	43.2	32.8	21.7	12.6	6.1
Calculated Quantity, percent	96.4	78.9	61.4	43.7	33.1	22.1	13.0	6.3
Difference, percent	-3.1	-1.7	-0.7	-0.5	-0.3	-0.4	-0.4	-0.2
Fuel Tank No. 2								
Measured Quanity, percent	95.0	79.3	62.0	44.0	33.6	22.7	13.7	7.0
Calculated Quantity, percent	96.4	78.9	61.4	43.7	33.1	22.1	13.0	6.3
Difference, percent	-1.4	0.4	0.6	0.3	0.5	0.6	0.7	0.7

TABLE 6

DPS START AND SHUTDOWN IMPULSE SUMMARY

	Apollo 15 LM-10/DPS-1	Apollo 14 LM-8/DPS-1	Apollo 12 LM-6/DPS-2	Apollo 10 LM-4/DPS-2	SPECIFICATION LIMITS
	STARTS				
Steady-State Throttle Position, Percent	13.1	13.1	16.2	13.1	
Total Vacuum Start Impulse (FS-1 to 90% steady-state), lbf-sec	440	710	591	728	
Start Time (FS-1 to 90% steady-state), sec	2.34	2.14	1.77	2.13	4.0
Coast Time from Prior Burn, Minutes	From Launch	From Launch	56	72	
	SHUTDOWNS				
Steady-State Throttle Position, Percent	29.0	27.0	23.4	FTP	
Total Vacuum Shutdown Impulse (FS-2 to 10% Steady-State), lbf-sec	1113	976	1540	2041	
Shutdown Time (FS-2 to 10% Steady-State), sec	2.06	1.23	2.06	0.34	0.25[4]
Repeatability, lbf-sec					±100[4]
Total Vacuum Shutdown Impulse (FS-2 to Zero Thrust) from Velocity Gain Data, lbf-sec	---[5]	---[5]	---[5]	2948	

[4]Specification value for shutdowns performed from FTP only.

[5]Not applicable to lunar landing shutdown.

TABLE 6a

DPS START AND SHUTDOWN IMPULSE SUMMARY

	Apollo 9 LM-3/DPS-1	Apollo 9 LM-3/DPS-2	Apollo 9 LM-3/DPS-3	Apollo 5 LM-3/DPS-3	Apollo 5 LM-1/DPS-3	SPECIFICATION LIMITS
STARTS						
Steady-State Throttle Position, Percent	12.7	12.7	12.7	12.4	12.4	
Total Vacuum Start Impulse (FS-1 to 90% steady-state), lbf-sec	805	1029	950	894	574	
Start Time (FS-1 to 90% steady-state), sec	2.5[1]	2.1	2.3[1]	2.66	2.13	4.0
Coast Time from Prior Burn, Minutes	From Launch	2640	111	131	0.5	
SHUTDOWNS						
Steady-State Throttle Position, Percent	40	40	12.7	FTP	FTP	
Total Vacuum Shutdown Impulse (FS-2 to 10% Steady-State), lbf-sec	---[2]	1730	748	1727	1713	
Shutdown Time (FS-2 to 10% Steady-State), sec	1.1[1]	1.1	1.8[1]	0.26	0.30	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	1777	---[2]	1040	2493	---[3]	

[1] Reference 5.

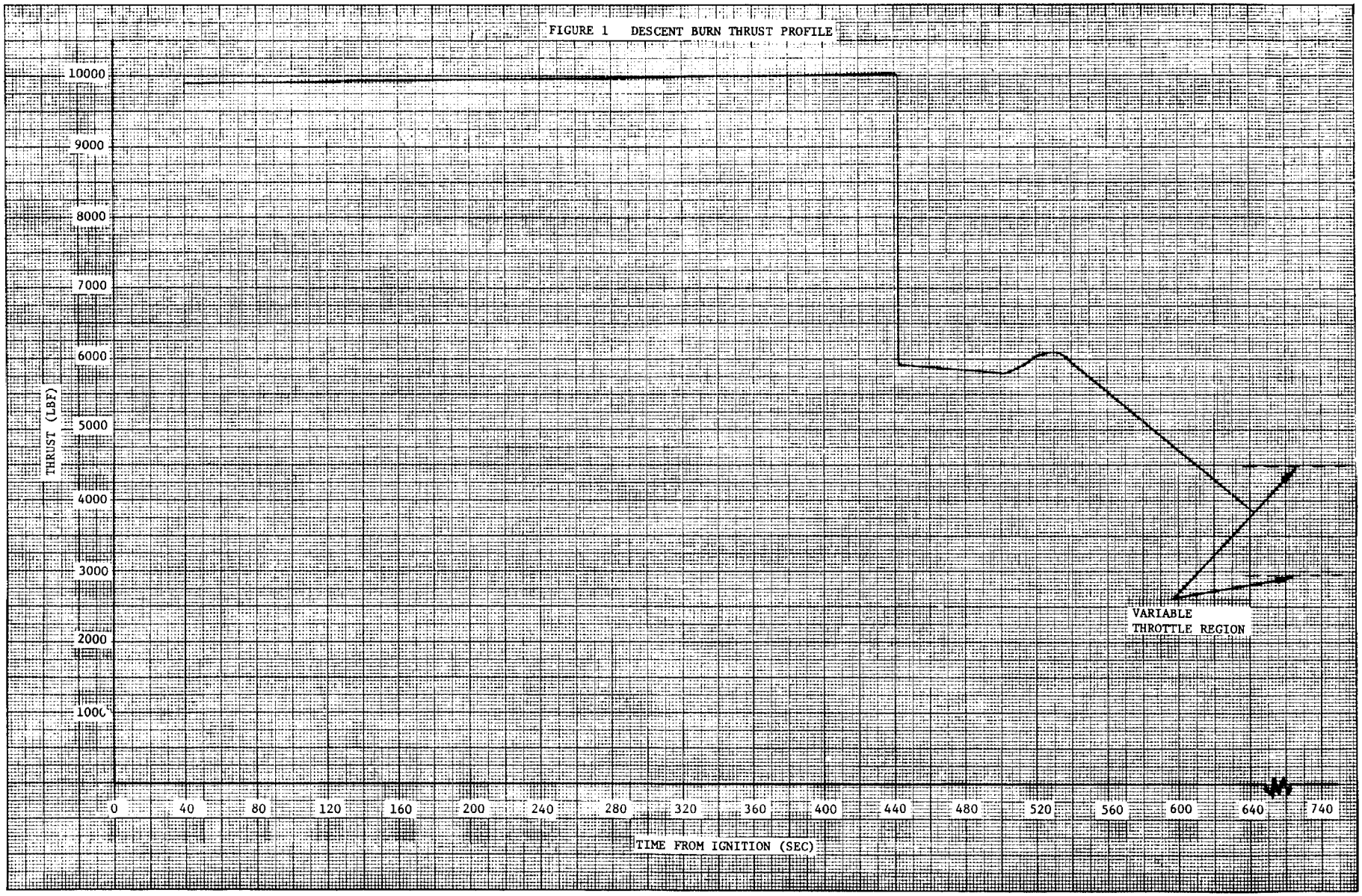
[2] Data Unavailable.

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

[4] Specification value for shutdowns performed from FTP only.

[5] Not applicable to lunar landing shutdown

FIGURE 1 DESCENT BURN THRUST PROFILE



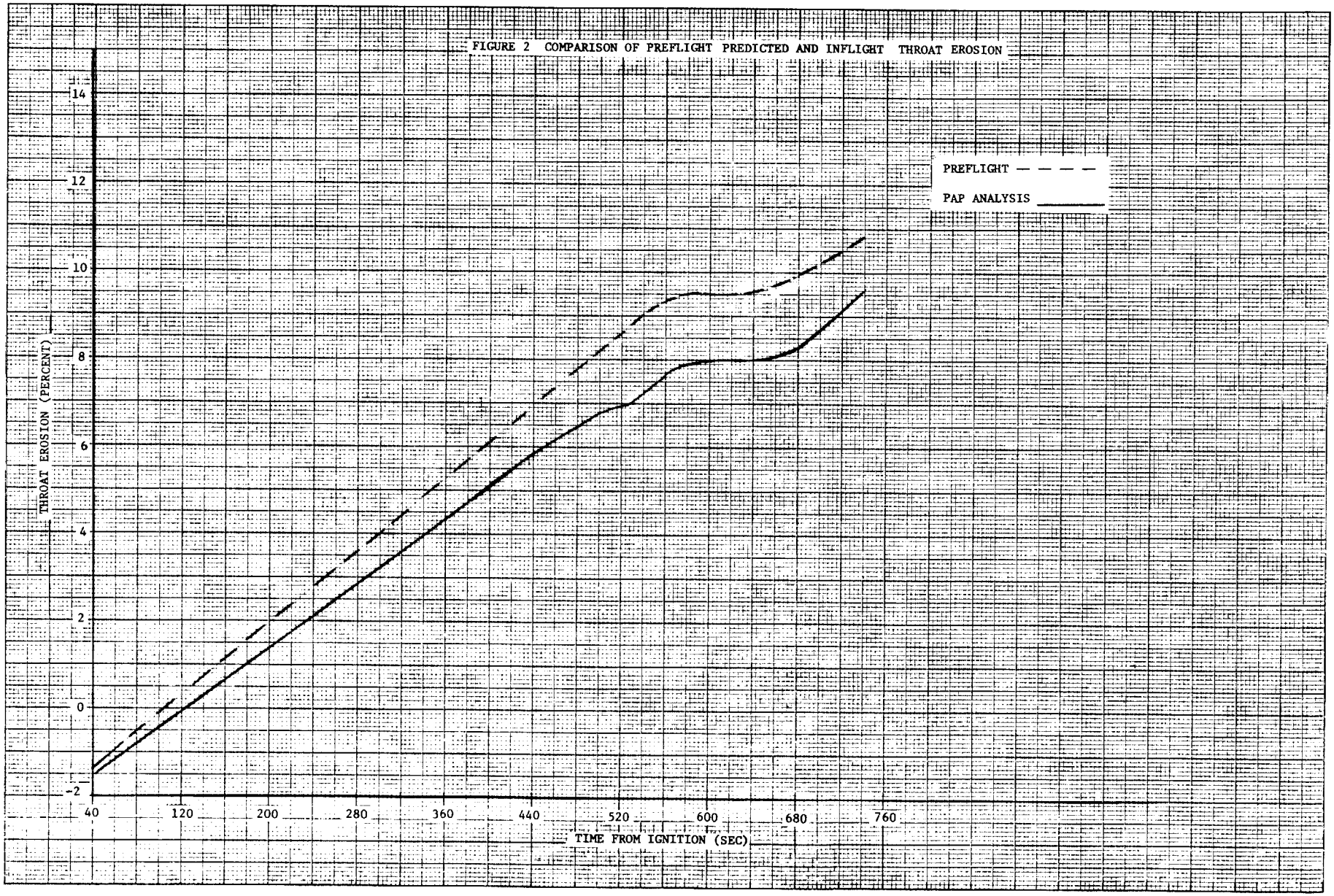
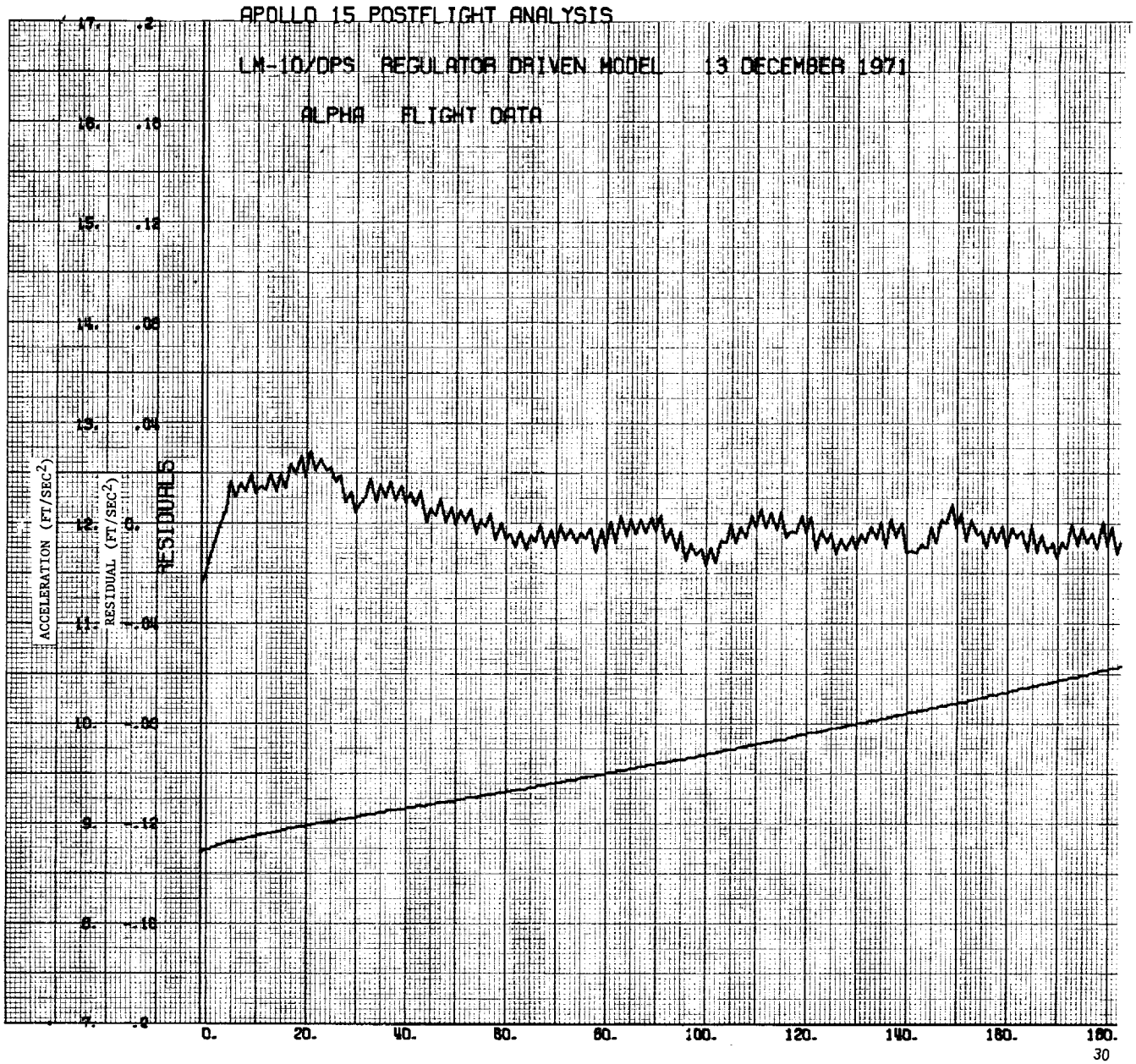


FIGURE 2 COMPARISON OF PREFLIGHT PREDICTED AND INFLIGHT THROAT EROSION
PREFLIGHT
PAP ANALYSIS
THROAT EROSION (PERCENT)
14
12
10
8
6
4
2
0
-2
40
120
200
280
360
440
520
600
680
760
TIME FROM IGNITION (SEC)


APOLLO 15 POSTFLIGHT ANALYSIS
LM-10/DPS REGULATOR DRIVEN MODEL 13 DECEMBER 1971
ALPHA FLIGHT DATA
RESIDUALS
ACCELERATION (FT/SEC2)
RESIDUAL (FT/SEC2)
17. .2
16. .16
15. .12
14. .08
13. .04
12. 0.
11. -.04
10. -.08
9. -.12
8. -.16
7. -.2
0.
20.
40.
60.
80.
100.
120.
140.
160.
180.

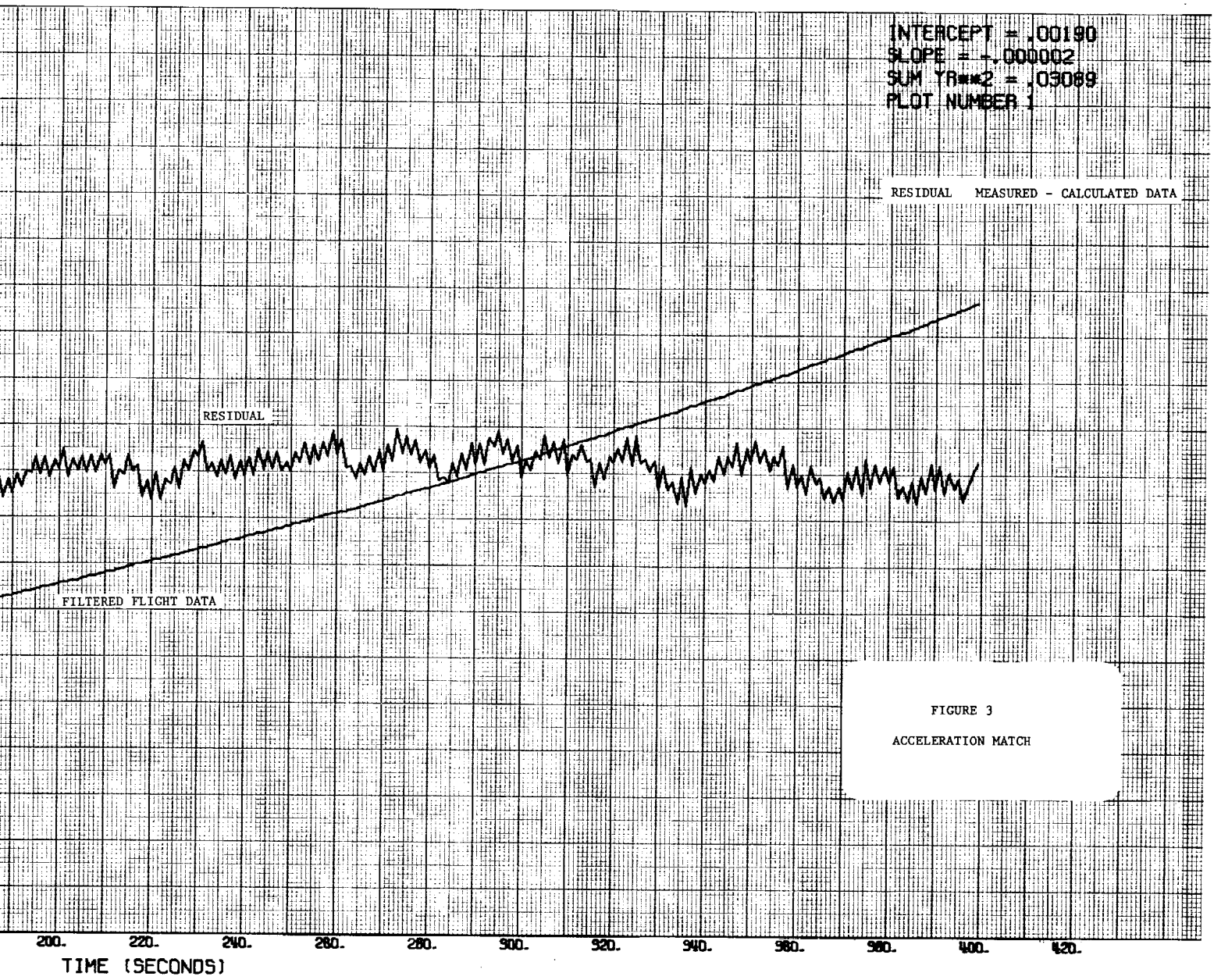
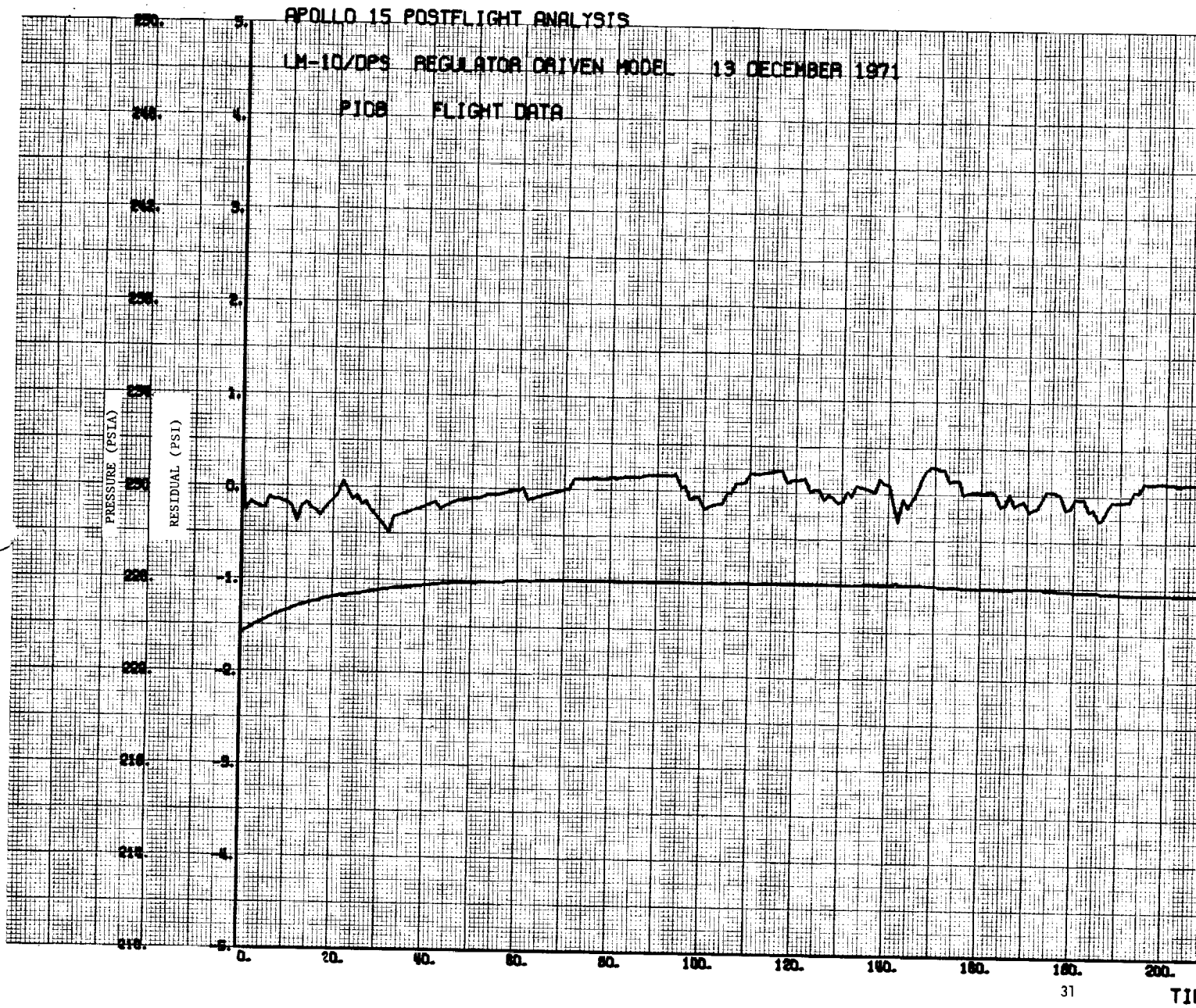


FIGURE 3

ACCELERATION MATCH


APOLLO 15 POSTFLIGHT ANALYSIS
LM-10/DPS REGULATOR DRIVEN MODEL 13 DECEMBER 1971
PIOB FLIGHT DATA
PRESSURE (PSIA)
RESIDUAL (PSI)
250.
248.
246.
244.
242.
240.
238.
236.
234.
232.
230.
228.
226.
224.
222.
220.
218.
216.
214.
212.
210.
5.
4.
3.
2.
1.
0.
-1.
-2.
-3.
-4.
-5.
0.
20.
40.
60.
80.
100.
120.
140.
160.
180.
200.
TI

