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APOLLO 16 MISSION REPORT

SUPPLEMENT 4

DESCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

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LM-11

DESCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

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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 16 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 16 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 16 Mission.

The following items are the major additions and changes to the preliminary 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-11 Descent Propulsion System during the Apollo 16 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 of 0.2 seconds. The engine performance corrected to standard inlet conditions for the FTP portion of the burn at 50 seconds after ignition was as follows: thrust, 9839 lbf; specific impulse, 306.9 sec; and propellant mixture ratio, 1.592. These values are +0.34, +0.03 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 noticeable drift, exhibiting a maximum error of about 1.5 psi at approximately 130 sec after engine ignition. This drift is due to thermal effects. Apparently, as the transducer temperature increases, its calibration "wanders." Larger errors occurred during the Apollo 14 and Apollo 15 DPS descent burns. Other flights have also had transducer drifts of smaller magnitude (less than 1 psi) except for LM-10 which had a drift of 5 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 (-0.35 and -1.70 psi for oxidizer and fuel, respectively). 3) The fuel propellant quantity gaging system did not perform within expected accuracies. The fuel probes indicated low during the entire burn with the Fu 1 and Fu 2 gaging showing a maximum gageable error of about nine percent at 70 seconds after ignition to about one percent at touchdown. These biases (seen as residual error) are shown in Figures 8 and 9.

The low level sensor activated at touchdown about 20 seconds earlier than predicted and is believed to be due to propellant slosh at landing.

3. INTRODUCTION

The Apollo 16 Mission was the ninth flight and the eighth manned flight of the Lunar Module (LM). The mission was the fifth successful lunar landing.

The space vehicle was launched from Kennedy Space Center (KSC) at 12:54:00 a.m. (EST) on 16 April 1972. Translunar injection was performed at 2 hours and 33 minutes (G.E.T.) after launch. Approximately 48 minutes after injection into the translunar trajectory, transposition and docking occurred. During the translunar coast, 1 midcourse correction was made using the SPS engine. Lunar orbit insertion was performed by the SPS engine at about 74 hours 28 minutes (G.E.T.) into the mission. At about 78 hours 34 minutes (G.E.T.) the Descent orbit insertion maneuver was performed with the SPS, bringing the CSM/LM into an orbit 11 miles above the landing site. CSM-LM separation occurred about 96 hours 34 minutes (G.E.T.) At 104:17:25 (G.E.T.), the Descent Burn (PDI) was initiated and lasted about 734 sec. The burn was started at the 20 percent throttle setting and after approximately 26 sec., the thrust was increased to the fixed throttle position (FTP). An automatic descent was maintained to approximately 676 seconds after ignition, at which time the crew 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 104:29:38 G.E.T. ending the DPS mission duty cycle. After a lunar stay of approximately 71 hours, the APS was ignited and the ascent stage of the LM was put into lunar orbit. Data from the DPS was terminated.

The actual ignition and shutdown times for the DPS firing are 104:17:23.6 G.E.T. and 104:29:37.7 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 16 Mission utilized LM-11 which was equipped with DPS engine S/N 1036. 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 not attempted due to the rapid changes in the engine thrust experienced during this portion of the burn making a detailed analysis impossible. 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 incorporated 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.

FTP Analysis Results

The engine performance during the FTP portion of the Descent Burn was satisfactory. The engine's inflight throat erosion characteristics were close to predicted, being 1.1 percent lower at the end of FTP than predicted (4.9 percent vs. 6.0 percent). This is within the 3 sigma uncertainty of $\pm$1.9 percent. The engine inflight specific impulse was 306.9 sec., 0.1 sec. higher than predicted. The 3 sigma uncertainty is $\pm$0.6 sec. The inflight thrust was 9839 lbf, 33 lbf higher than predicted but within the $\pm$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 92 lbm of consumables other than the DPS propellant were used. Of that amount, 59 lbm were RCS propellant. Since there was little RCS activity during the analyzed portion of the burn, it was assumed that during that segment, the non-DPS consumed weight was used at a rate of 0.05 lbm/sec.

The DPS steady-state FTP performance was determined from the analysis of a 409 second segment of the burn. The segment of the burn analyzed commenced approximately 30 seconds after DPS ignition (FS-1) and included the flight time between 104:17:56 hours and 104:24:45 hours ground elapsed time. Engine throttle down to 60 percent occurred 3 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 predicted values. The values presented are approximate end point conditions of the segment analyzed and are considered representative of the actual flight values throughout the segment. In general, the actual (calculated) 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 4.9 percent or 1.1 percent less than the predicted value of 6.0 percent. 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 measurements for oxidizer tanks 1 and 2, quantity gaging system measurements for fuel tanks 1 and 2, and chamber pressure. The chamber pressure plot indicates how poor the chamber pressure measurement was as a source of data. Because of this, chamber pressure was not used in the PAP analysis 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) 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 zero 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 higher than that from GQ 3018P. The pressure measurement biases determined by the program were +0.95 psi for GQ 3018P and -1.41 psi for GQ 3025P.

The inflight value of the fuel interface pressure (GQ 4111P) was biased by -1.73 psi, although this is within the instrument accuracy. The oxidizer interface pressure was also biased by -0.36 psi.

The gaging system data (Figures 16-19) could not be used in the analysis. Although the oxidizer PQGS data was within the expected accuracy (see Section 7), the data was not of sufficient quality to include the measurement in the PAP analysis. At 60 seconds after ignition, the fuel gages read approximately 10 percent low and although the data improved as the burn progressed, there was not sufficient confidence in the gages to use them in the analysis as measurement variables. 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 for low level sensor comparisons and propellant depletion calculations.

Comparison with Preflight Performance Predictions

Prior to the Apollo 16 Mission the expected inflight performance of the DPS was presented in Reference 3. The preflight performance 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 conditions. Comparing the corrected engine flight performance at FTP during the Descent Burn to the corrected ground test data shows the flight data to be 0.34 percent, 0.03 percent, and 0.0 percent greater 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	9806	9839	9712 - 10027	9742 - 9840
Specific Impulse, sec	306.8	306.9	> 305.0	306.1 - 307.6
Mixture Ratio	1.592	1.592	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 (GH1331V) determines the engine throttle setting. The program then calculates the 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 0.95 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 + 441 seconds) and continued for 292 seconds. This includes all of the powered descent burn after throttle down and includes the flight time between 104:24:45 hours to 104:29:37 hours G.E.T. 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 percent, mixture ratio, and specific impulse.

Figures 15 through 19 present the inflight values of several measured propulsion parameters. Because of the large amount of machine time required to plot the inflight measured parameters, some parameters were deleted from the report. For Figure 15, measured chamber pressure, 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 0.05 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[1] Isp(sec)	Engine Effective[1] Isp(sec)
Preflight Prediction	11096.0	6965.0	1.593	302.9	305.8
Analysis Program	11180.3	7014.1	1.594	302.2	305.7

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 6731.8 ft/sec. The total measured thrust velocity gain, 6734.3 ft/sec., includes the contribution of both the DPS engine and RCS activity. The best estimate of the actual velocity gain was reported in the Apollo 16 Mission Report (Reference 1) as 6705 ft/sec. The higher value from the analysis was due to an inaccurate acceleration match during the last portion of the throttling region

1 Calculated from FS-1 plus 30 seconds.

simulation caused by data dropout. The difference between these values of thrust gain had negligible effect on the analysis of the throttled performance. 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 the analysis is within this uncertainty.

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

7. PQGS EVALUATION AND PROPELLANT LOADING

As mentioned in Section 4, the PQGS measurements for Apollo 16 were not used in the PAP program as active measurement inputs. The residual errors (difference between the measured and calculated values) for the FTP portion of the burn are shown in Figures 6 through 9. Figures 16 through 19 present the flight data and the best estimate of the actual propellant quantities during the Descent Burn.

At 70 seconds after engine ignition, both fuel gages read considerably lower (a total of about 450 lbm) than expected. As the burn progressed the differences between measured and actual quantities decreased. This large discrepancy at the beginning of the Descent Burn has occurred on previous missions. The cause is apparently due to a chemical reaction between the fuel and a protective coating on the gaging probe. This reaction results in a localized change in the fuel conductivity which effects the gaging reference sensor differently than the measurement probe and generates an erroneous quantity value. As the fuel tanks deplete, the inaccuracies of the system decrease due both to the mixing of the fuel and to the inherent increased accuracy at lower quantities.