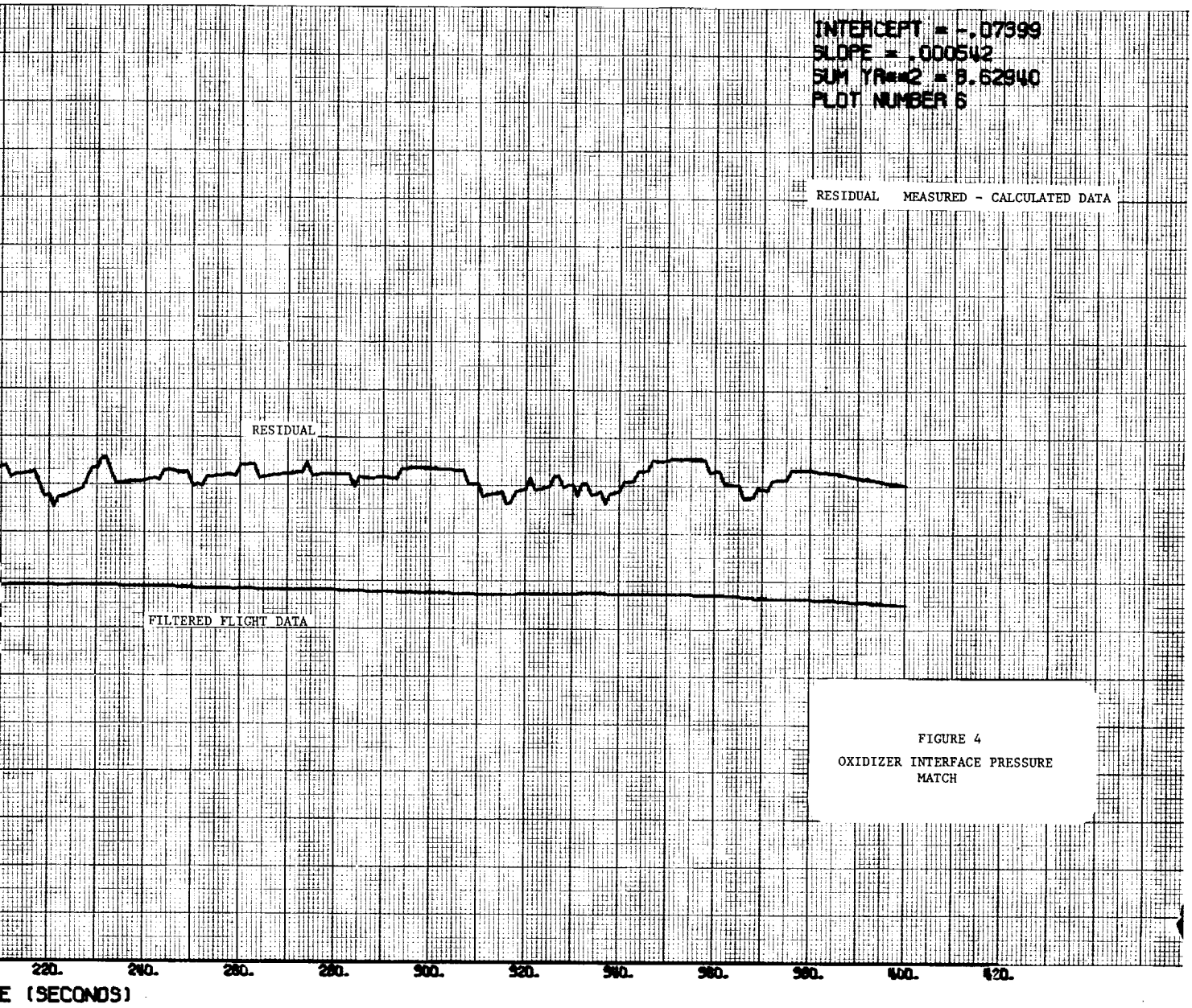
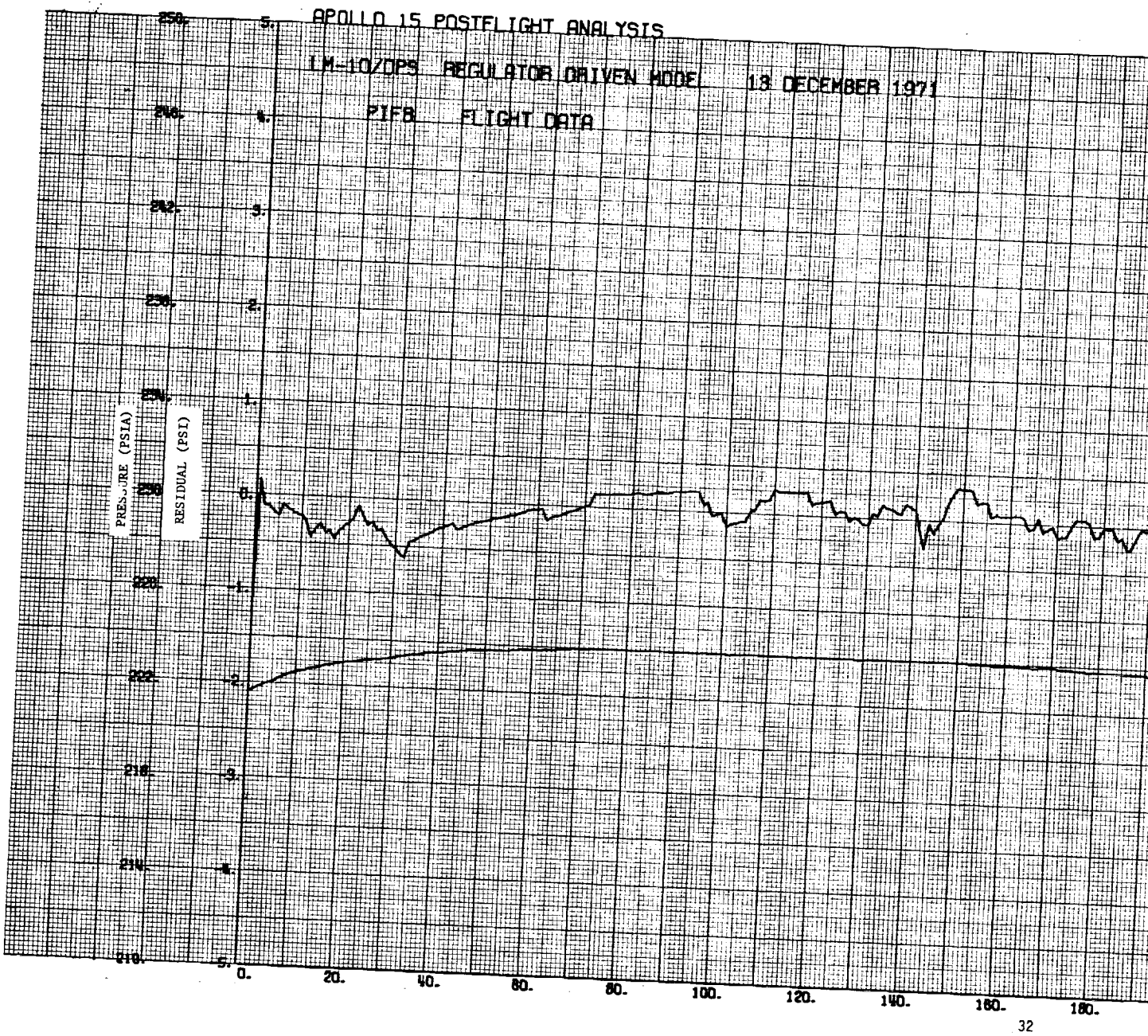


FIGURE 4

OXIDIZER INTERFACE PRESSURE MATCH


APOLLO 15 POSTFLIGHT ANALYSIS
LM-10/DPS REGULATOR DRIVEN MODEL 13 DECEMBER 1971
PIFB FLIGHT DATA
PRESSURE (PSIA)
RESIDUAL (PSI)
250.
246.
242.
238.
234.
230.
226.
222.
218.
214.
210.
5.
4.
3.
2.
1.
0.
-1.
-2.
-3.
-4.
-5.
0.
20.
40.
60.
80.
100.
120.
140.
160.
180.

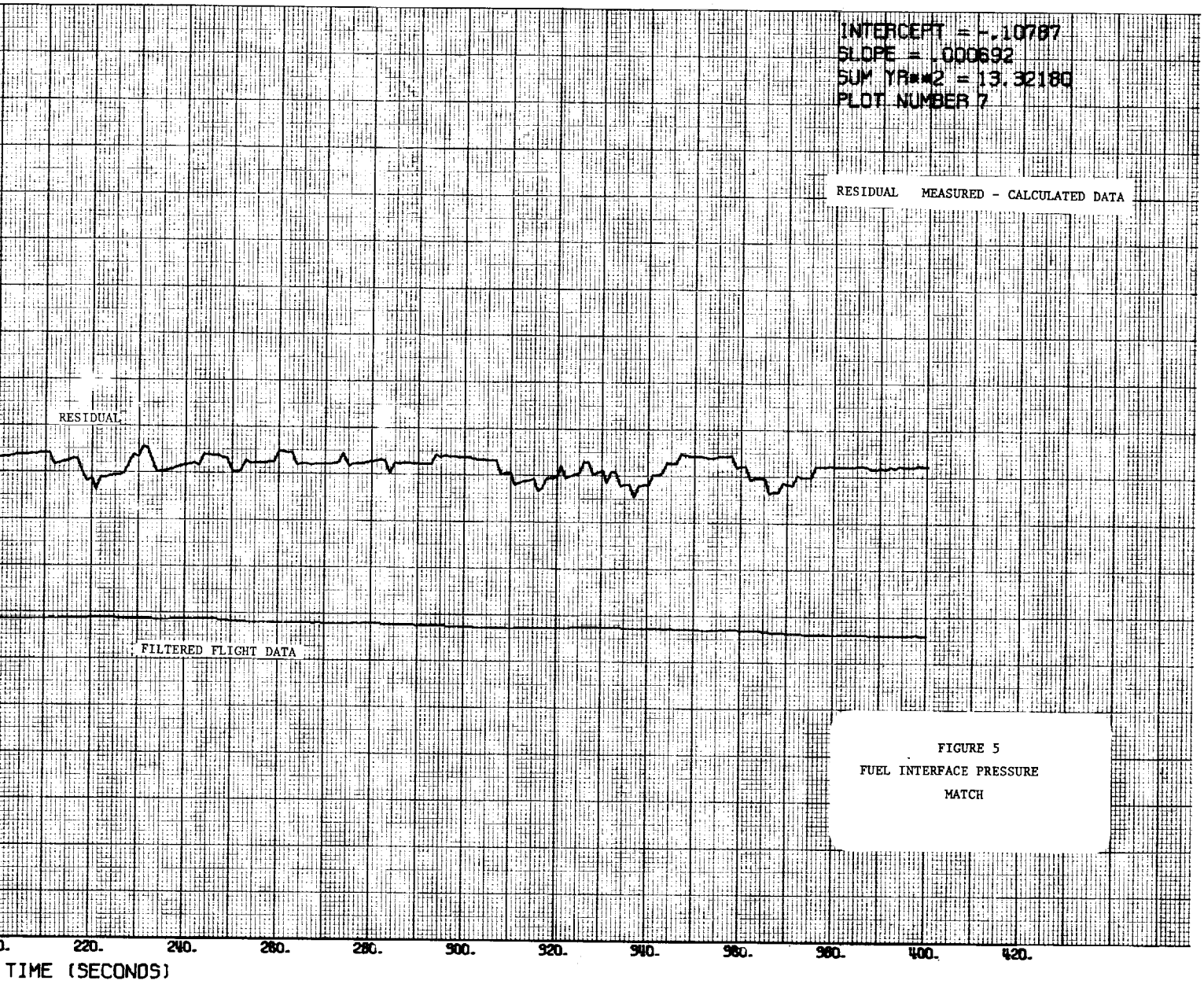
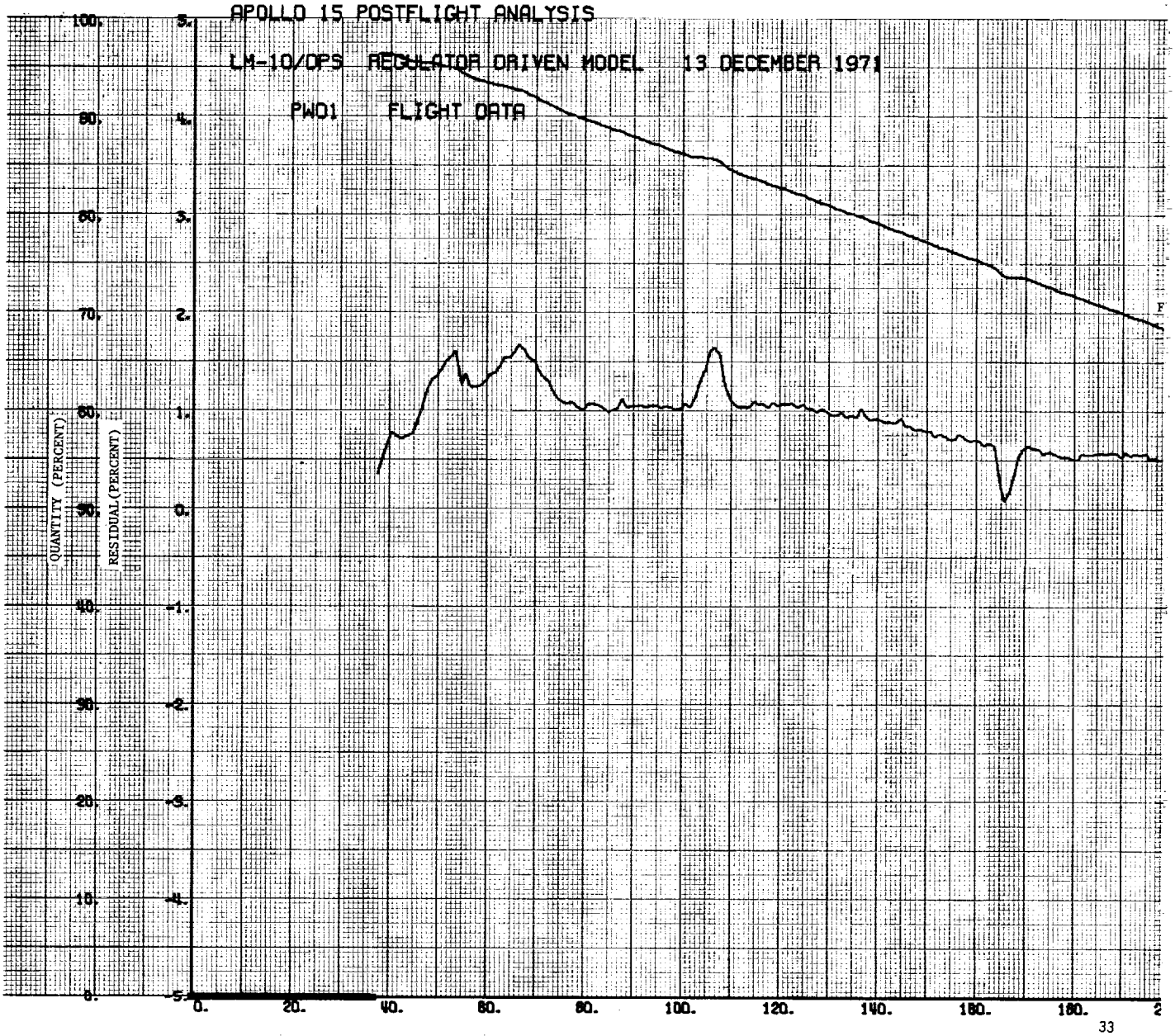


FIGURE 5
FUEL INTERFACE PRESSURE
MATCH


APOLLO 15 POSTFLIGHT ANALYSIS
LM-10/DPS REGULATOR DRIVEN MODEL 13 DECEMBER 1971
PWD1 FLIGHT DATA
QUANTITY (PERCENT)
RESIDUAL (PERCENT)
100.
80.
60.
70.
60.
50.
40.
30.
20.
10.
0.
5.
4.
3.
2.
1.
0.
-1.
-2.
-3.
-4.
-5.
0.
20.
40.
60.
80.
100.
120.
140.
160.
180.

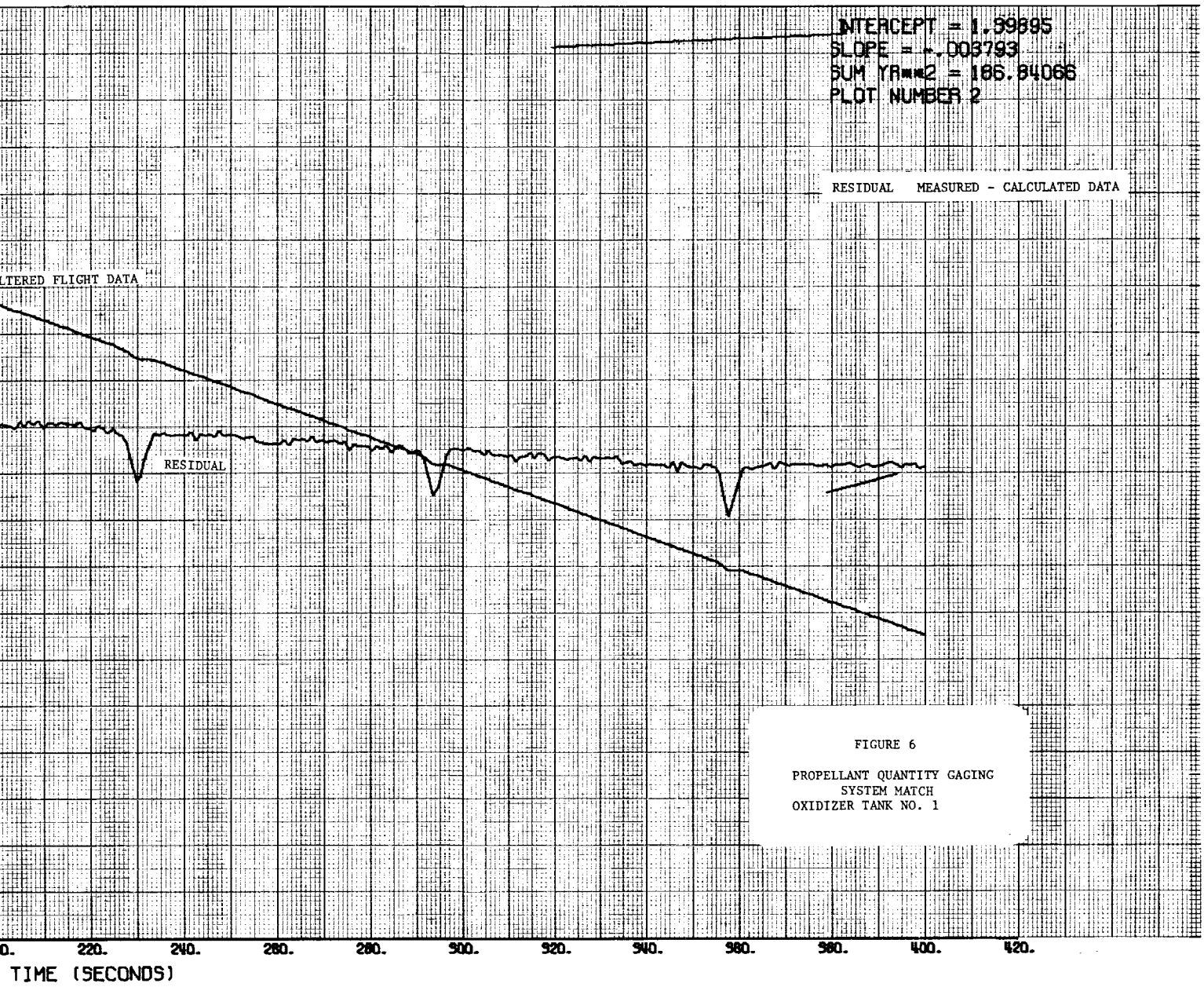
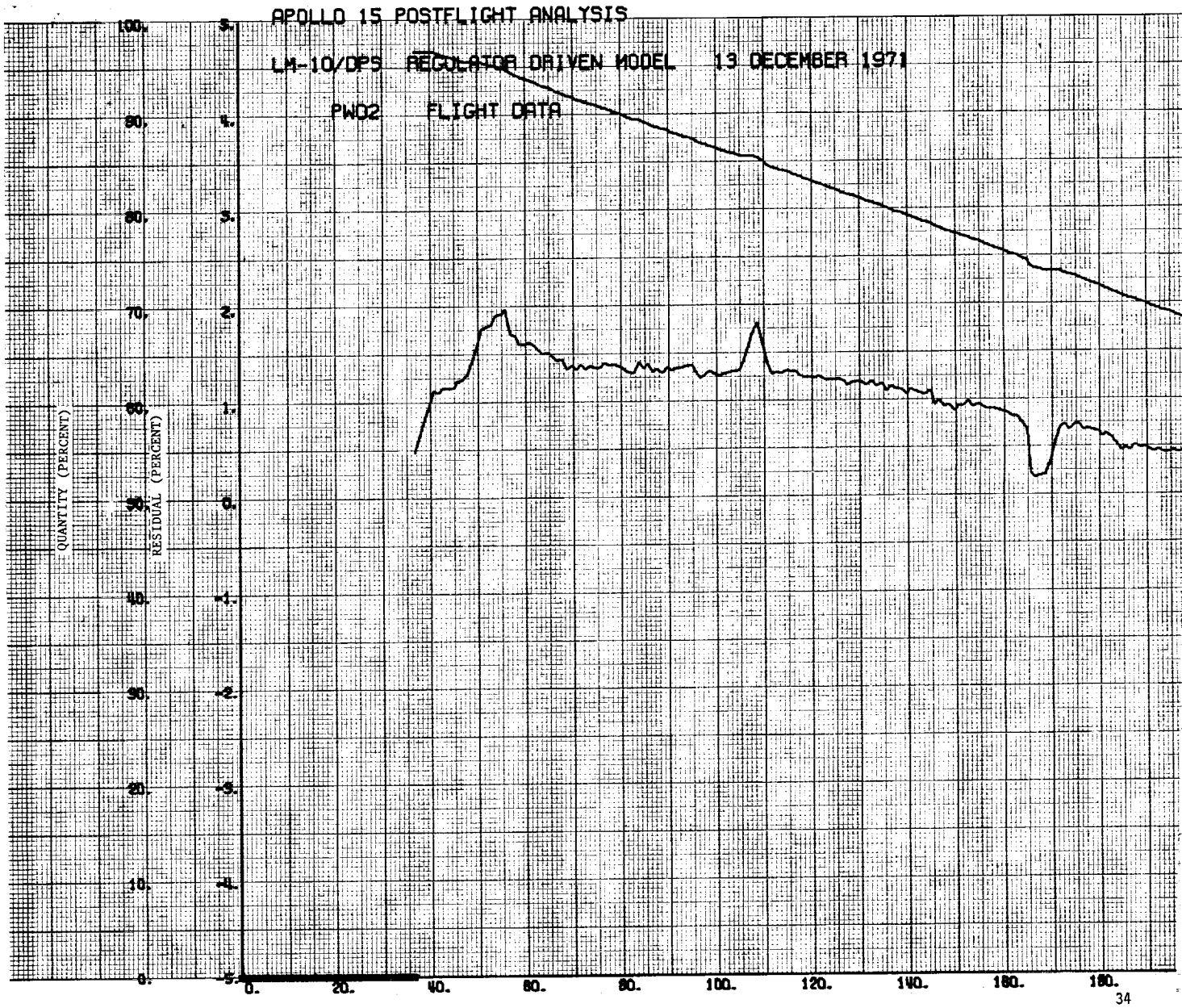


FIGURE 6

PROPELLANT QUANTITY GAGING
SYSTEM MATCH
OXIDIZER TANK NO. 1


APOLLO 15 POSTFLIGHT ANALYSIS
LM-10/DPS REGULATOR DRIVEN MODEL 13 DECEMBER 1971
PW02 FLIGHT DATA
QUANTITY (PERCENT)
RESIDUAL (PERCENT)
100.
90.
80.
70.
60.
50.
40.
30.
20.
10.
0.
5.
4.
3.
2.
1.
0.
-1.
-2.
-3.
-4.
-5.
0.
20.
40.
60.
80.
100.
120.
140.
160.
180.