At the end of the analyzed portion of the FTP burn (approximately 30 percent remaining quantity), the difference between the measured and calculated propellant liquid levels were -0.6, 0.6, -2.5, -2.0 percent for the Ox 1, Ox 2, Fu 1 and Fu 2, respectively. At the end of Descent Burn, the differences were -0.4, 0.2, -1.3, and -0.8 percent, respectively.

The anticipated 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 WSTF tests indicated could 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, the oxidizer quantities were always within the expected accuracy of the gaging probe based on WSTF results. The fuel quantities were frequently outside the expected accuracy range. At engine shutdown, the quantities of propellants remaining in the tanks were computed to be 796.7 lbm and 491.0 lbm for oxidizer and fuel, respectively. Of these quantities, 755.6 lbm of oxidizer and 484.4 lbm of fuel are usable to depletion (including burning usable propellants in the feed lines). Applying the propellant flow rates at engine shutdown, 124 seconds of hover time remained based on computed residual propellants. The measured quantities indicate 109 seconds of hover time, that is, about 746 lbm of usable oxidizer and 416 lbm of usable fuel. The calculated data indicates an oxidizer depletion while the measured data indicated a fuel depletion.

The low level sensor activated at touchdown is believed to be due to the landing shock causing propellant sloshing. The low level sensor was activated about 20 seconds early.

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 90.24 lbm/ft^3 and an average fuel density of 56.40 lbm/ft^3 at pressures of 240 psia and temperatures of 70°F were determined from the samples. The propellant loads were 7505.1 lbm of fuel and 11977.0 lbm of oxidizer. The total DPS propellant onboard was 19482.1 lbm

8. PRESSURIZATION SYSTEM EVALUATION

The DPS Supercritical Helium (SHe) Pressurization System performed satisfactorily during the Apollo 16 mission. The data plotted in Figure 20 shows that the flight data generally falls within the predicted performance range (nominal ± 3 sigma).

A postflight simulation for the SHe system generated with the SHe program with flight data as input, is presented in Figure 21. The flight data used as input include: 1.) SHe bottle pressure at PDI; 2.) DPS engine cycle (throttle setting versus burn time, Figure 1); and 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	1194.	1249.	1249.	-55	---
Max. Pressure	1346.	1376.	1368.	-22	+8
Press at T/D	415.	417.	470.	-55	-53

Although the match during the first part of the DPS burn is good, the predictions indicate lower pressures 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 prelaunch and coast pressure rise rates for the SHe system were found to be 9.0 and 7.0 psia/hour, respectively.

9. ENGINE TRANSIENT ANALYSIS

The mission duty cycle of the Descent Propulsion System for Apollo 16 included one start at the 20 percent throttle setting, and one shutdown at approximately 27 percent 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 16 mission, and for comparison, similar parameters for the other Apollo missions which incorporated 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 0.7 seconds. The delay time compared favorably with the first burn delays observed during Apollo 13, 14 and 15.

The start transient from FS-1 to 90 percent of the minimum steady-state throttle setting required 2.10 seconds with a start impulse of 818 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 percent to 100 percent of the minimum throttle setting required 0.10 seconds with an impulse of 190 lbf-sec. The start impulse was greater than expected. This was because the engine was inadvertently set at 20 percent throttle at ignition rather than the planned minimum of 13 percent. Approximately seven seconds after ignition, the manual throttle was moved to the minimum thrust position.

The shutdown transient required 1.60 seconds from FS-2 to 10 percent of the steady-state throttle setting with an impulse of 946 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 (13 percent of full thrust) to FTP, which was executed 26 seconds into the burn, required approximately one second. The engine then remained at FTP for 418 seconds. The second command, from FTP to 59 percent, occurred 444 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 120 seconds. The LM Guidance Computer then commanded a ramping decrease in the throttle setting from 60 percent to 33 percent over 102 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 60 seconds of the burn, the LGC commanded many throttle changes in the 28 percent to 45 percent range. Data dropout made an exact count impossible. The command time from one throttle setting to the next was generally less than 0.30 second. 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 this portion of the burn, the response of the DPS engine was considered satisfactory.