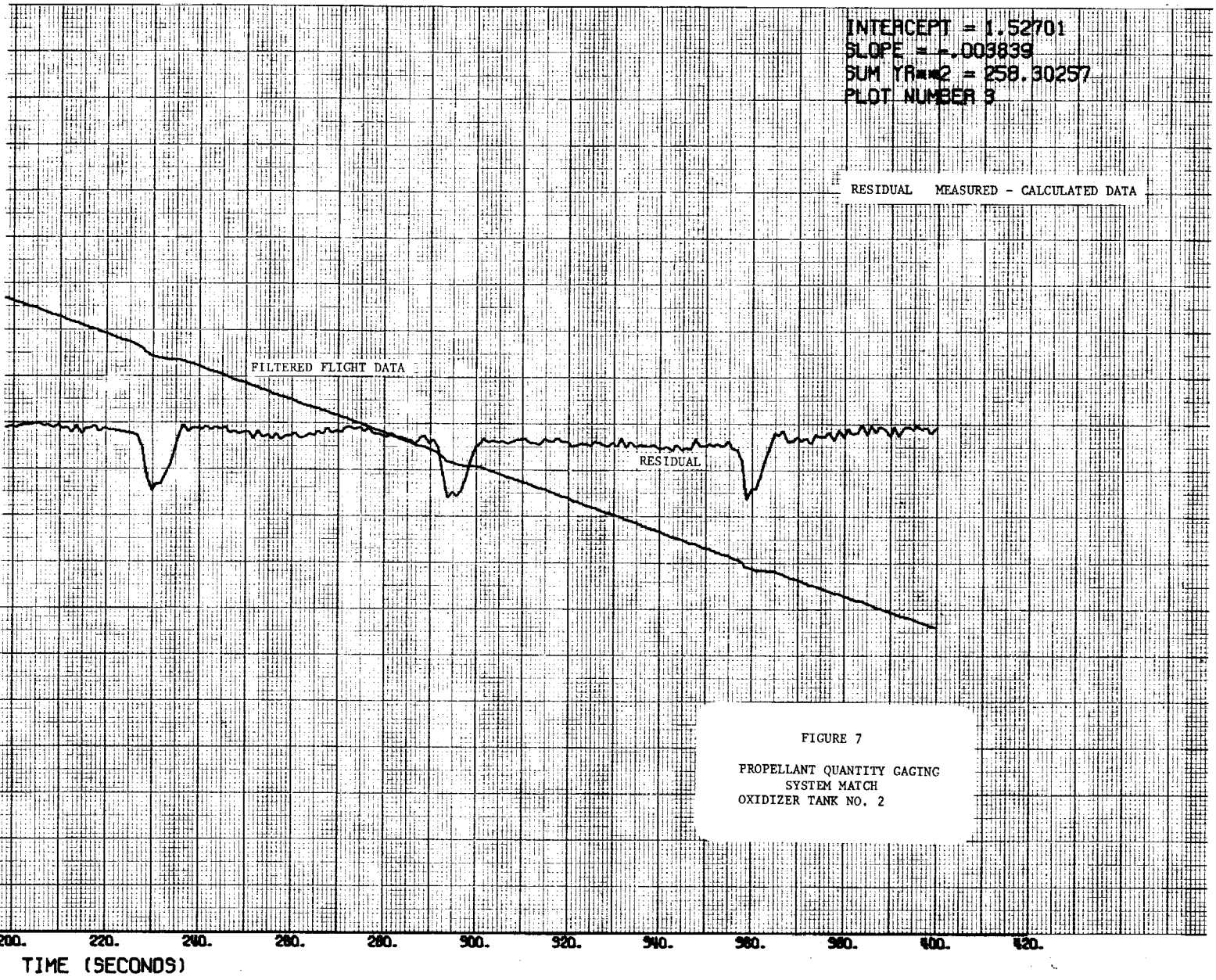
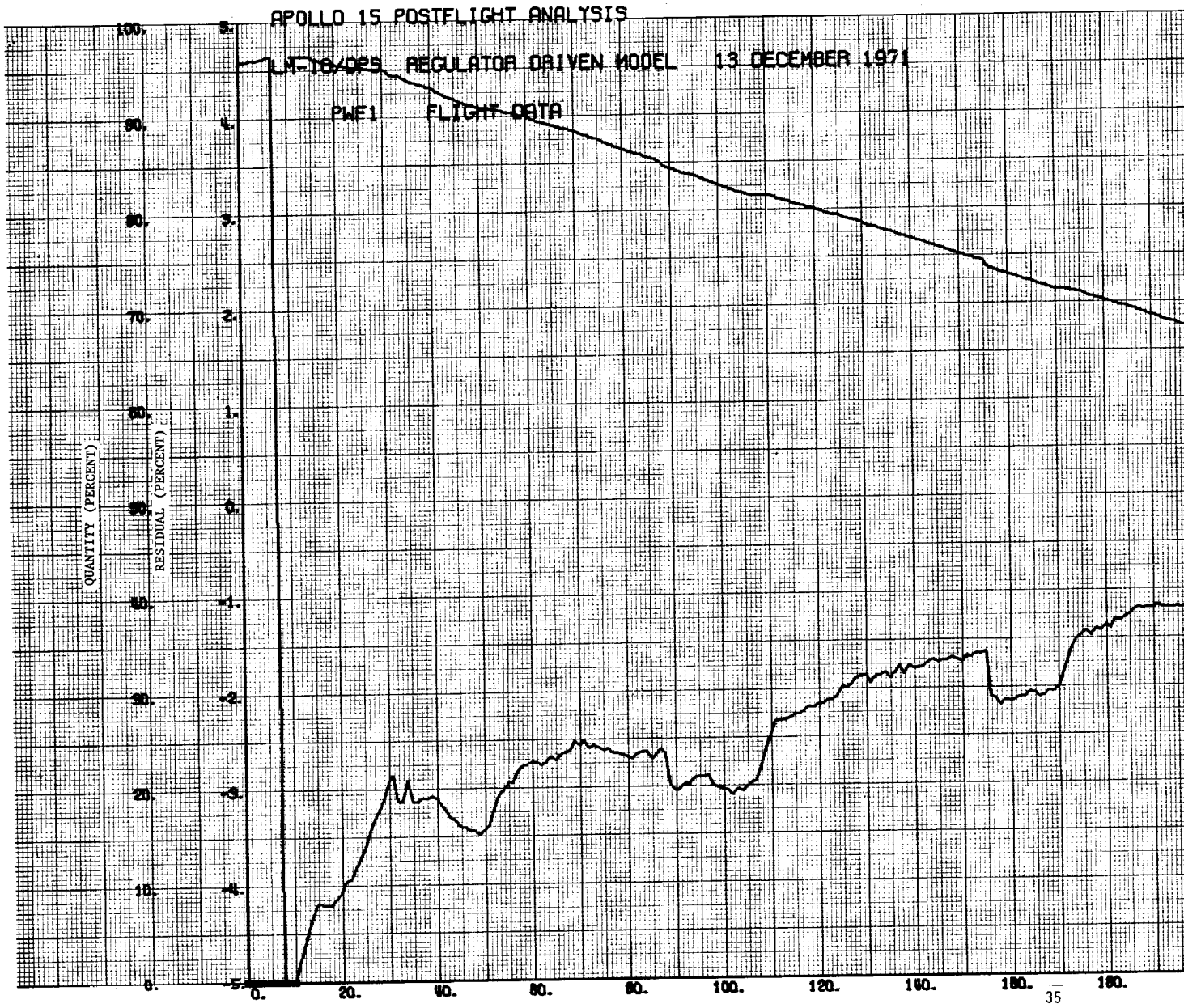


FIGURE 7

PROPELLANT QUANTITY GAGING
SYSTEM MATCH
OXIDIZER TANK NO. 2


APOLLO 15 POSTFLIGHT ANALYSIS
LM-10/DPS REGULATOR DRIVEN MODEL 13 DECEMBER 1971
PWF1 FLIGHT DATA
QUANTITY (PERCENT)
RESIDUAL (PERCENT)
100.
90.
80.
70.
60.
50.
40.
30.
20.
10.
0.
5.
4.
3.
2.
1.
0.
-1.
-2.
-3.
-4.
-5.
0.
20.
40.
60.
80.
100.
120.
140.
160.
180.

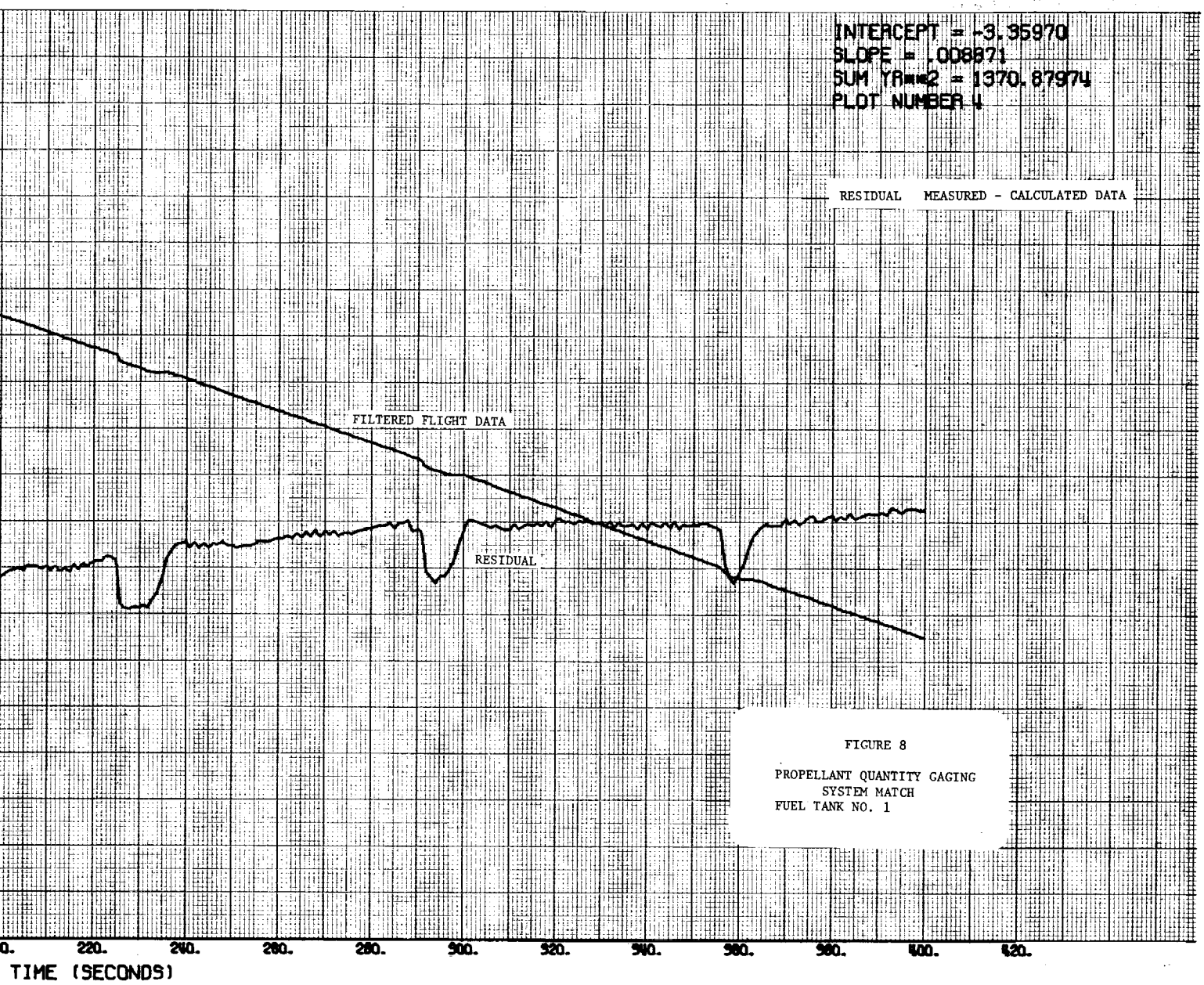
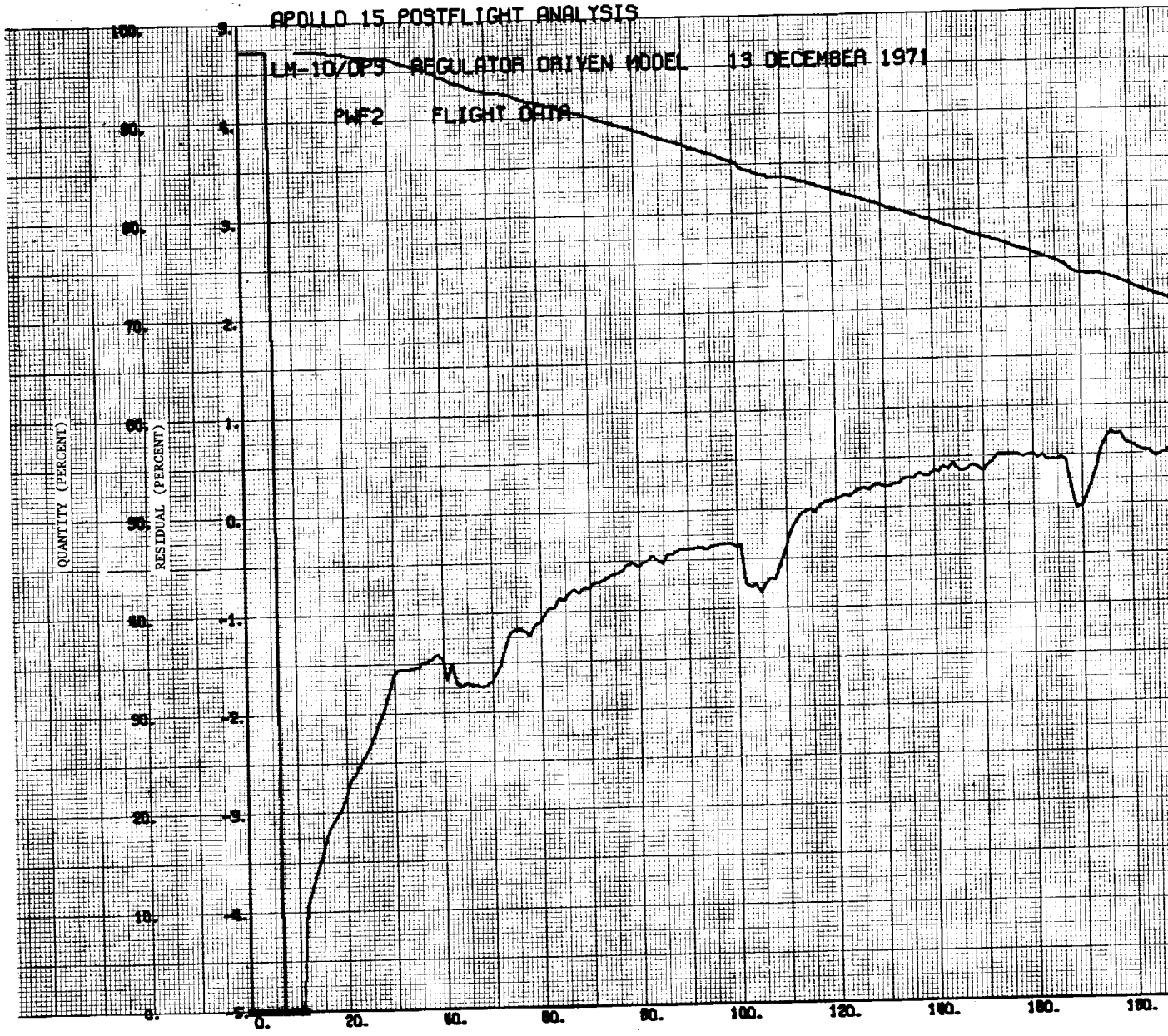


FIGURE 8

PROPELLANT QUANTITY GAGING
SYSTEM MATCH
FUEL TANK NO. 1


APOLLO 15 POSTFLIGHT ANALYSIS
LM-10/DPS REGULATOR DRIVEN MODEL 13 DECEMBER 1971
PWF2 FLIGHT DATA
QUANTITY (PERCENT)
RESIDUAL (PERCENT)
100.
90.
80.
70.
60.
50.
40.
30.
20.
10.
0.
3.
2.
1.
0.
-1.
-2.
-3.
-4.
-5.
0.
20.
40.
60.
80.
100.
120.
140.
160.
180.

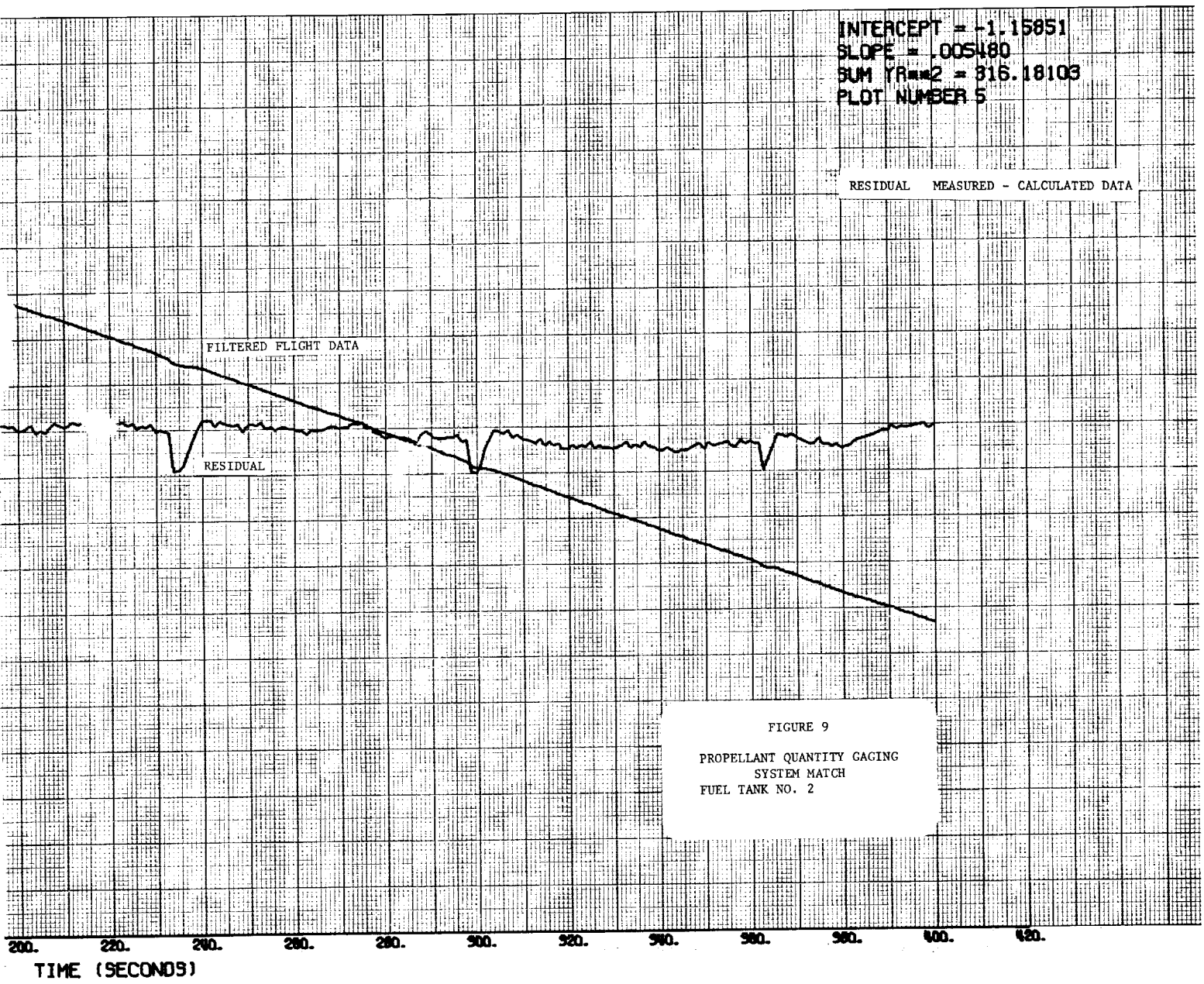
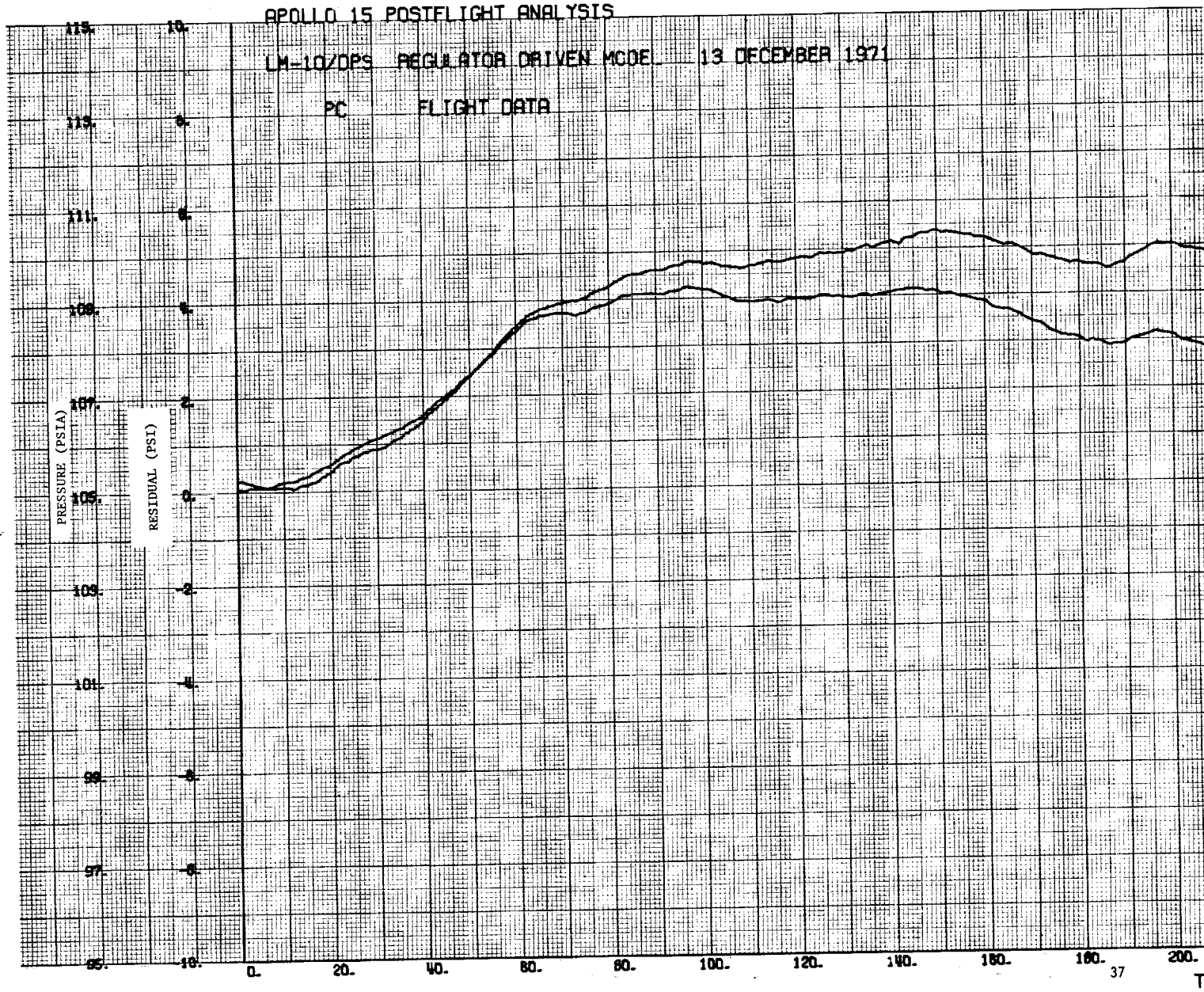


FIGURE 9

PROPELLANT QUANTITY GAGING
SYSTEM MATCH
FUEL TANK NO. 2


APOLLO 15 POSTFLIGHT ANALYSIS
LM-10/DPS REGULATOR DRIVEN MODEL 13 DECEMBER 1971
PC FLIGHT DATA
PRESSURE (PSIA)
RESIDUAL (PSI)
115.
113.
111.
109.
107.
105.
103.
101.
99.
97.
95.
10.
8.
6.
4.
2.
0.
-2.
-4.
-6.
-8.
-10.
0.
20.
40.
60.
80.
100.
120.
140.
160.
180.
200.
T