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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.

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TABLE 1

LM-11 DESCENT PROPULSION ENGINE AND FEED SYSTEM PHYSICAL CHARACTERISTICS

ENGINE	
Engine Number	1036
Chamber Throat Area, in^2	53.781 [1]
Nozzle Exit Area, in^2	2937.6 [2]
Nozzle Expansion Ratio	54.0 [2]
FEED SYSTEM	
Oxidizer Propellant Tanks, Total Ambient [3] Volume, Ft^3	135.4 [2]
Fuel Propellant Tanks, Total Ambient Volume, Ft^3	135.4 [2]
Oxidizer Tank to Interface Resistance, $\frac{lbf\text{-}sec^2}{lbm\text{-}ft^5}$	422.9 [4]
Fuel Tank to Interface Resistance, $\frac{lbf\text{-}sec^2}{lbm\text{-}ft^5}$	673.6 [4]

[1] TRW IOC 4600.12-10-73, "Acceptance Test Performance Report, S/N 1036," F. E. Amon, 9 September 1970.

[2] Approximate Values

[3] 14.7 PSIA and 70°F

[4] GAEC Cold Flow Tests

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

LM-11 DESCENT PROPULSION SYSTEM STEADY-STATE FTP PERFORMANCE

PARAMETER	FS-1 + 50 SECONDS			FS-1 + 430 SECONDS		
INSTRUMENTED	PREDICTED	MEASURED	CALCULATED	PREDICTED	MEASURED	CALCULATED
Regulator Outlet Pressures, psia	243.8	241.4 (1) 243.5 (2)	241.9	244.1	241.9 244.2	242.6
Oxidizer Interface Pressure, psia	225.2	222.9	223.3	224.4	222.5	222.9
Fuel Interface Pressure, psia	225.0	221.2	222.9	224.4	221.0	222.8
Engine Chamber Pressure, psia	106.0	104.7	105.4		101.9	101.5
Oxidizer Bulk Temperature, Tank No. 1, °F	68	65	---	68	65	---
Oxidizer Bulk Temperature, Tank No. 2, °F	68	67	---	68	67	---
Fuel Bulk Temperature, Tank No. 1, °F	68	66	---	68	66	---
Fuel Bulk Temperature, Tank No. 2, °F	68	66	---	68	66	---
DERIVED						
Oxidizer Flowrate, lbm/sec	19.77	---	19.73	20.06	---	20.01
Fuel Flowrate, lbm/sec	12.41	---	12.37	12.60	---	12.58
Propellant Mixture Ratio	1.594	---	1.597	1.591	---	1.592
Vacuum Specific Impulse, sec	307.3	---	307.3	305.0	---	305.2
Vacuum Thrust, lbf	9889	---	9866	9965	---	9944
Throat Erosion, %	-1.37	---	-1.42	5.56	---	4.60

(1) GQ 3018P
(2) GQ 3025P

TABLE 4

LM-11 DESCENT PROPULSION SYSTEM THROTTLED PERFORMANCE

PARAMETER	FS-1 + 465 SECONDS			FS-1 + 605 SECONDS		
INSTRUMENTED	PREDICTED	MEASURED	SIMULATION	PREDICTED	MEASURED	SIMULATION
Regulator Outlet Pressure, psia	246.0	241.9	242.5	244.9	241.9	242.9
Oxidizer Interface Pressure, psia	235.9	234.2	233.8	238.0	234.6	236.0
Fuel Interface Pressure, psia	236.0	234.5	233.9	238.0	235.9	236.0
Engine Chamber Pressure, psia	61.2	58.5	57.9	48.8	50.0	50.4
Oxidizer Bulk Temperature, Tank No. 1, °F	68	65		68	65	
Oxidizer Bulk Temperature, Tank No. 2, °F	68	67		68	67	
Fuel Bulk Temperature, Tank No. 1, °F	68	66		68	66	
Fuel Bulk Temperature, Tank No. 2, °F	68	66		68	66	
Throttle Command Voltage, VDC	0	8.02	8.02	-	7.30	7.30
DERIVED						
Oxidizer Flowrate, lbm/sec	11.97	---	11.39	9.83	---	10.3
Fuel Flowrate, lbm/sec	7.50	---	7.14	6.17	---	6.48
Propellant Mixture Ratio, O/F	1.595	---	1.595	1.593	---	1.594
Vacuum Specific Impulse, sec	305.6	---	304.6	301.8	---	301.9
Vacuum Thrust, lbf	5952	---	5644	4827	---	5057
Throat Erosion, %	6.50	---	5.46	8.37	---	9.32