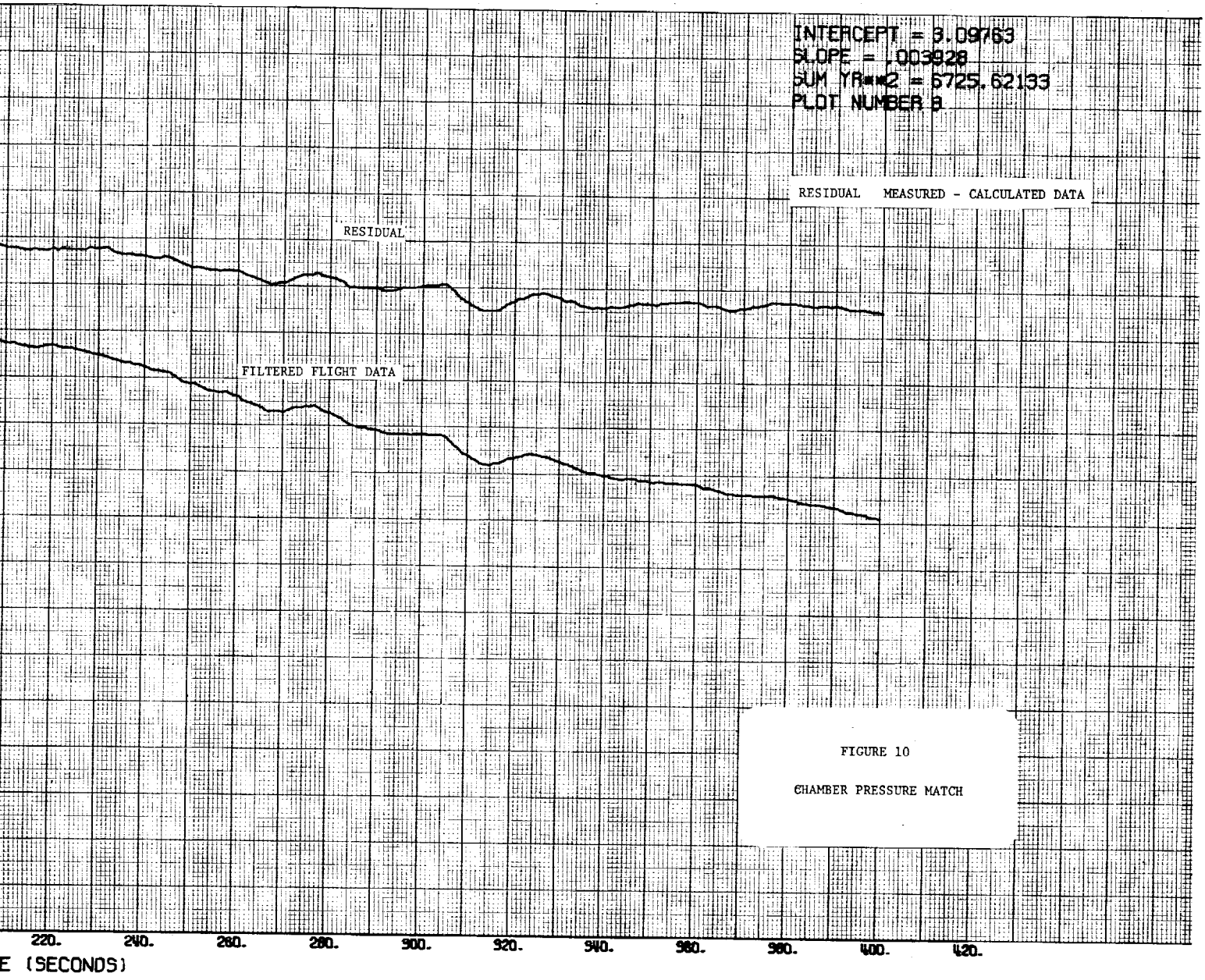


FIGURE 10

CHAMBER PRESSURE MATCH

FIGURE 11 COMPARISON OF PREFLIGHT PREDICTED AND INFLIGHT PERFORMANCE FTP

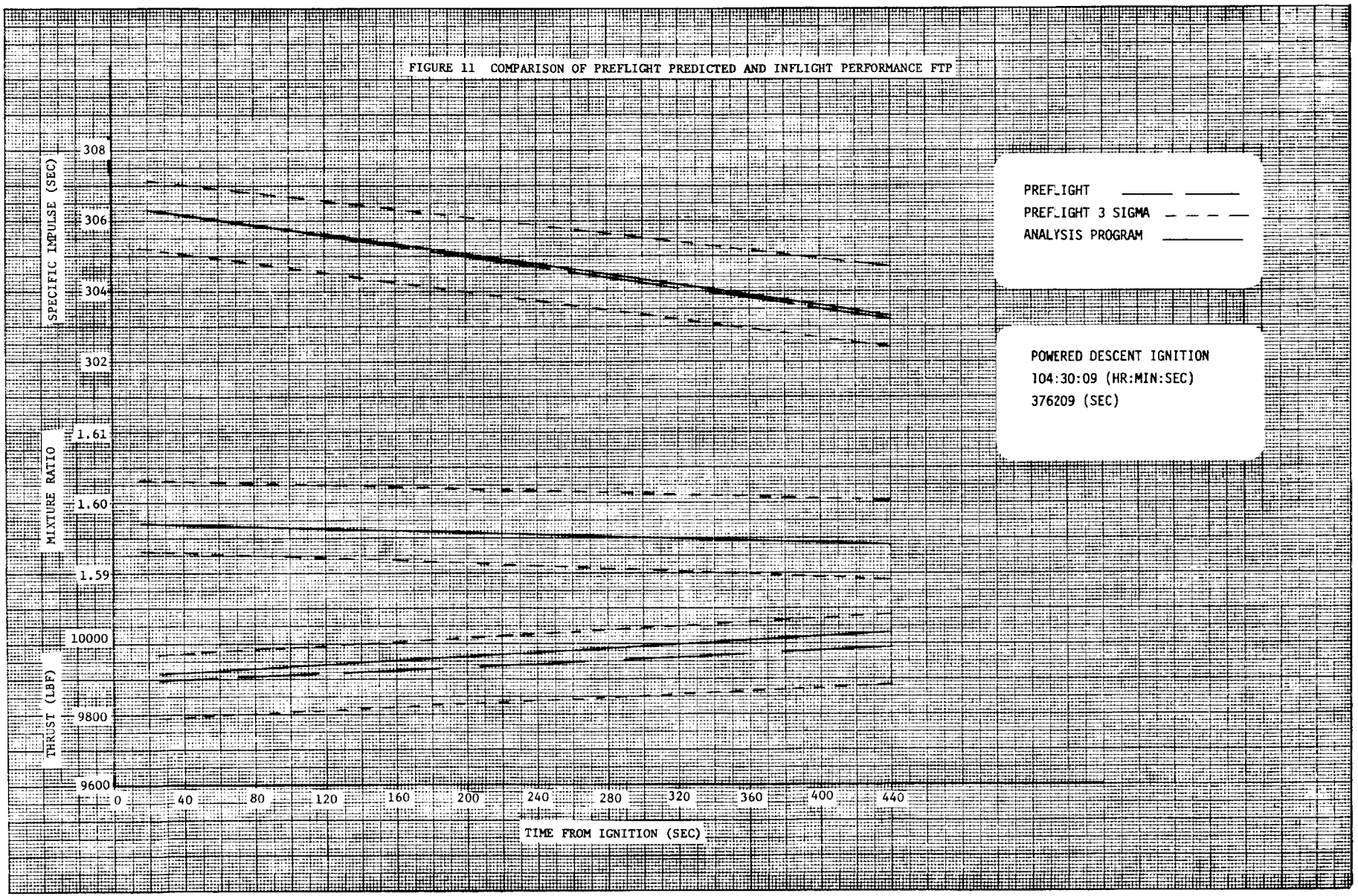
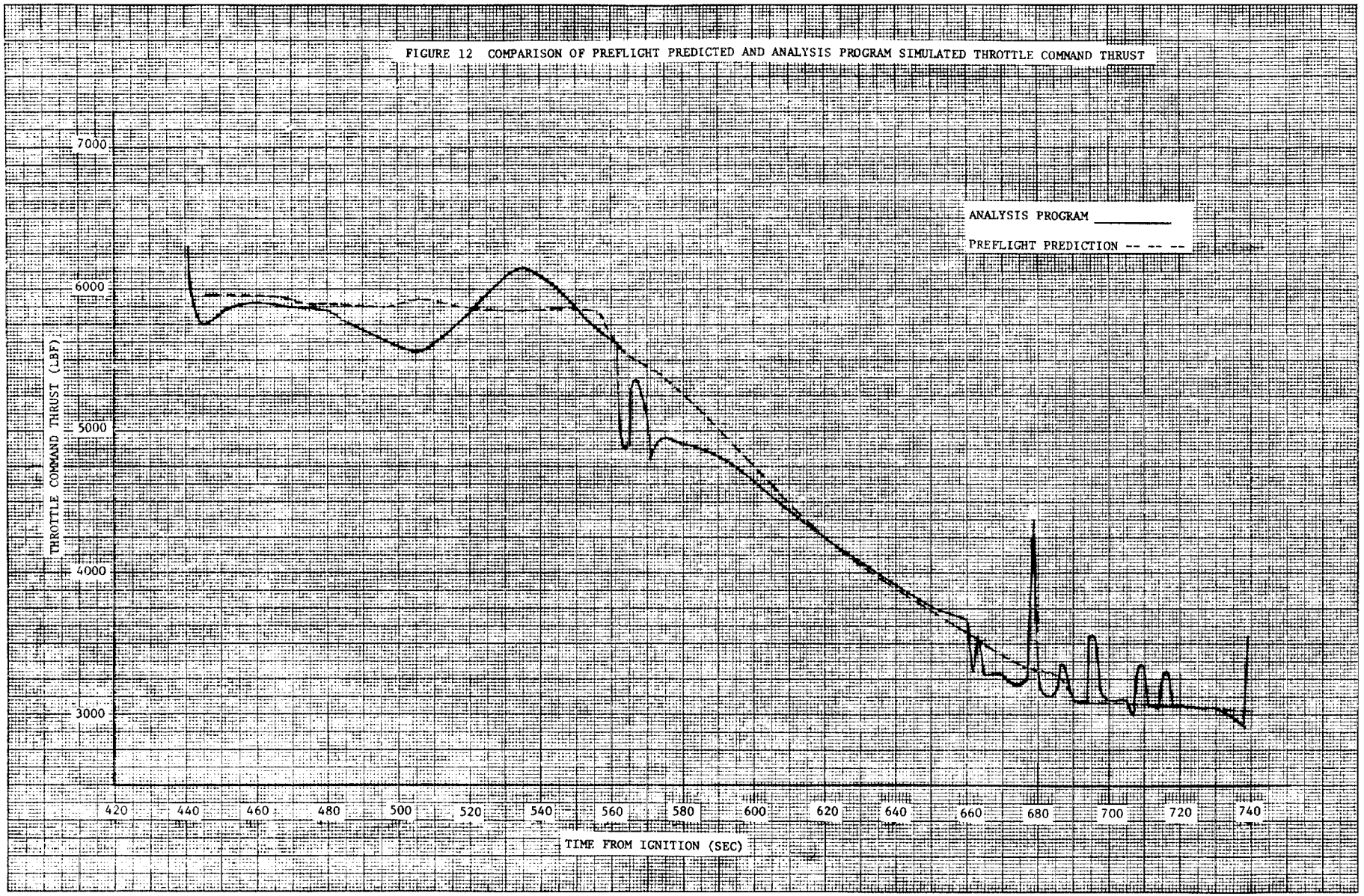


FIGURE 12 COMPARISON OF PREFLIGHT PREDICTED AND ANALYSIS PROGRAM SIMULATED THROTTLE COMMAND THRUST



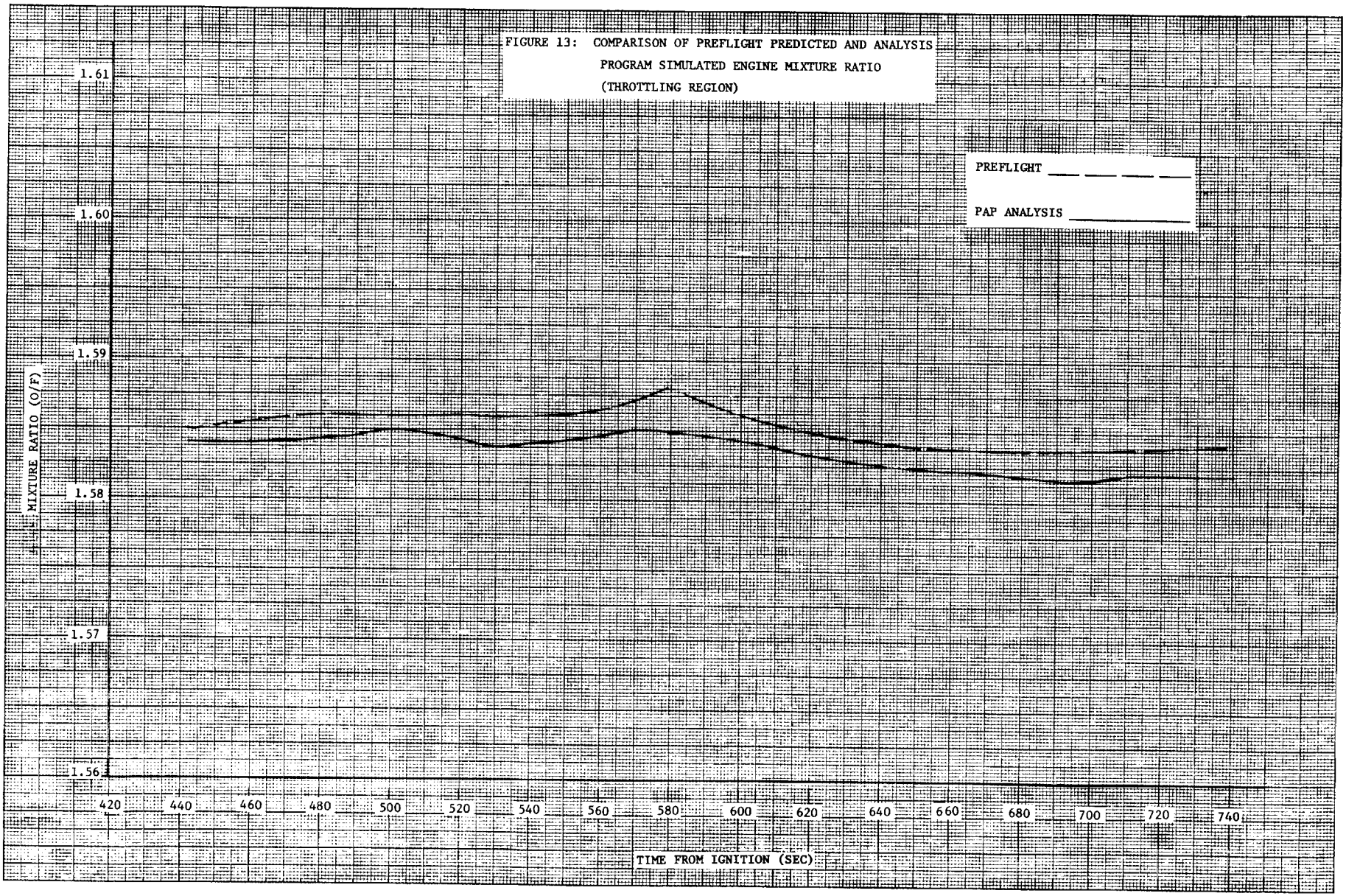
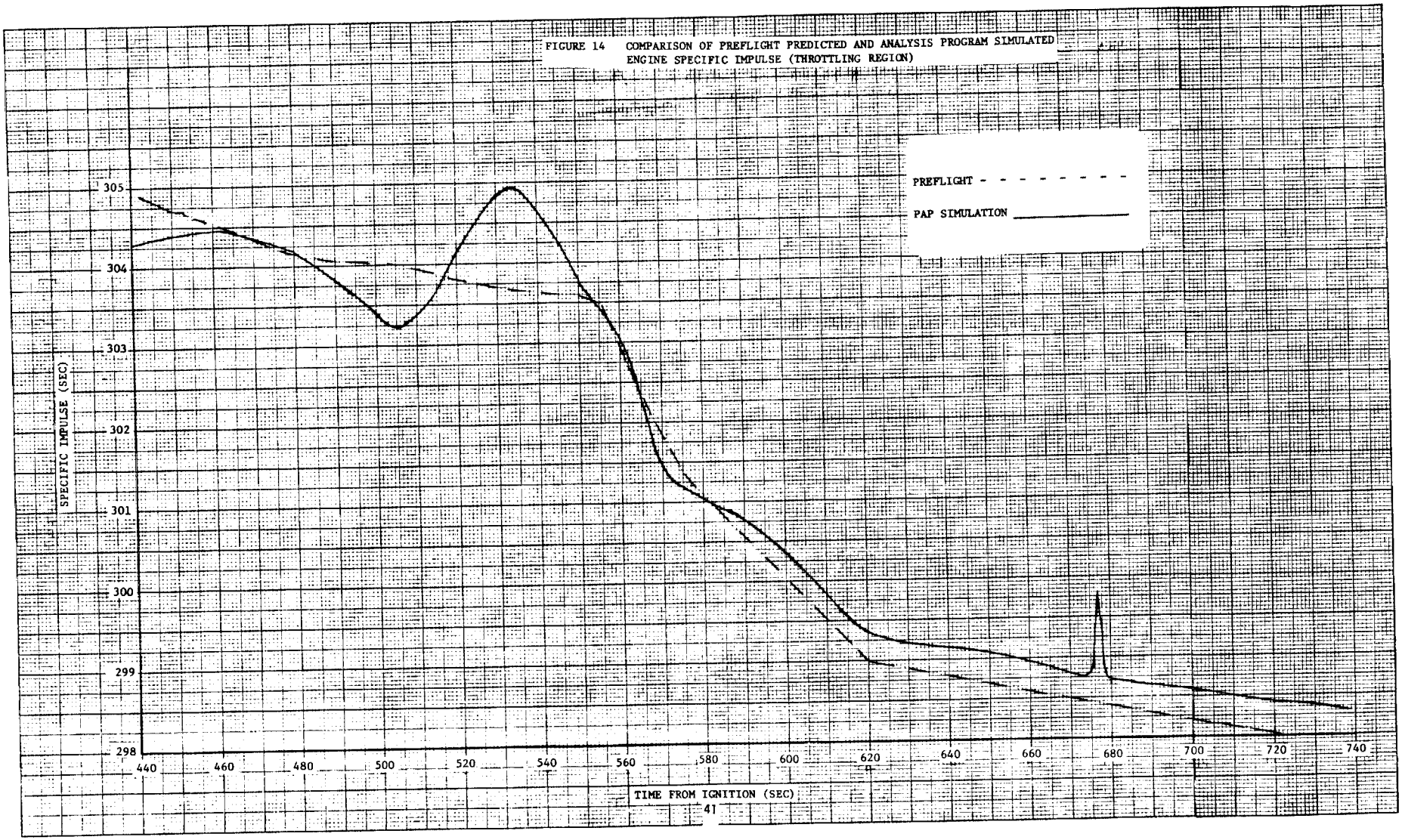
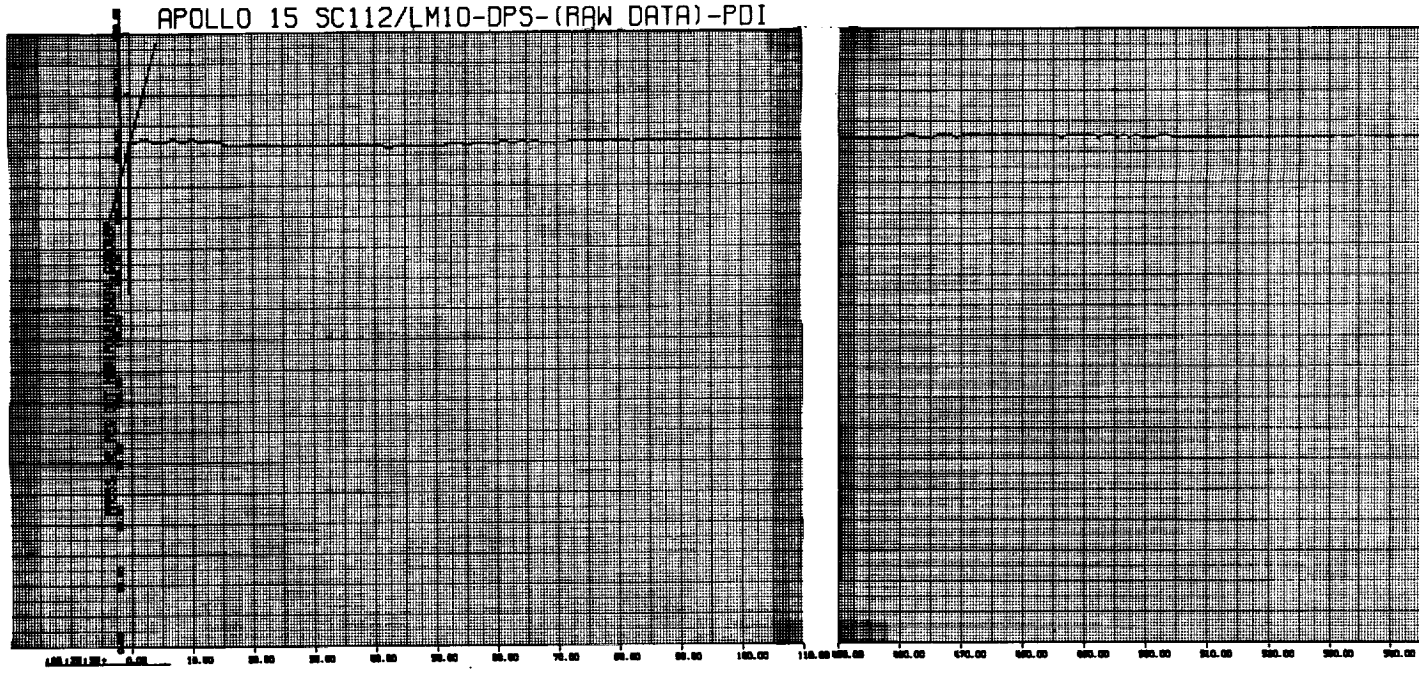
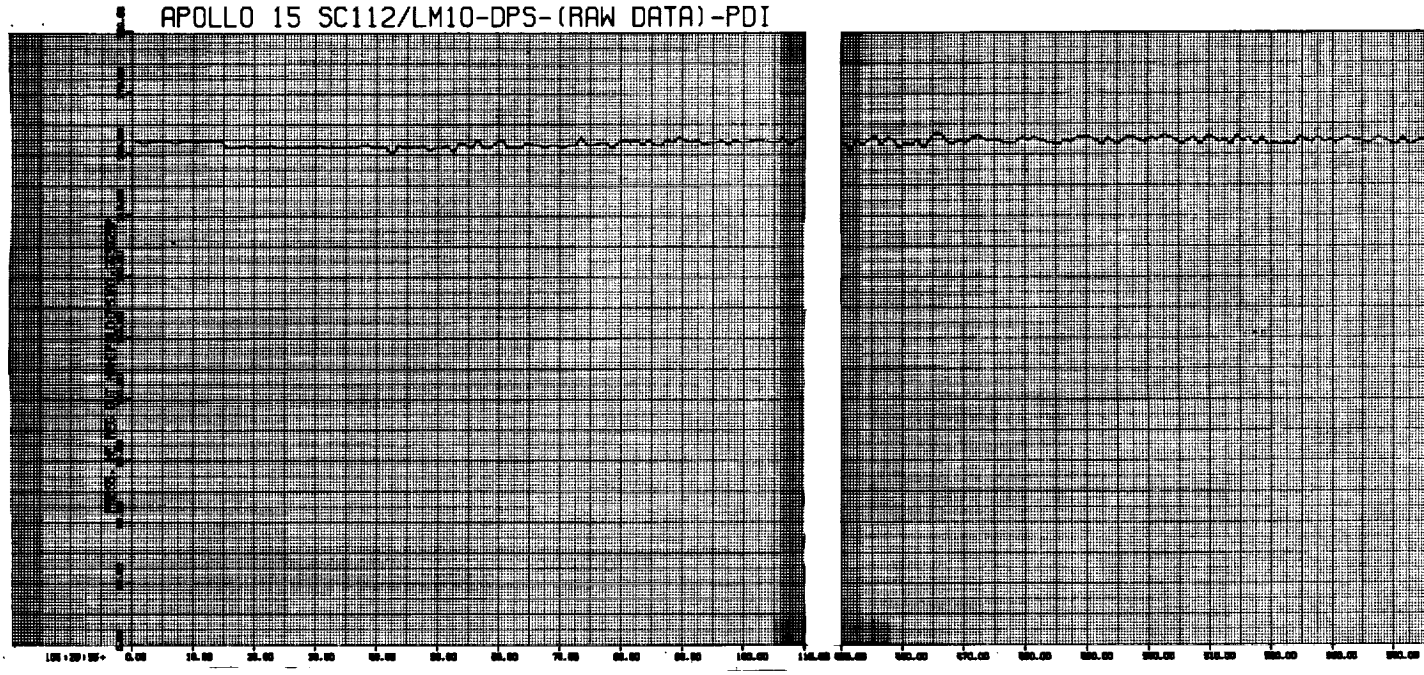


FIGURE 13: COMPARISON OF PREFLIGHT PREDICTED AND ANALYSIS PROGRAM SIMULATED ENGINE MIXTURE RATIO (THROTTLING REGION)
PREFLIGHT
PAP ANALYSIS
1.61
1.60
1.59
1.58
1.57
1.56
MIXTURE RATIO (O/F)
420
440
460
480
500
520
540
560
580
600
620
640
660
680
700
720
740
TIME FROM IGNITION (SEC)

FIGURE 14 COMPARISON OF PREFLIGHT PREDICTED AND ANALYSIS PROGRAM SIMULATED ENGINE SPECIFIC IMPULSE (THROTTLING REGION)




APOLLO 15 SC112/LM10-DPS-(RAW DATA)-PDI


APOLLO 15 SC112/LM10-DPS-(RAW DATA)-PDI

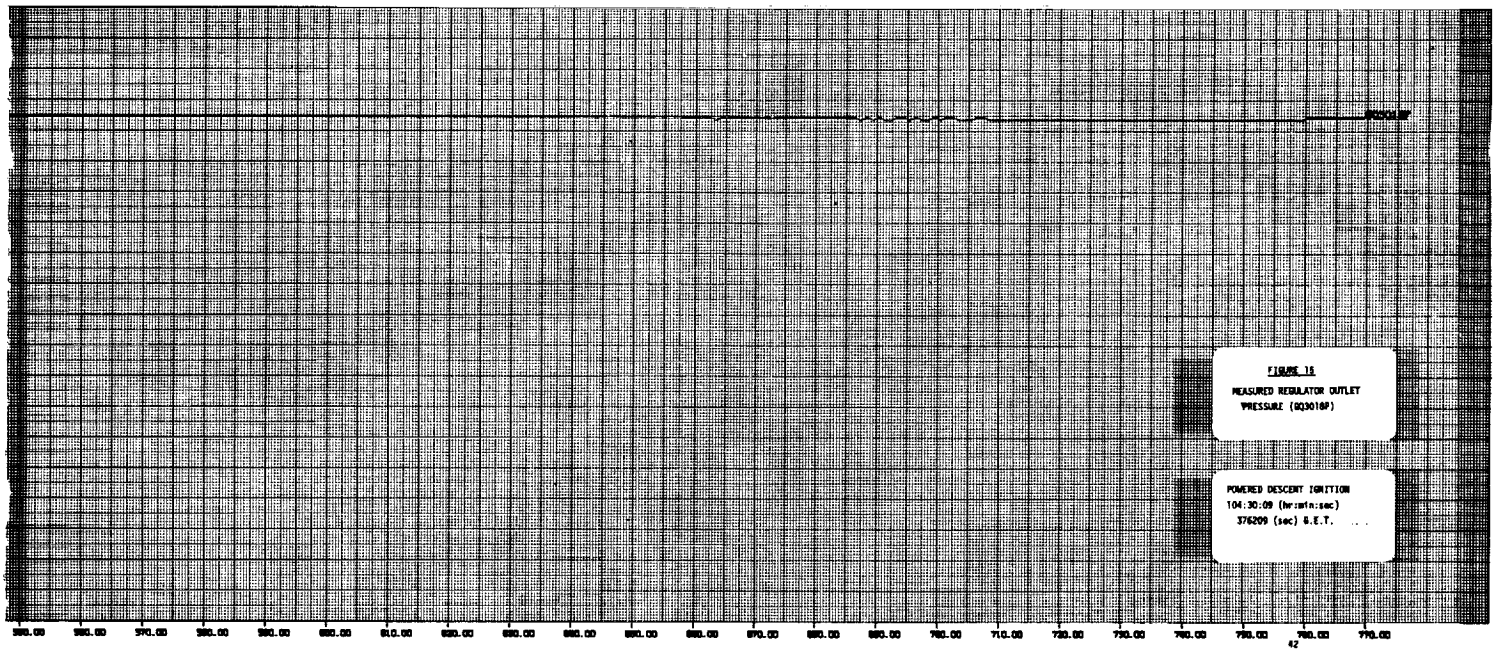


FIGURE 15
MEASURED REGULATOR OUTLET PRESSURE (GQ3010P)

POWERED DESCENT IGNITION
104:30:09 (hr:min:sec)
376209 (sec) G.E.T.

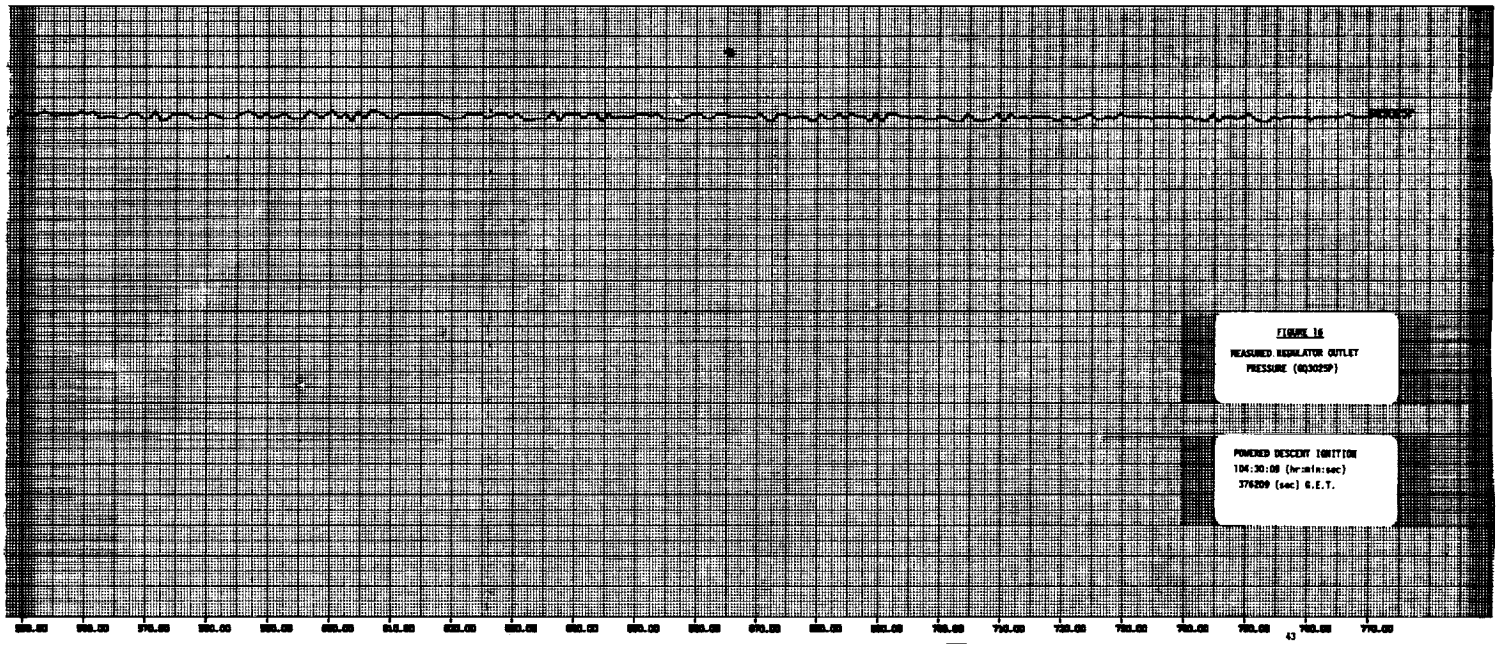
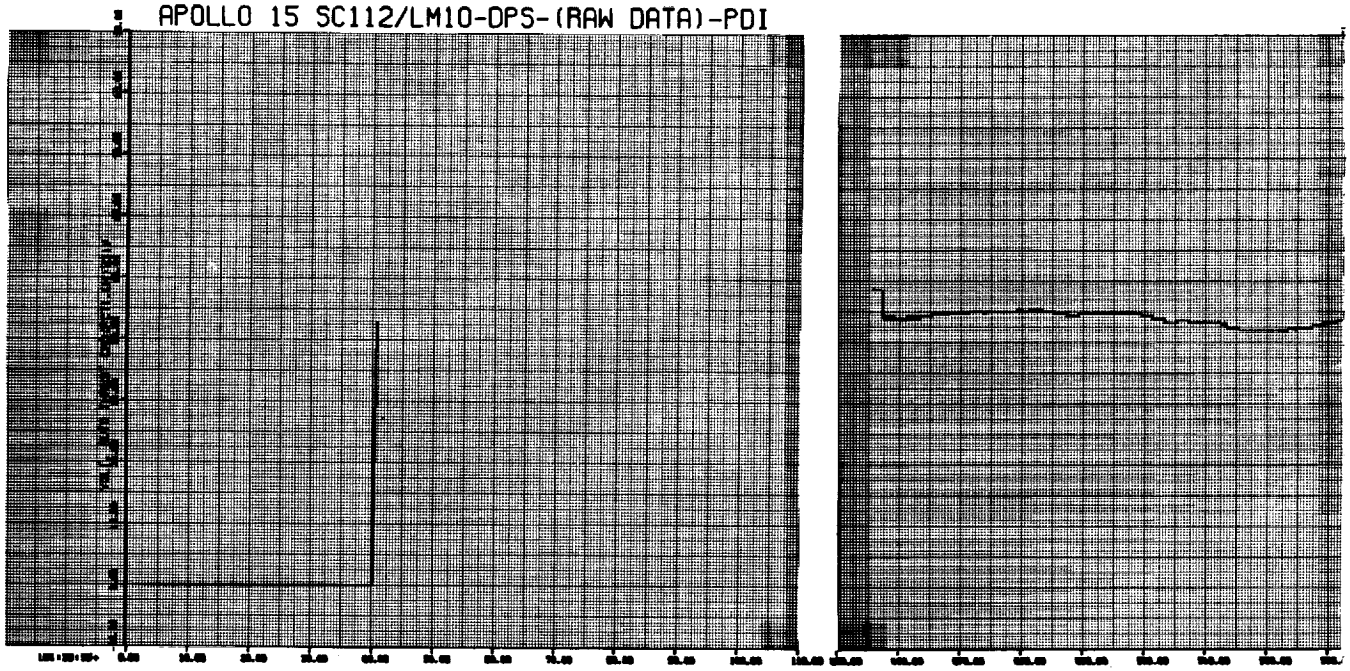


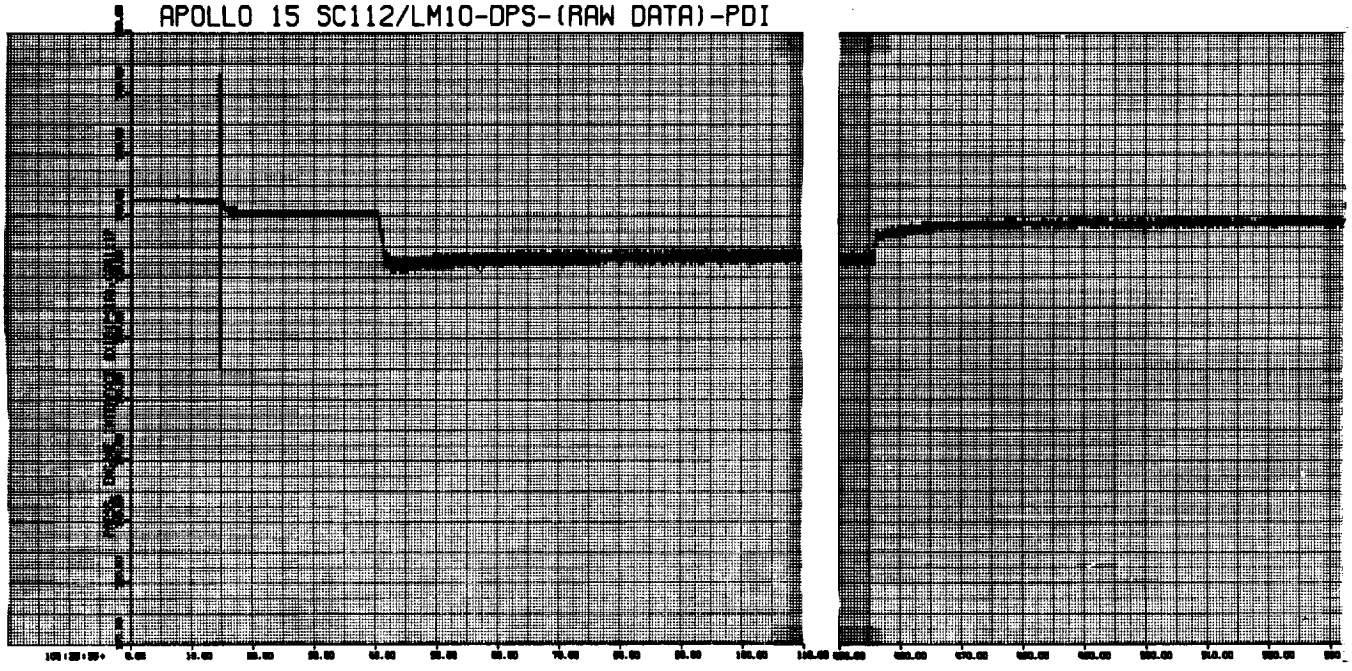
FIGURE 16
MEASURED REGULATOR OUTLET PRESSURE (GQ3025P)

POWERED DESCENT IGNITION
104:30:08 (hr:min:sec)
376209 (sec) G.E.T.

APOLLO 15 SC112/LM10-DPS-(RAW DATA)-PDI



APOLLO 15 SC112/LM10-DPS-(RAW DATA)-PDI



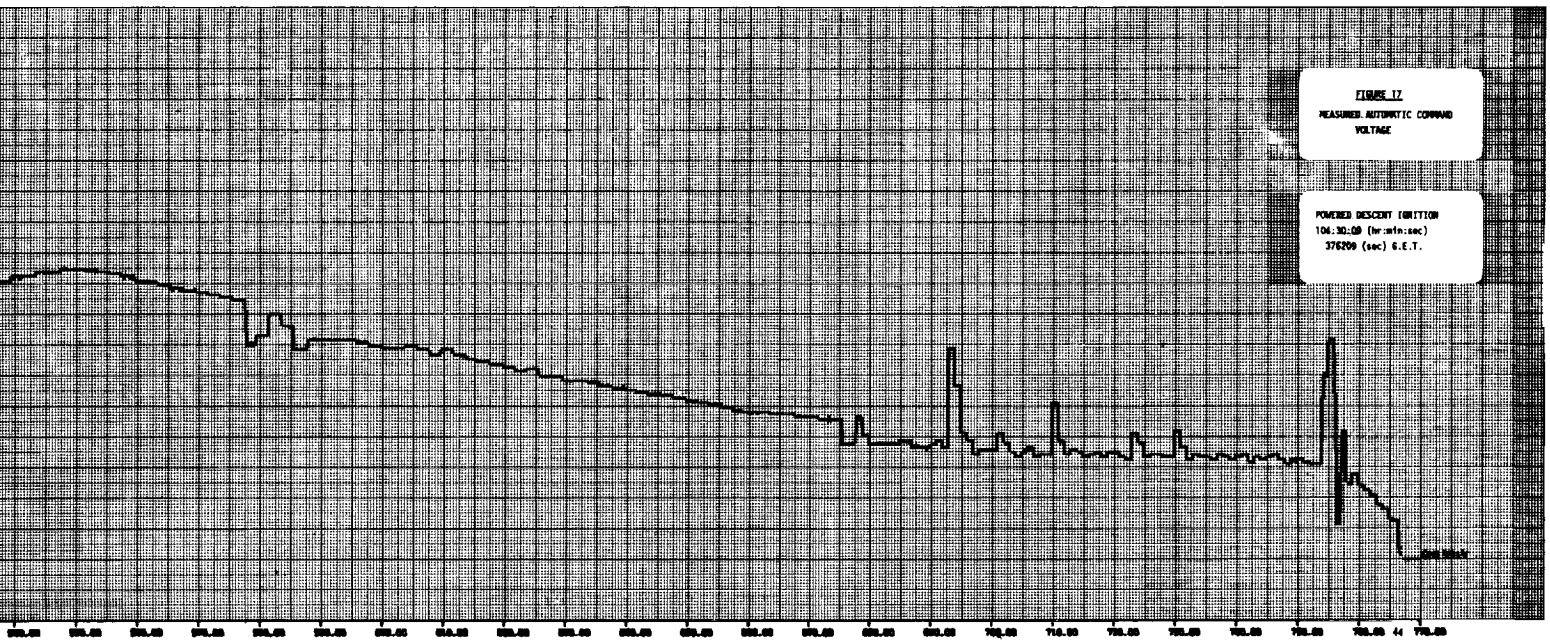


FIGURE 17
MEASURED AUTOMATIC COMMAND VOLTAGE

POWERED DESCENT IGNITION
104:30:09 (hr:min:sec)
376209 (sec) G.E.T.

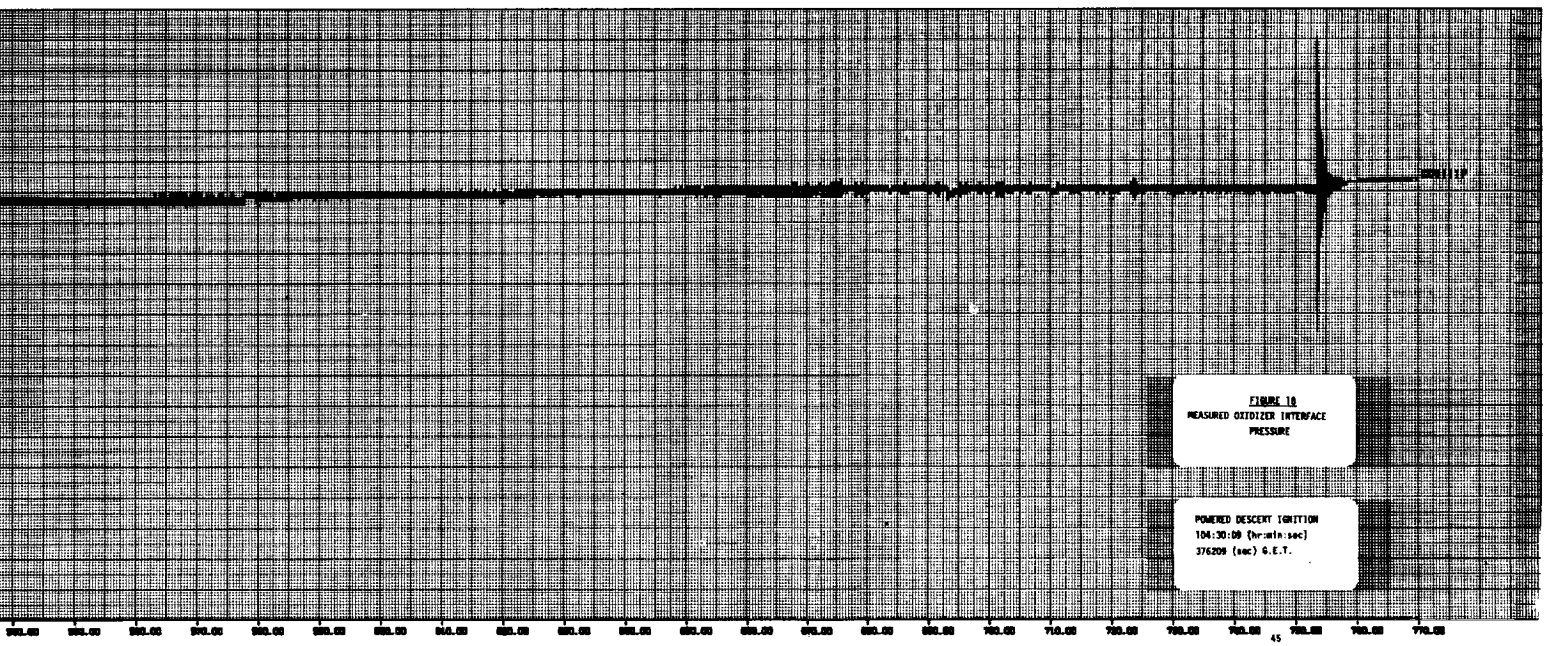
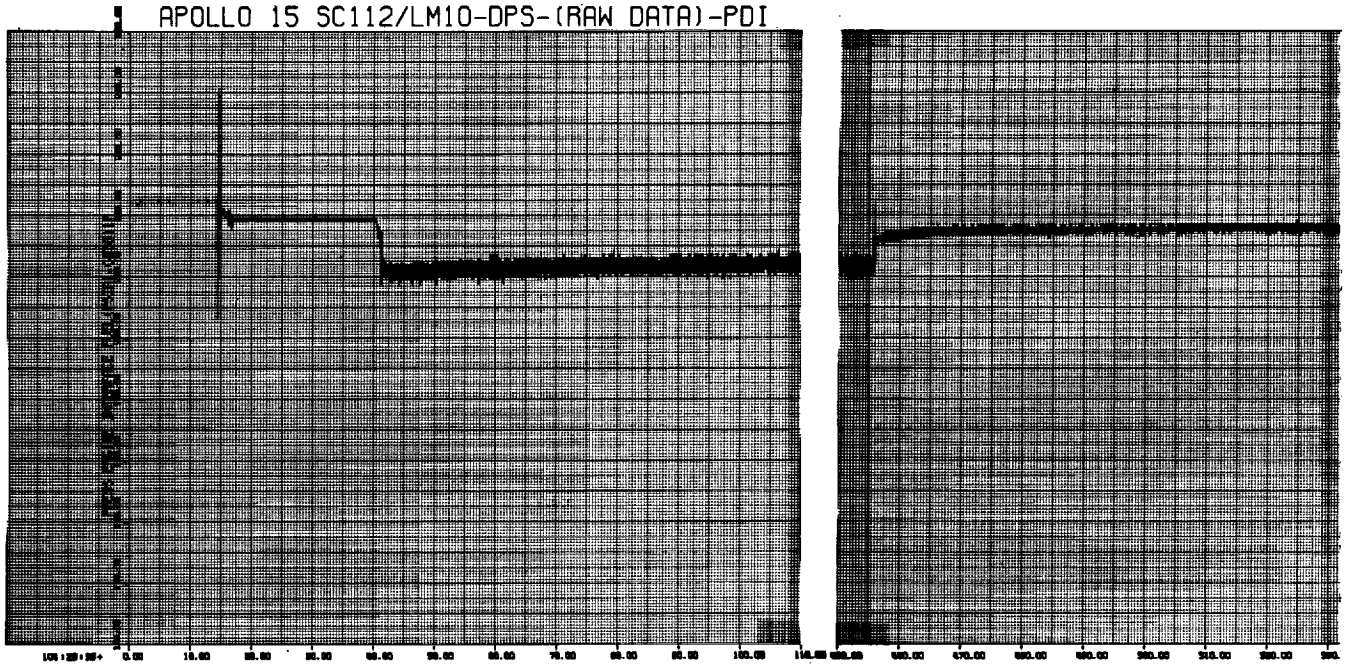


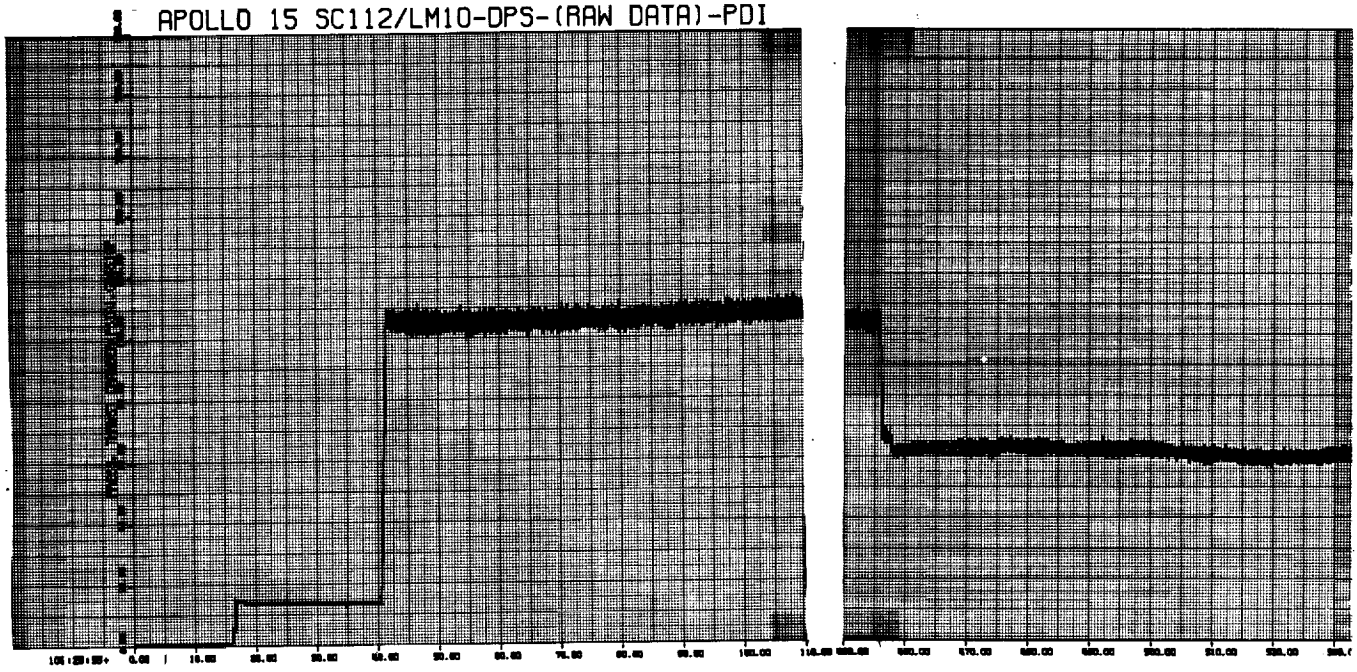
FIGURE 18
MEASURED OXIDIZER INTERFACE PRESSURE

POWERED DESCENT IGNITION
104:30:09 (hr:min:sec)
376209 (sec) G.E.T.

APOLLO 15 SC112/LM10-DPS-(RAW DATA)-PDI



APOLLO 15 SC112/LM10-DPS-(RAW DATA)-PDI



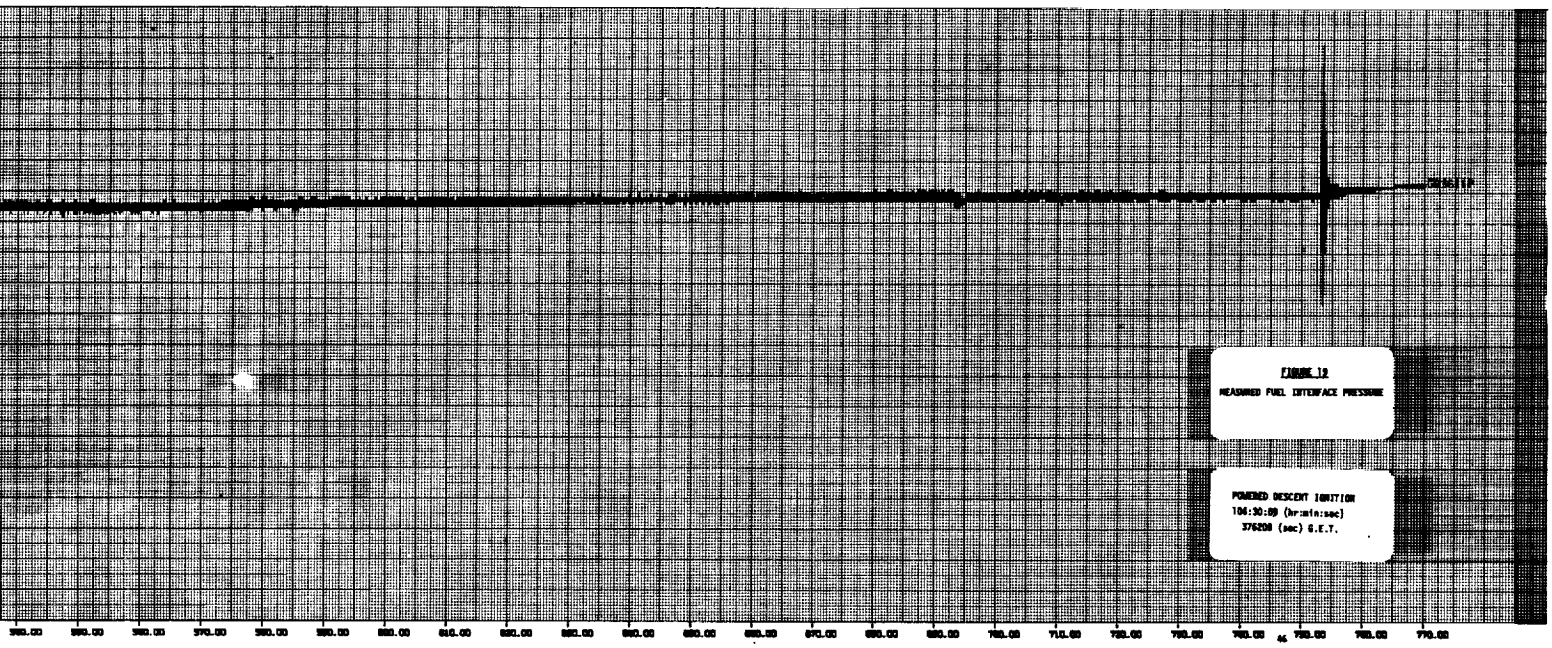


FIGURE 19
MEASURED FUEL INTERFACE PRESSURE

POWERED DESCENT IGNITION
104:30:09 (hr:min:sec)
376209 (sec) G.E.T.

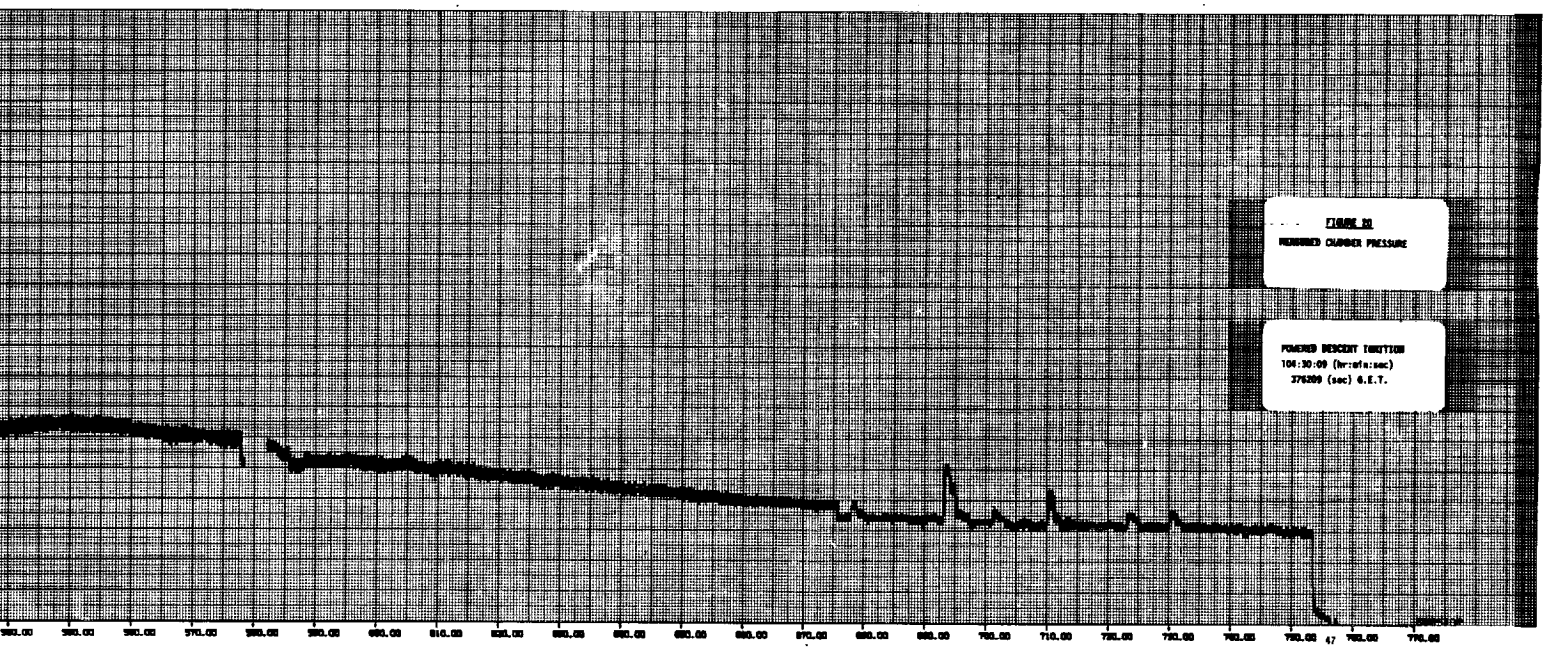
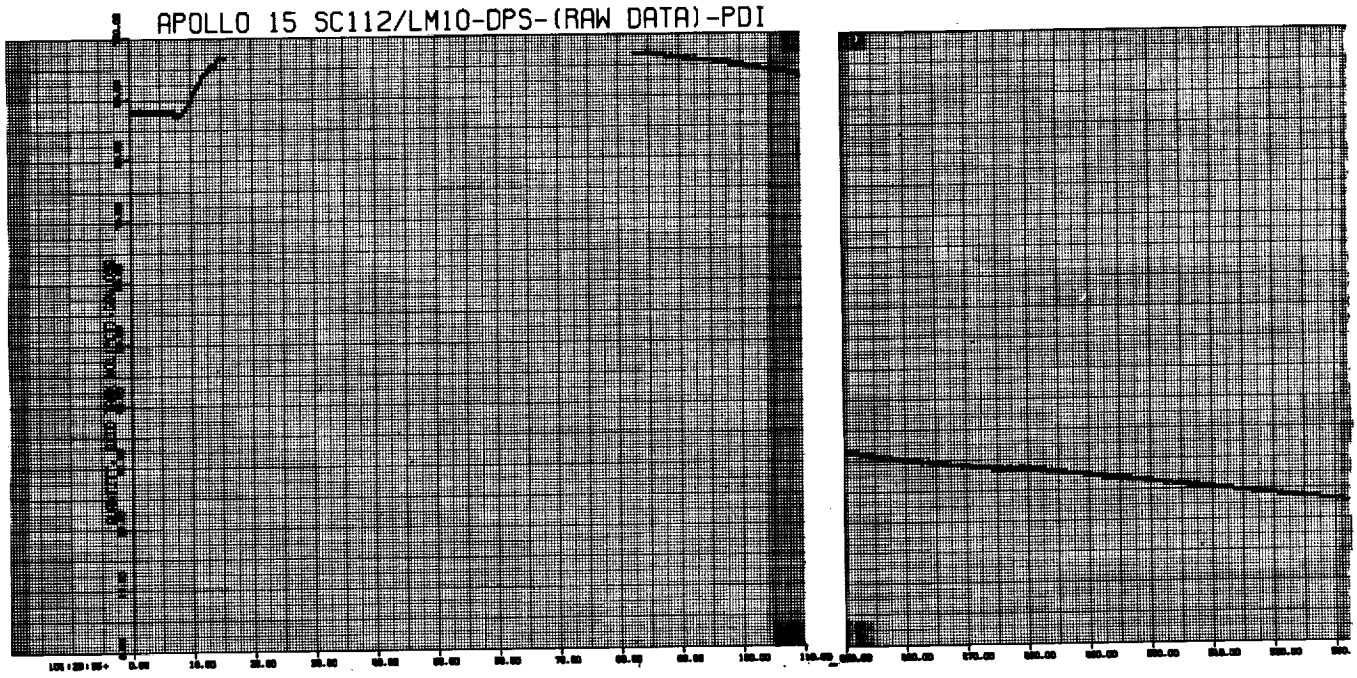
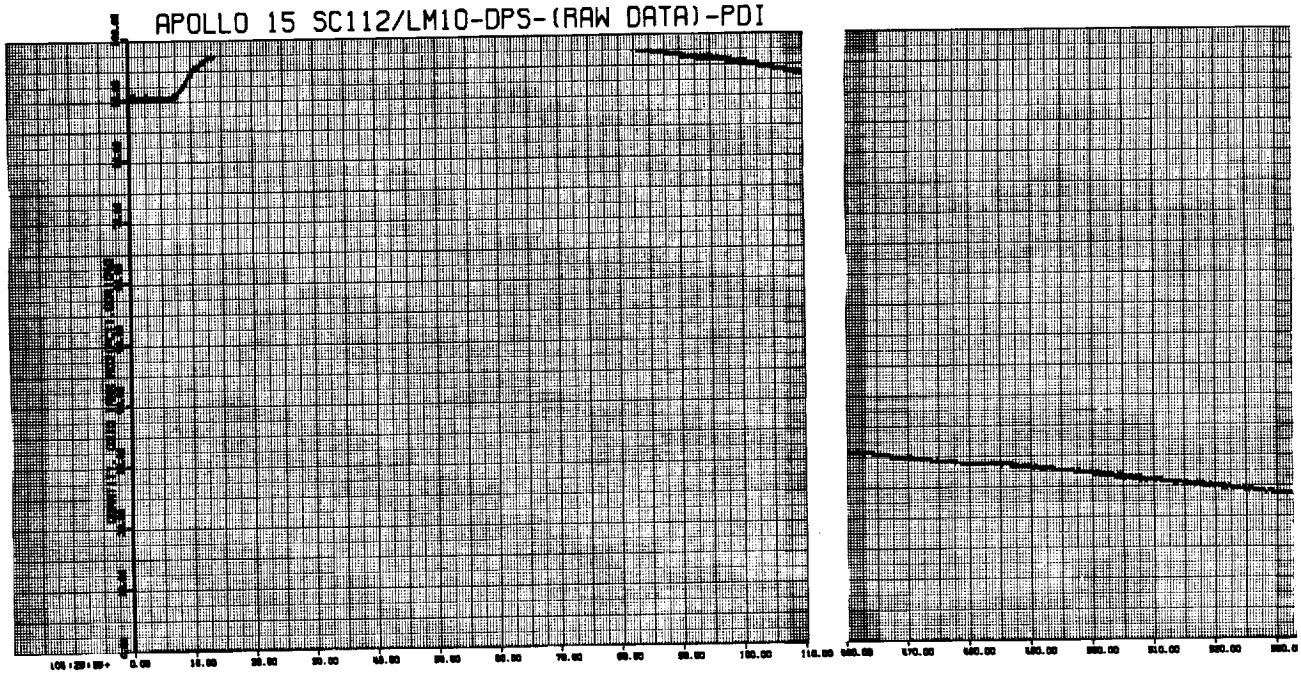


FIGURE 20
MEASURED CHAMBER PRESSURE

POWERED DESCENT IGNITION
104:30:09 (hr:min:sec)
376209 (sec) G.E.T.


APOLLO 15 SC112/LM10-DPS-(RAW DATA)-PDI


APOLLO 15 SC112/LM10-DPS-(RAW DATA)-PDI

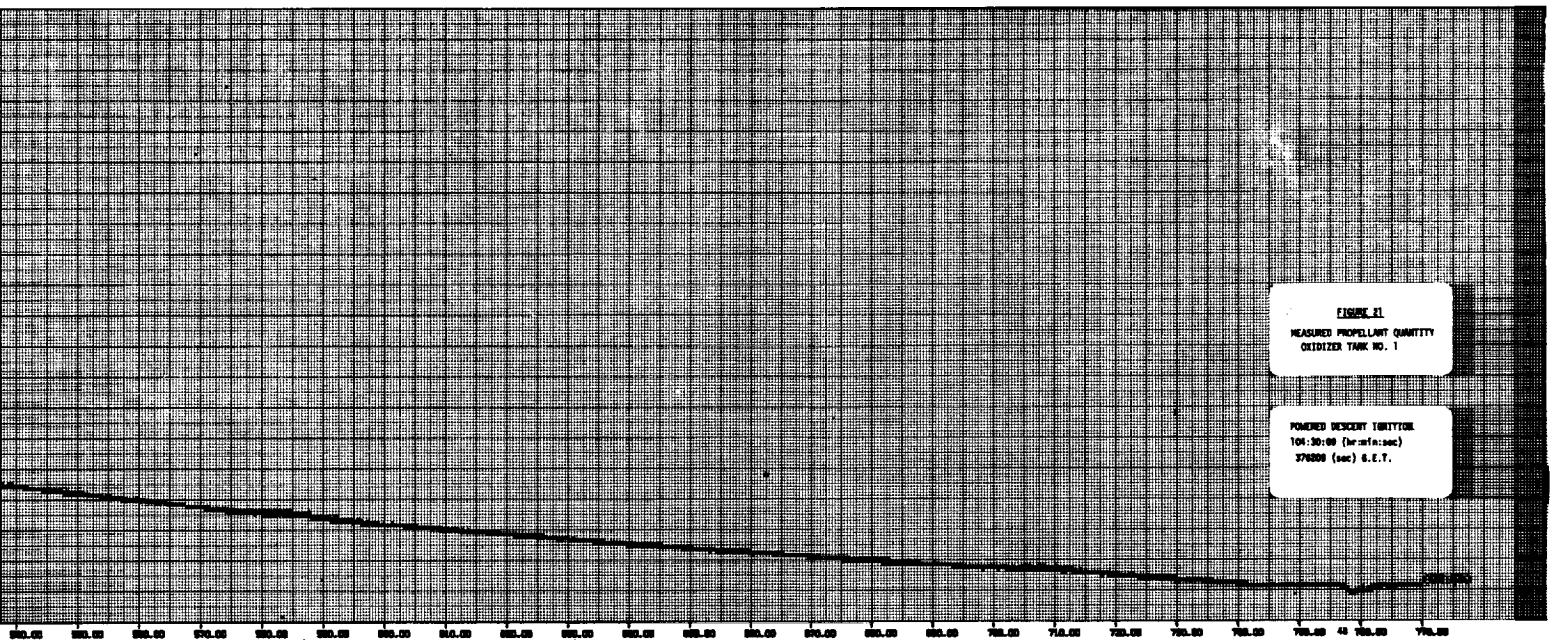


FIGURE 21
MEASURED PROPELLANT QUANTITY
OXIDIZER TANK NO. 1
POWERED DESCENT IGNITION
104:30:08 (hr:min:sec)
376808 (sec) G.E.T.

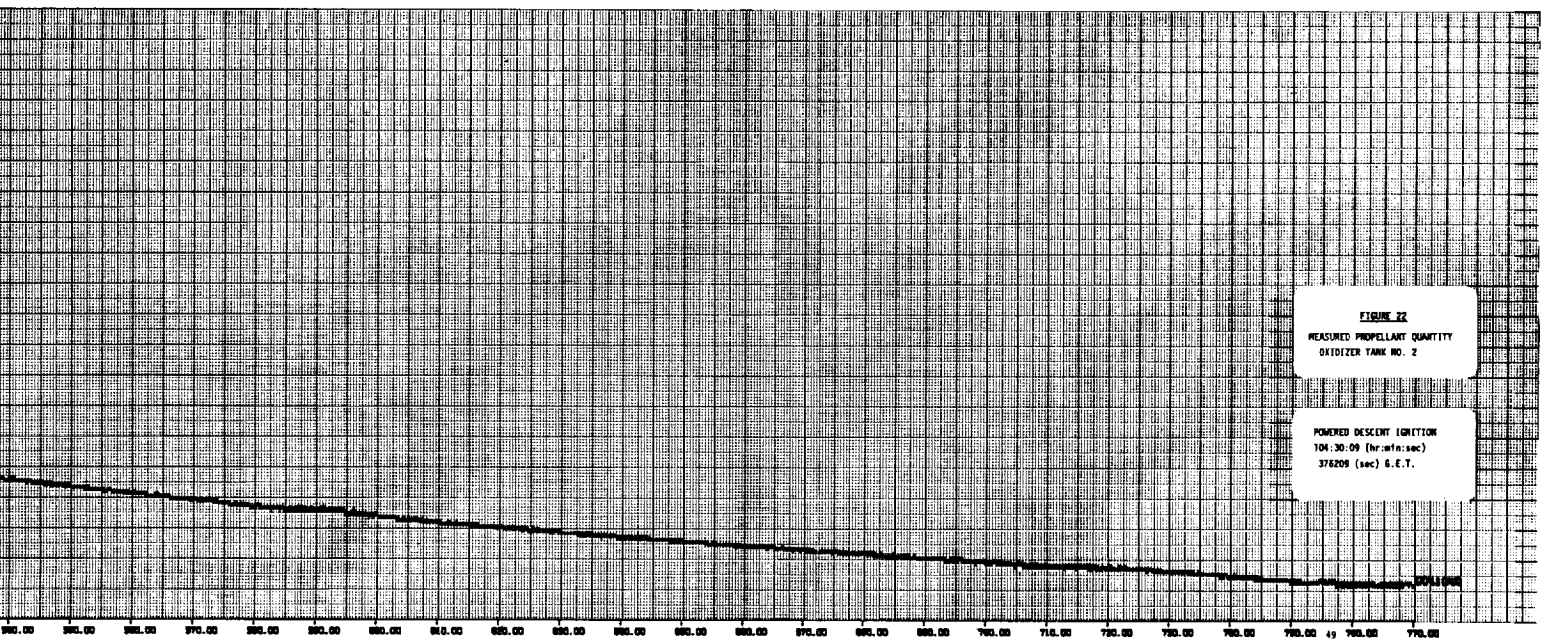
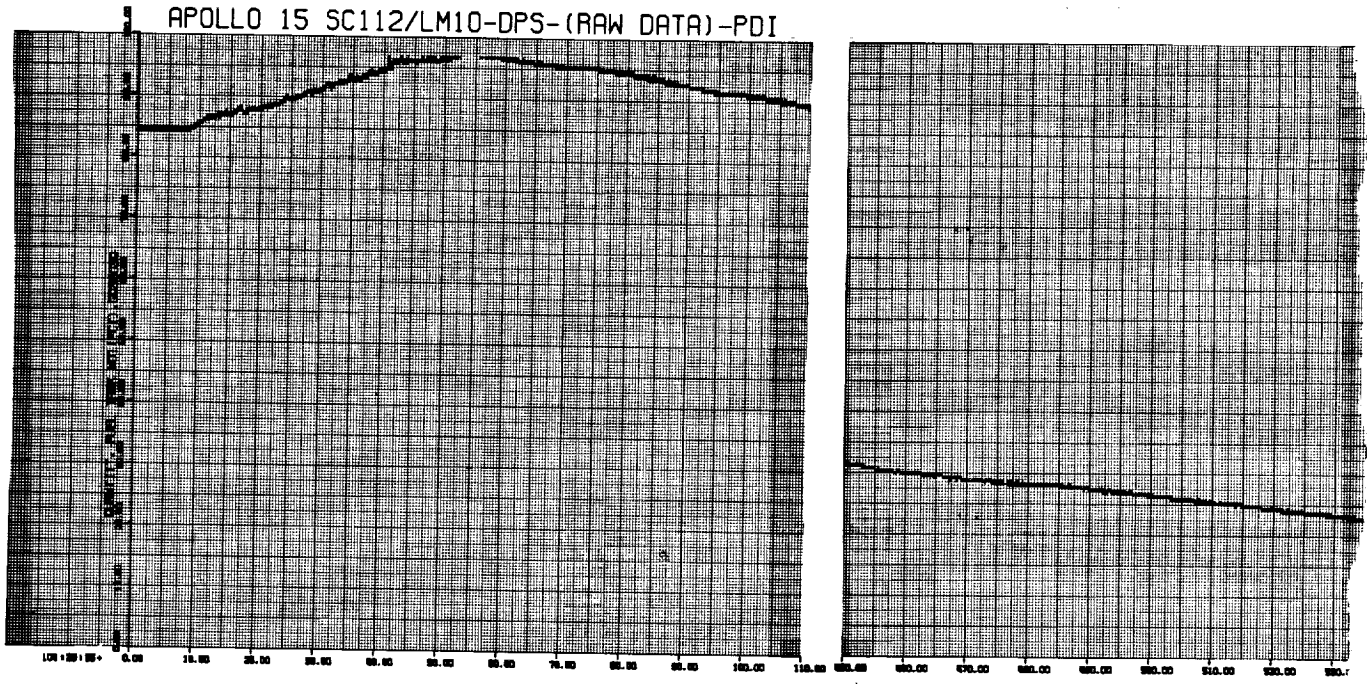
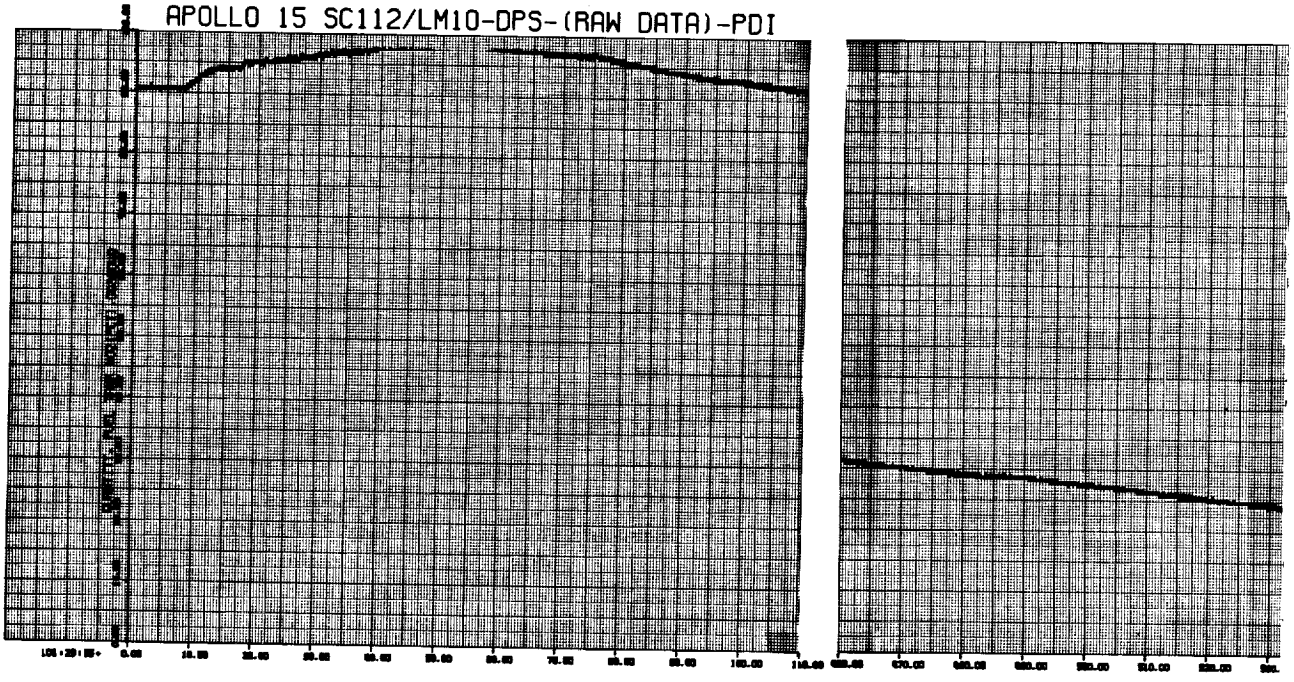


FIGURE 22
MEASURED PROPELLANT QUANTITY
OXIDIZER TANK NO. 2
POWERED DESCENT IGNITION
104:30:09 (hr:min:sec)
376209 (sec) G.E.T.


APOLLO 15 SC112/LM10-DPS-(RAW DATA)-PDI


APOLLO 15 SC112/LM10-DPS-(RAW DATA)-PDI

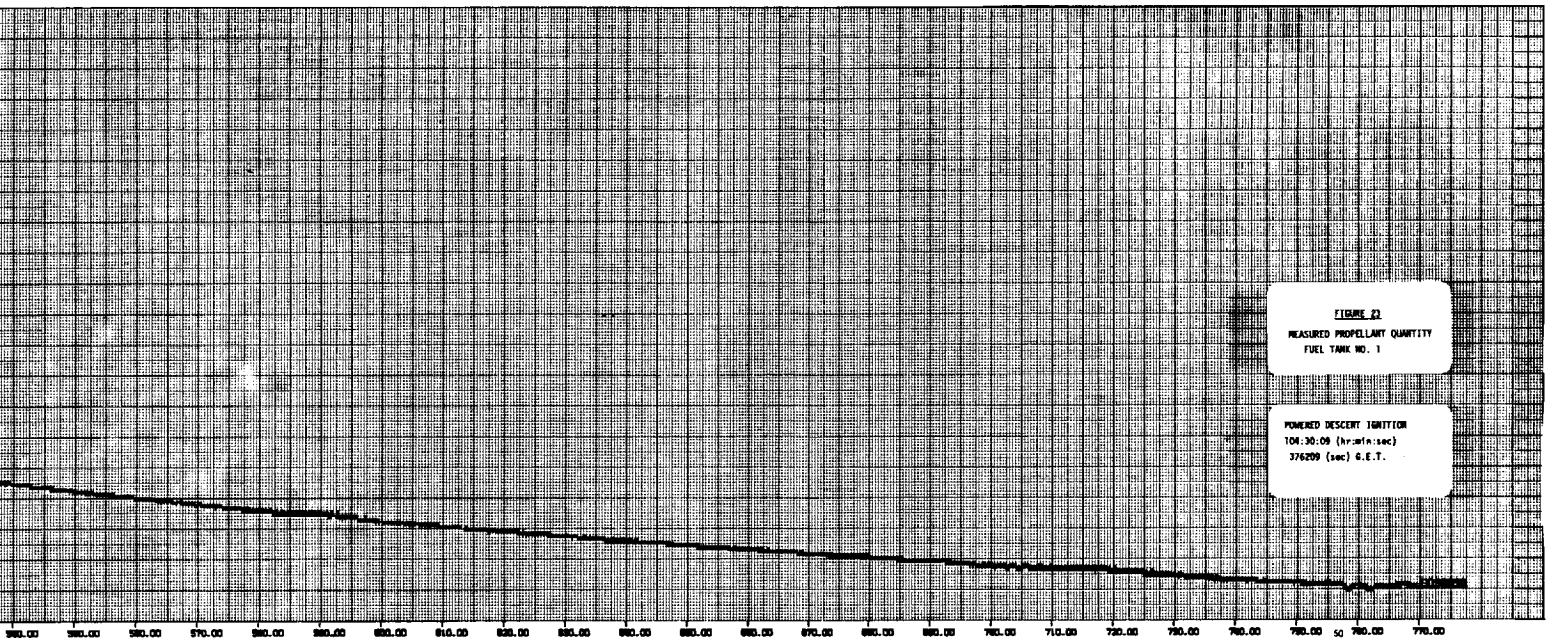


FIGURE 23
MEASURED PROPELLANT QUANTITY
FUEL TANK NO. 1
POWERED DESCENT IGNITION
104:30:09 (hr:min:sec)
376209 (sec) G.E.T.

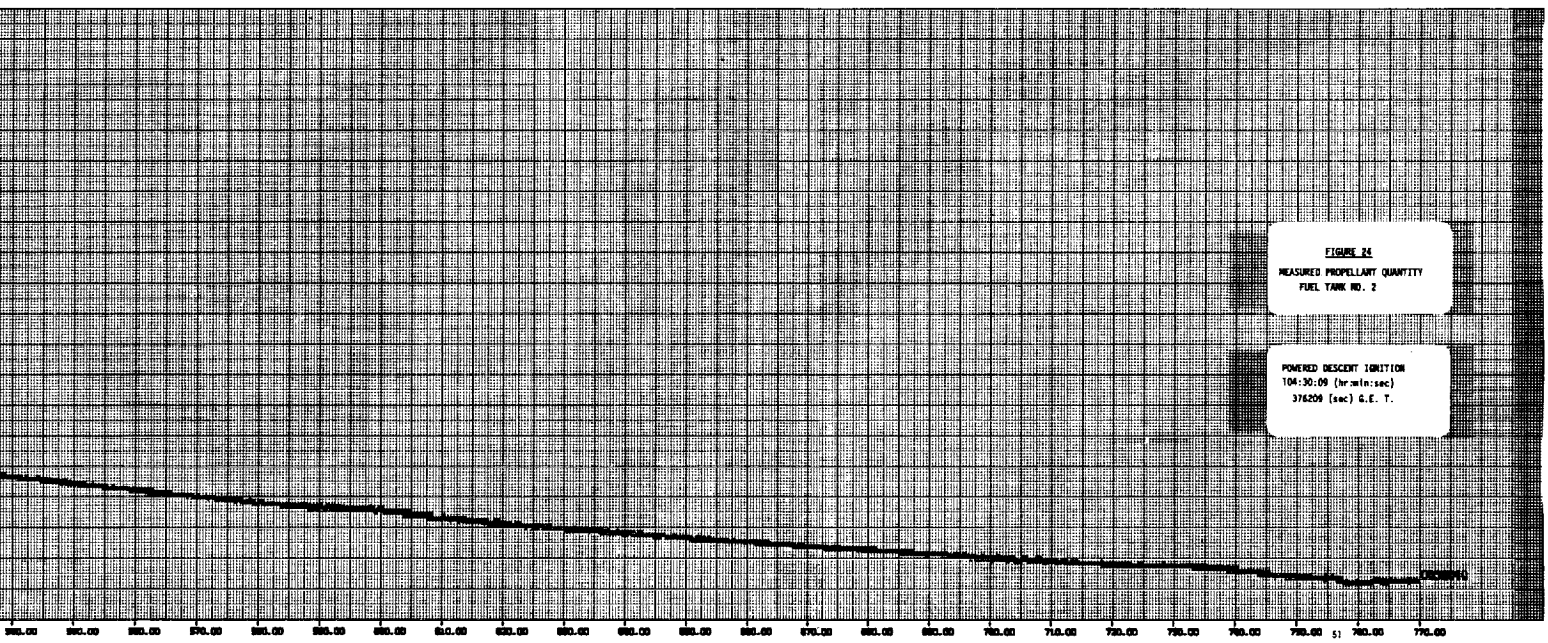
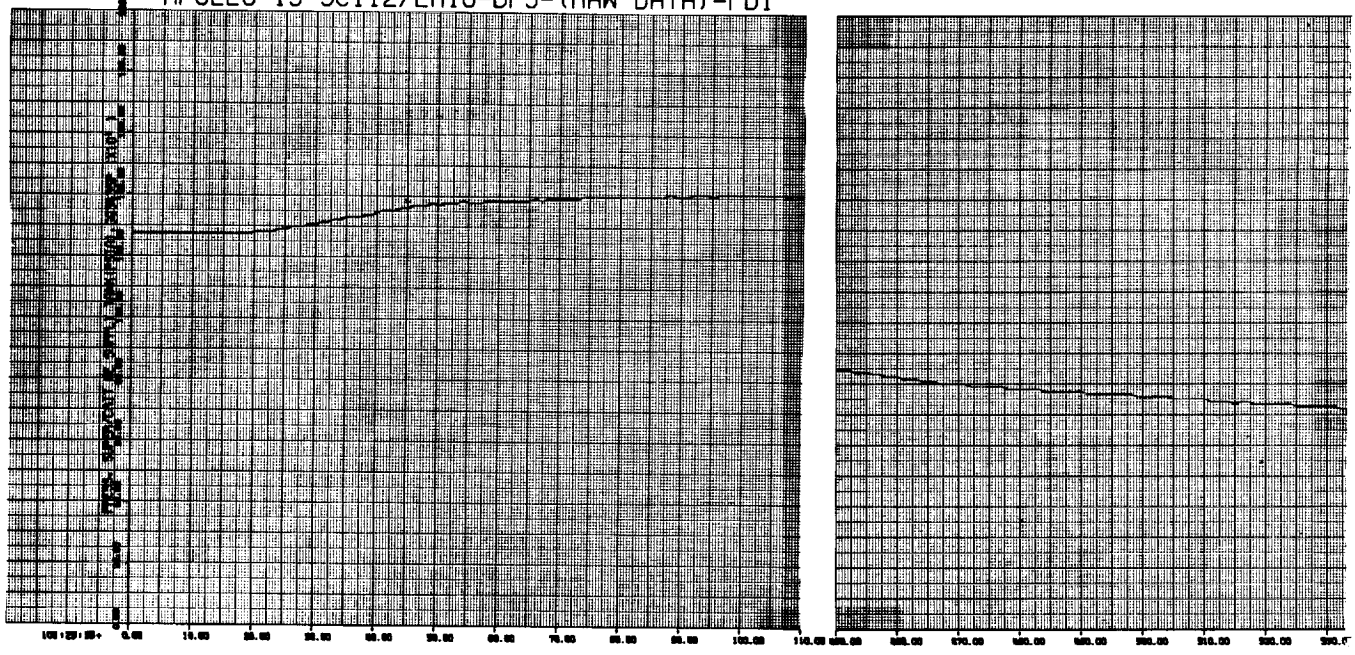


FIGURE 24
MEASURED PROPELLANT QUANTITY
FUEL TANK NO. 2
POWERED DESCENT IGNITION
104:30:09 (hr:min:sec)
376209 (sec) G.E. T.

APOLLO 15 SC112/LM10-DPS-(RAW DATA)-PDI



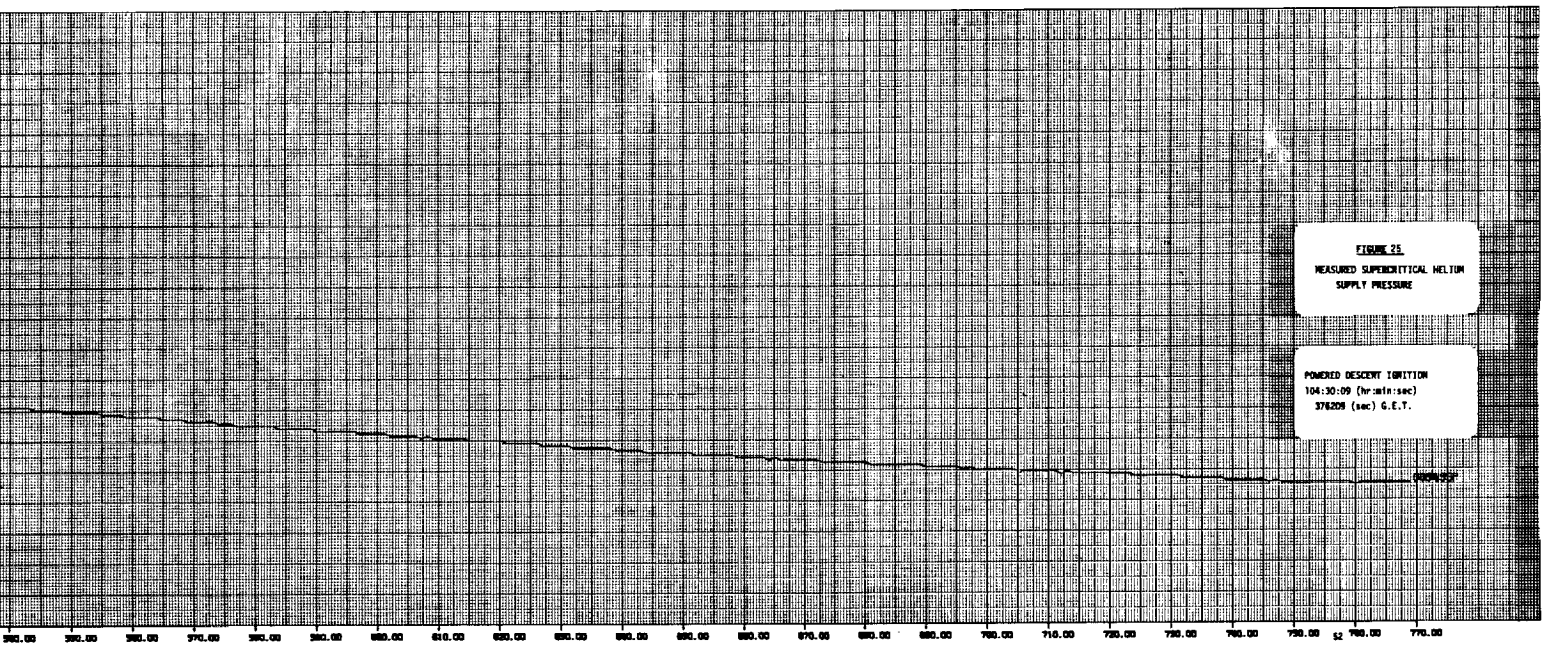


FIGURE 25
MEASURED SUPERCRITICAL HELIUM
SUPPLY PRESSURE
POWERED DESCENT IGNITION
104:30:09 (hr:min:sec)
376209 (sec) G.E.T.
52

FIGURE 26

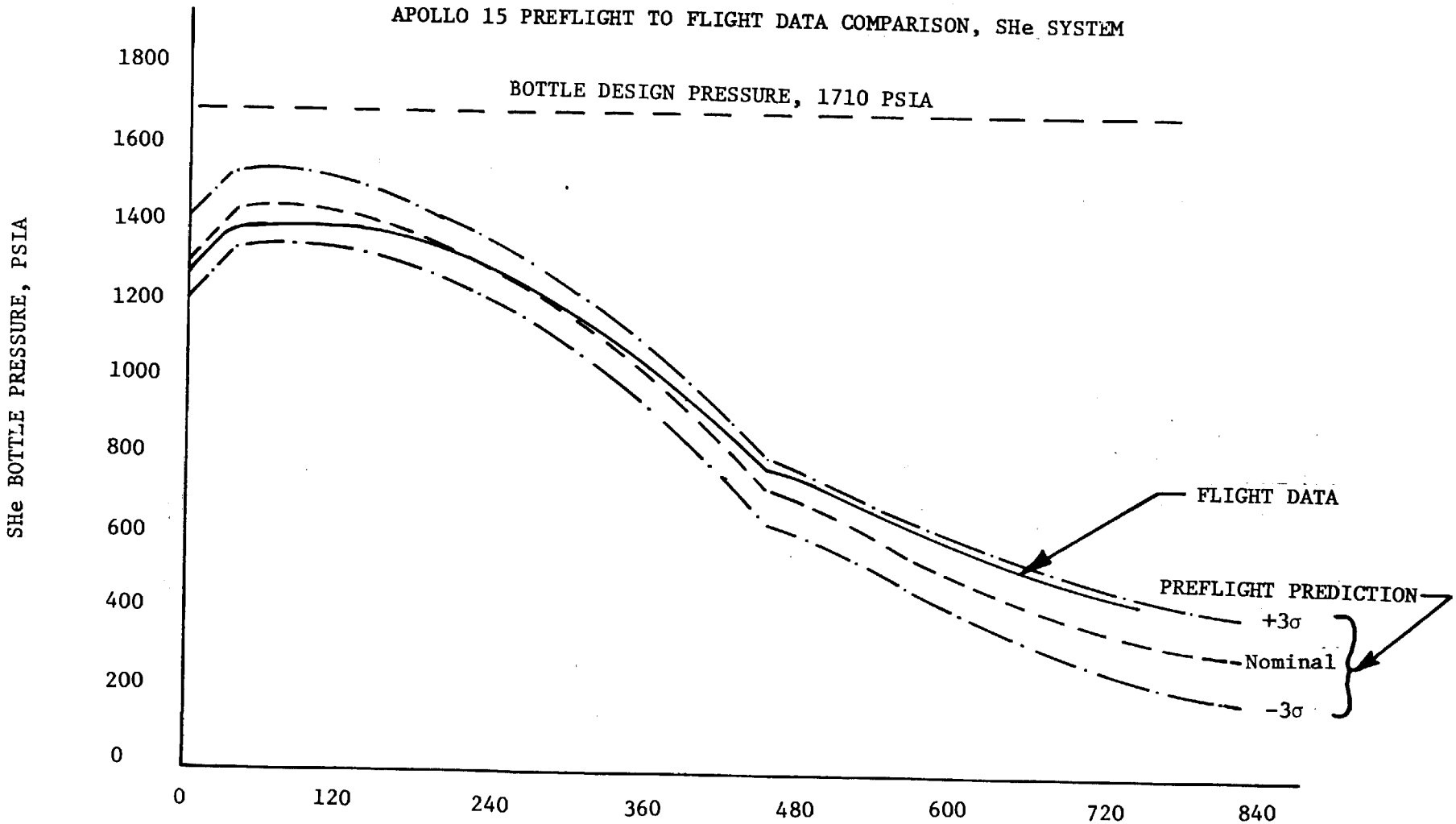
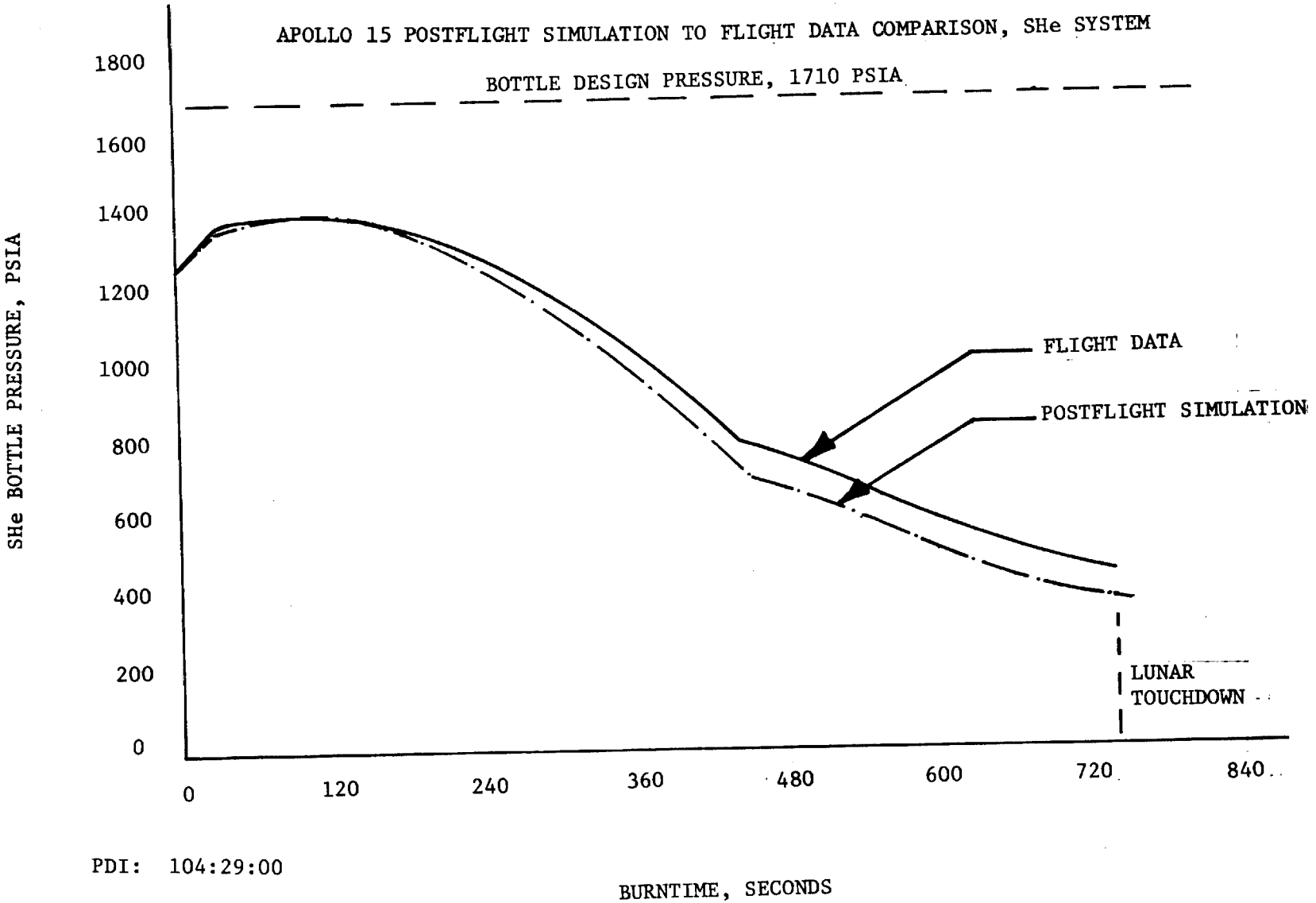
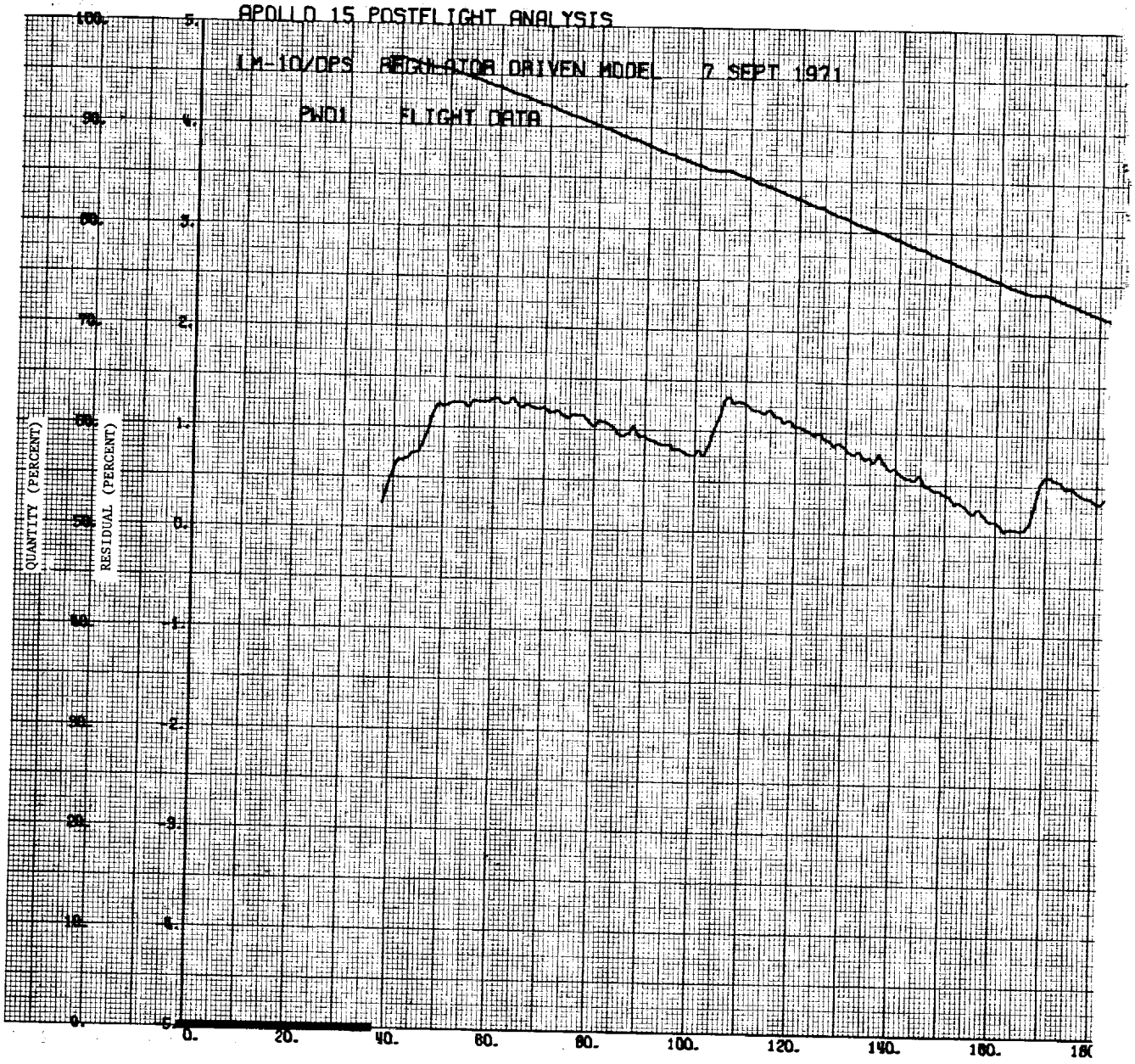

APOLLO 15 PREFLIGHT TO FLIGHT DATA COMPARISON, SHe SYSTEM
BOTTLE DESIGN PRESSURE, 1710 PSIA
SHe BOTTLE PRESSURE, PSIA
1800
1600
1400
1200
1000
800
600
400
200
0
0
120
240
360
480
600
720
840
FLIGHT DATA
PREFLIGHT PREDICTION
+3σ
Nominal
-3σ
PDI: 104:29:00
BURNTIME, SECONDS

FIGURE 27


APOLLO 15 POSTFLIGHT SIMULATION TO FLIGHT DATA COMPARISON, SHe SYSTEM
BOTTLE DESIGN PRESSURE, 1710 PSIA
SHe BOTTLE PRESSURE, PSIA
1800
1600
1400
1200
1000
800
600
400
200
0
FLIGHT DATA
POSTFLIGHT SIMULATION
LUNAR TOUCHDOWN
0
120
240
360
480
600
720
840
PDI: 104:29:00
BURNTIME, SECONDS


APOLLO 15 POSTFLIGHT ANALYSIS
LM-10/DPS REGULATOR DRIVEN MODEL 7 SEPT 1971
PWO1 FLIGHT DATA
QUANTITY (PERCENT)
RESIDUAL (PERCENT)
100.
90.
80.
70.
60.
50.
40.
30.
20.
10.
0.
5.
4.
3.
2.
1.
0.
-1.
-2.
-3.
-4.
-5.
0.
20.
40.
60.
80.
100.
120.
140.
160.

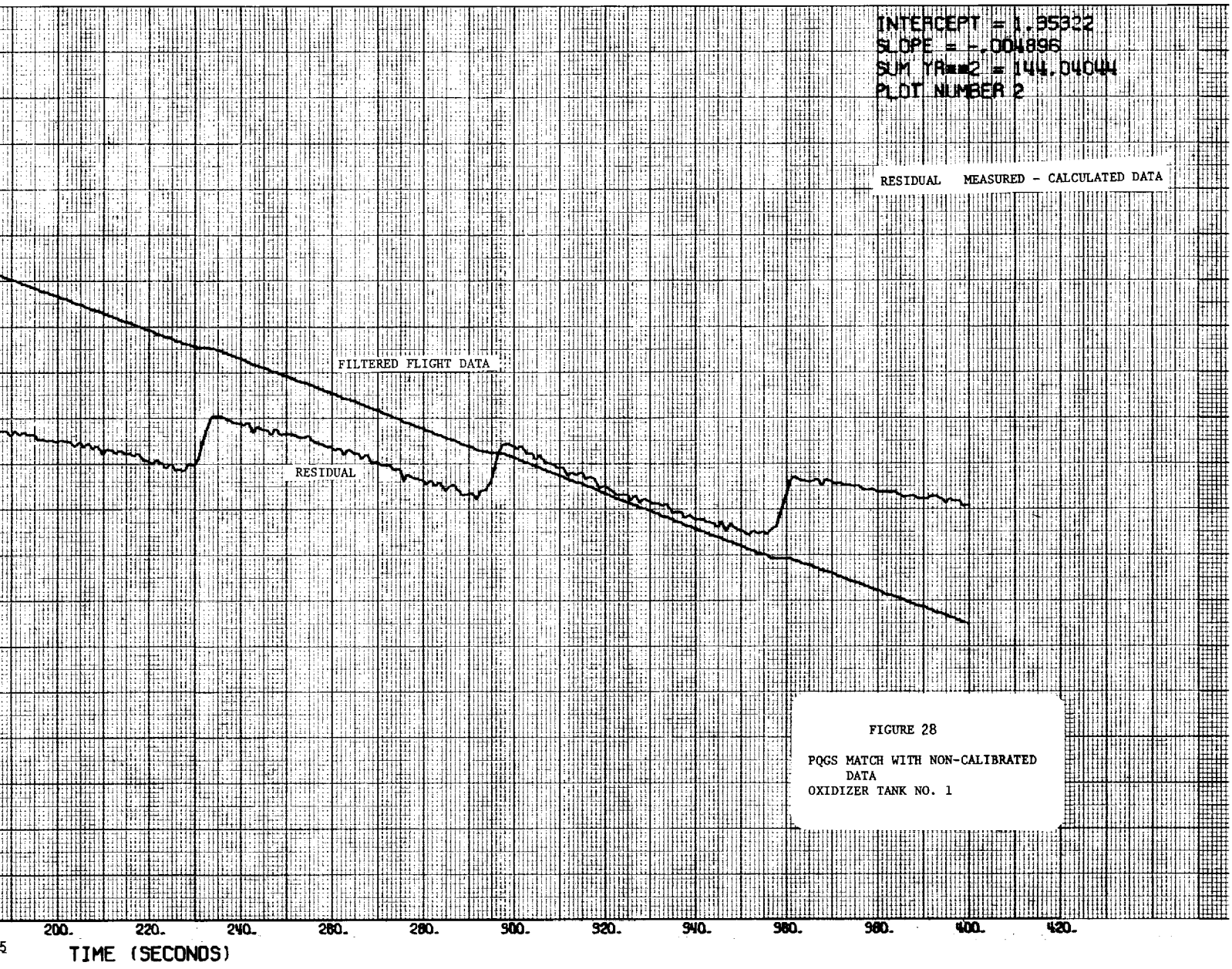
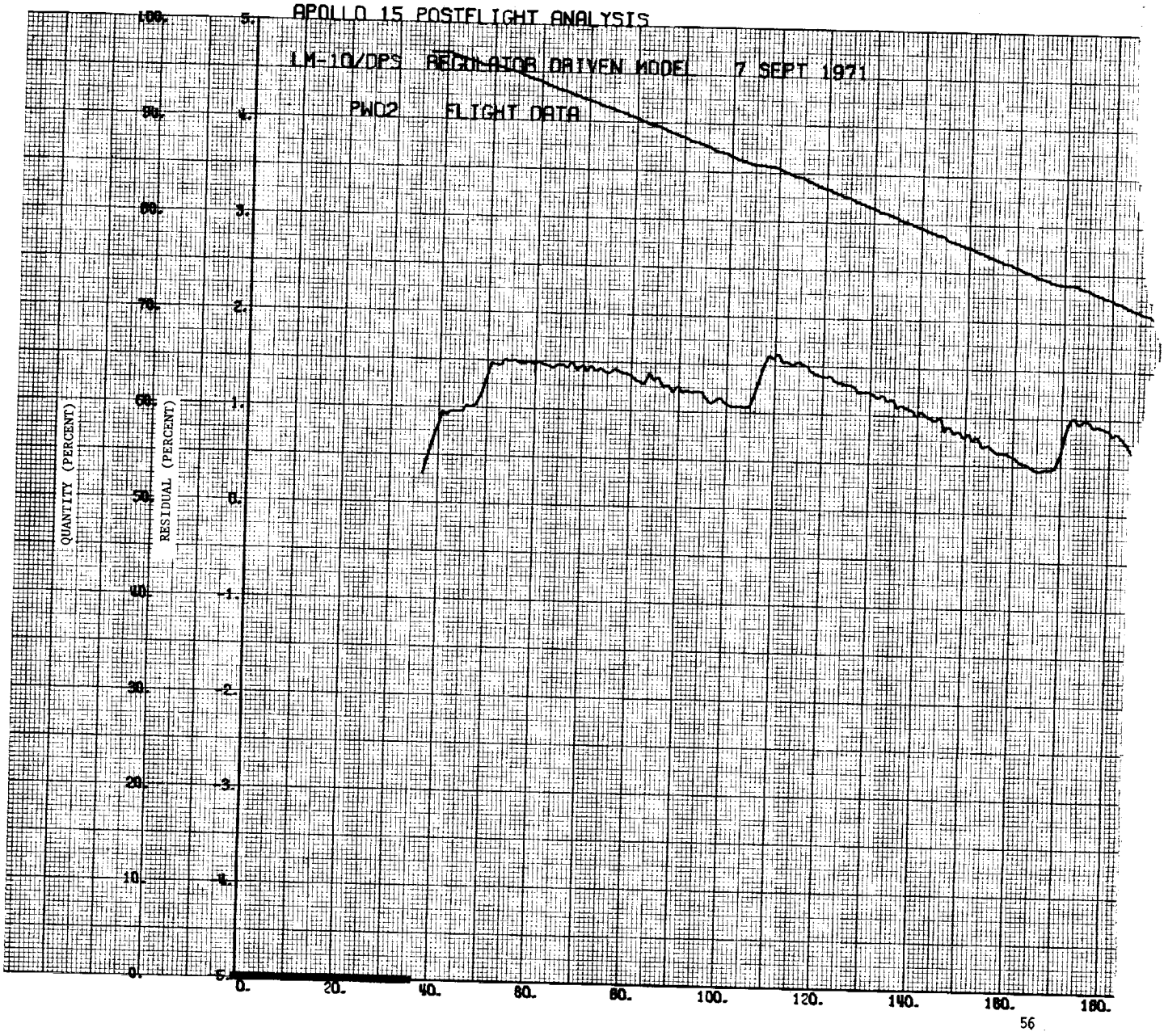


FIGURE 28

PQGS MATCH WITH NON-CALIBRATED DATA
OXIDIZER TANK NO. 1


APOLLO 15 POSTFLIGHT ANALYSIS
LM-10/DPS REGULATOR DRIVEN MODEL 7 SEPT 1971
PWO2 FLIGHT DATA
QUANTITY (PERCENT)
RESIDUAL (PERCENT)
100.
90.
80.
70.
60.
50.
40.
30.
20.
10.
0.
5.
4.
3.
2.
1.
0.
-1.
-2.
-3.
-4.
-5.
0.
20.
40.
60.
80.
100.
120.
140.
160.
180.

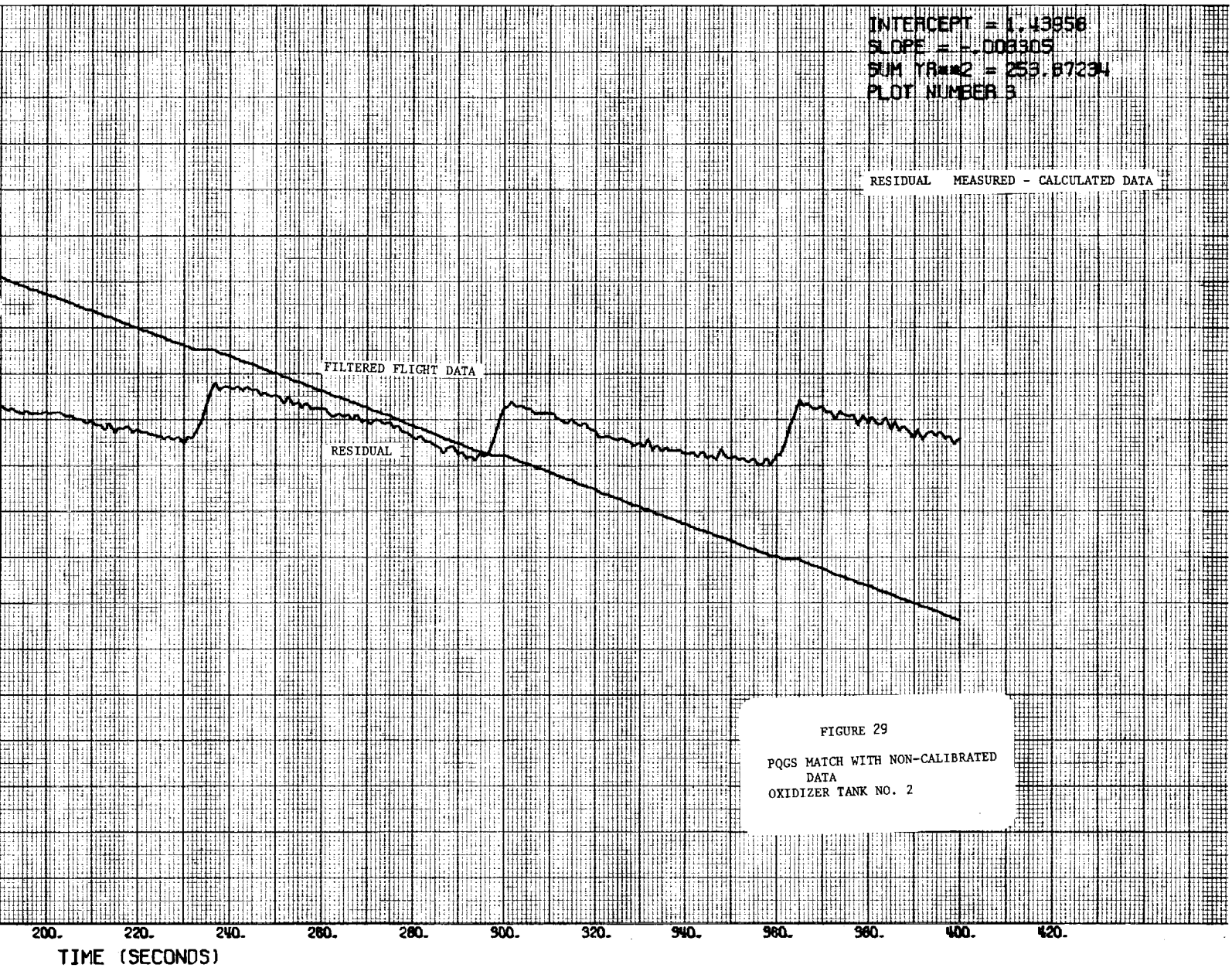
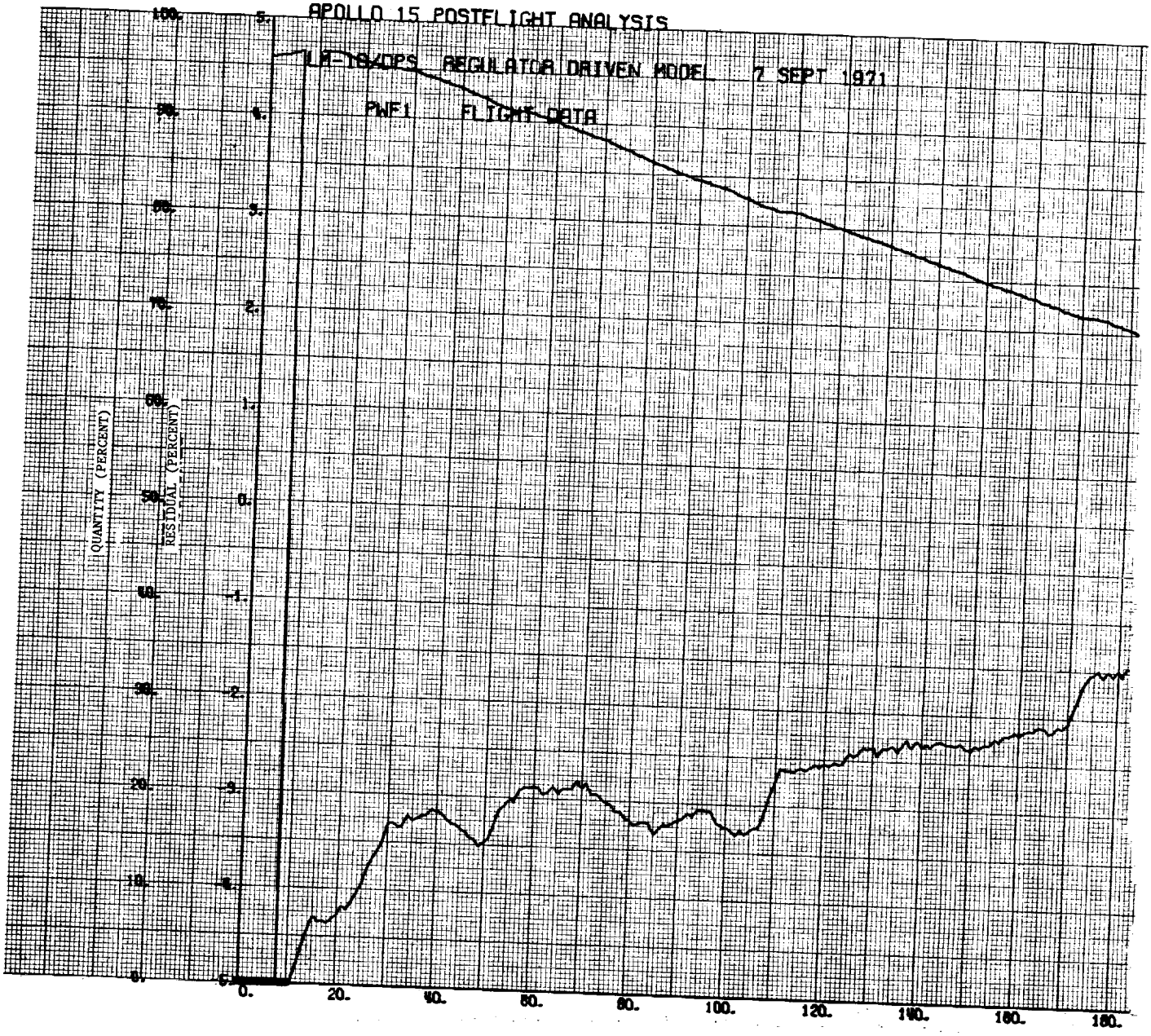


FIGURE 29

PQGS MATCH WITH NON-CALIBRATED DATA
OXIDIZER TANK NO. 2


APOLLO 15 POSTFLIGHT ANALYSIS
LM-ISA-CDPS REGULATOR DRIVEN MODEL 7 SEPT 1971
PWFI FLIGHT DATA
QUANTITY (PERCENT)
RESIDUAL (PERCENT)
100.
90.
80.
70.
60.
50.
40.
30.
20.
10.
0.
5.
4.
3.
2.
1.
0.
-1.
-2.
-3.
-4.
-5.
0.
20.
40.
60.
80.
100.
120.
140.
160.
180.

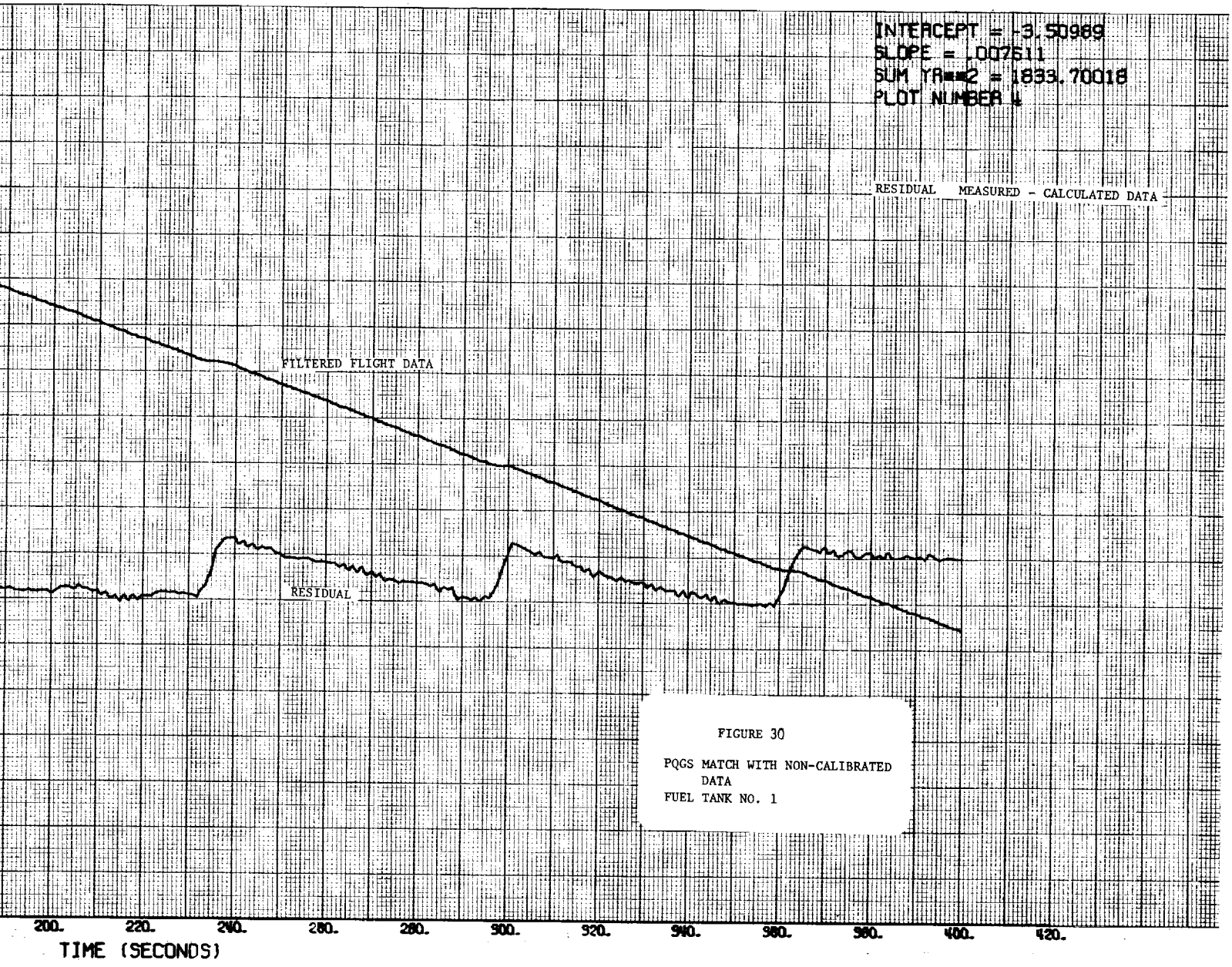
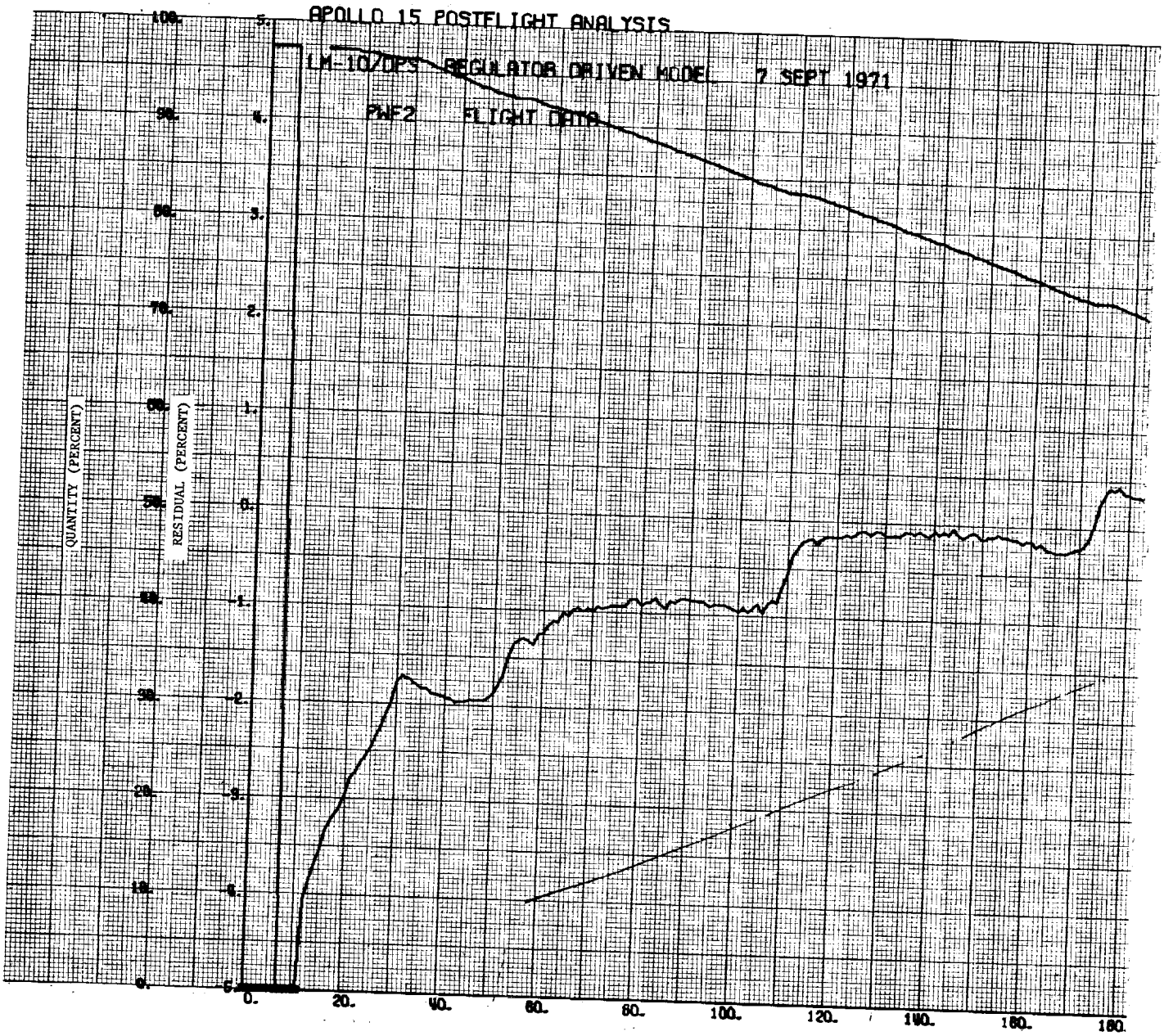


FIGURE 30

PQGS MATCH WITH NON-CALIBRATED DATA
FUEL TANK NO. 1


APOLLO 15 POSTFLIGHT ANALYSIS
LM-10/DPS REGULATOR DRIVEN MODEL 7 SEPT 1971
PWF2 FLIGHT DATA
QUANTITY (PERCENT)
RESIDUAL (PERCENT)
100.
90.
80.
70.
60.
50.
40.
30.
20.
10.
0.
5.
4.
3.
2.
1.
0.
-1.
-2.
-3.
-4.
-5.
0.
20.
40.
60.
80.
100.
120.
140.
160.
180.

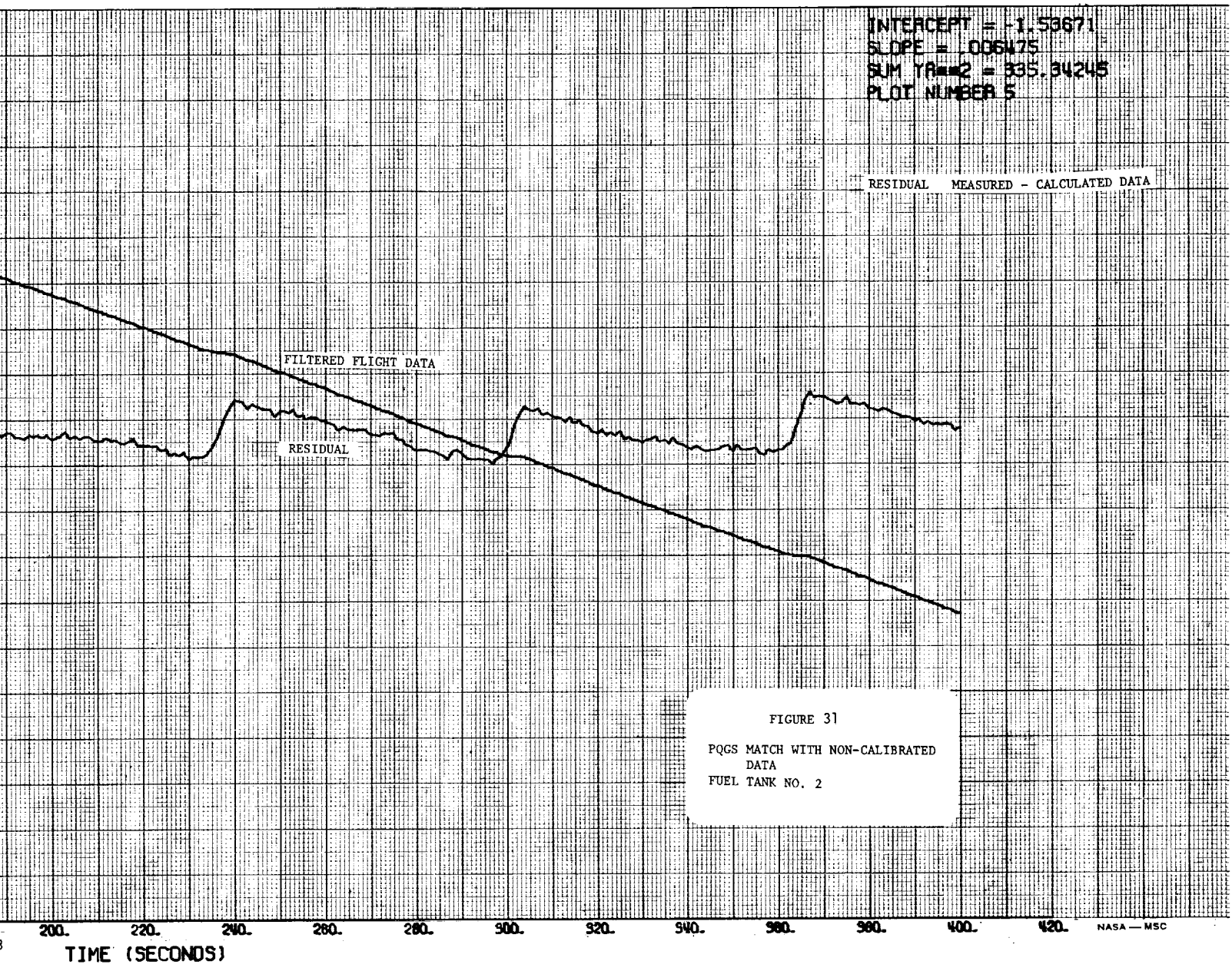


FIGURE 31

PQGS MATCH WITH NON-CALIBRATED DATA
FUEL TANK NO. 2