TABLE 5

LM-11 DPS PROPELLANT QUANTITY GAGING SYSTEM PERFORMANCE

Parameter	Time (From Descent Burn Ignition), sec							
	70	170	270	370	450	530	630	732
Oxidizer Tank No. 1								
Measured Quantity, percent	96.2	79.2	61.7	43.8	30.3	22.0	13.0	6.6
Calculated Quantity, percent	96.6	79.5	62.2	44.9	30.9	22.8	13.0	7.0
Difference, percent	-0.4	-0.3	-0.4	-0.9	-0.6	-0.8	-0.0	-0.4
Oxidizer Tank No. 2								
Measured Quantity	97.0	80.5	62.9	45.4	31.5	23.2	13.6	7.2
Calculated Quantity, percent	96.6	79.5	62.3	44.7	30.9	22.6	13.0	7.0
Difference, percent	0.4	1.0	0.6	0.7	0.6	0.6	0.6	0.2
Fuel Tank No. 1								
Measured Quantity, percent	88.3	75.8	59.1	41.7	28.3	20.7	11.0	5.6
Calculated Quantity, percent	96.6	79.3	61.7	44.7	30.8	22.3	13.2	6.9
Difference, percent	-8.3	-3.5	-2.6	-3.0	-2.5	-1.6	-2.2	-1.3
Fuel Tank No. 2								
Measured Quantity, percent	87.7	75.7	59.8	42.7	28.9	21.1	11.5	6.1
Calculated Quantity, percent	96.6	79.4	61.9	44.5	30.9	22.6	13.0	6.9
Difference, percent	-8.9	-3.7	-2.1	-1.8	-2.0	-1.5	-1.5	-0.8

TABLE 6

DPS START AND SHUTDOWN IMPULSE SUMMARY

	Apollo 16 LM-11/DPS-1	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	20.0	13.1	13.1	16.2	13.1	
Total Vacuum Start Impulse (FS-1 to 90% steady-state), lbf-sec	818	440	710	591	728	
Start Time (FS-1 to 90% steady-state), sec	2.10	2.34	2.14	1.77	2.13	4.0
Coast Time from Prior Burn, Minutes	From Launch	From Launch	From Launch	56	72	
SHUTDOWNS						
Steady-State Throttle Position, Percent	26.6	29.0	27.0	23.4	FTP	
Total Vacuum Shutdown Impulse (FS-2 to 10% Steady-State), lbf-sec	946	1113	976	1540	2041	
Shutdown Time (FS-2 to 10% steady-state), sec	1.60	2.06	1.23	2.06	0.34	0.25 [1]
Repeatability, lfb-sec						±100 [1]
Total Vacuum Shutdown Impulse (FS-2 to Zero Thrust) from Velocity Gain Data, lbf-sec	--- [2]	--- [2]	--- [2]	--- [2]	2948	

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

[2] Not applicable to lunar landing shutdown.

TABLE 6 (Continued)

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[3]	2.1	2.3[3]	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[3]	1.1	1.8[3]	0.26	0.30	0.25[1]
Repeatability, lbf-sec				1734±7	1734±7	±100[1]
Total Vacuum Shutdown Impulse (FS-2 to Zero Thrust) From Velocity Gain Data, lbf-sec	1777	---[4]	1040	2493	---[5]	

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

[2] Not applicable to lunar landing shutdown.

[3] Reference 5.

[4] Data Unavailable.

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

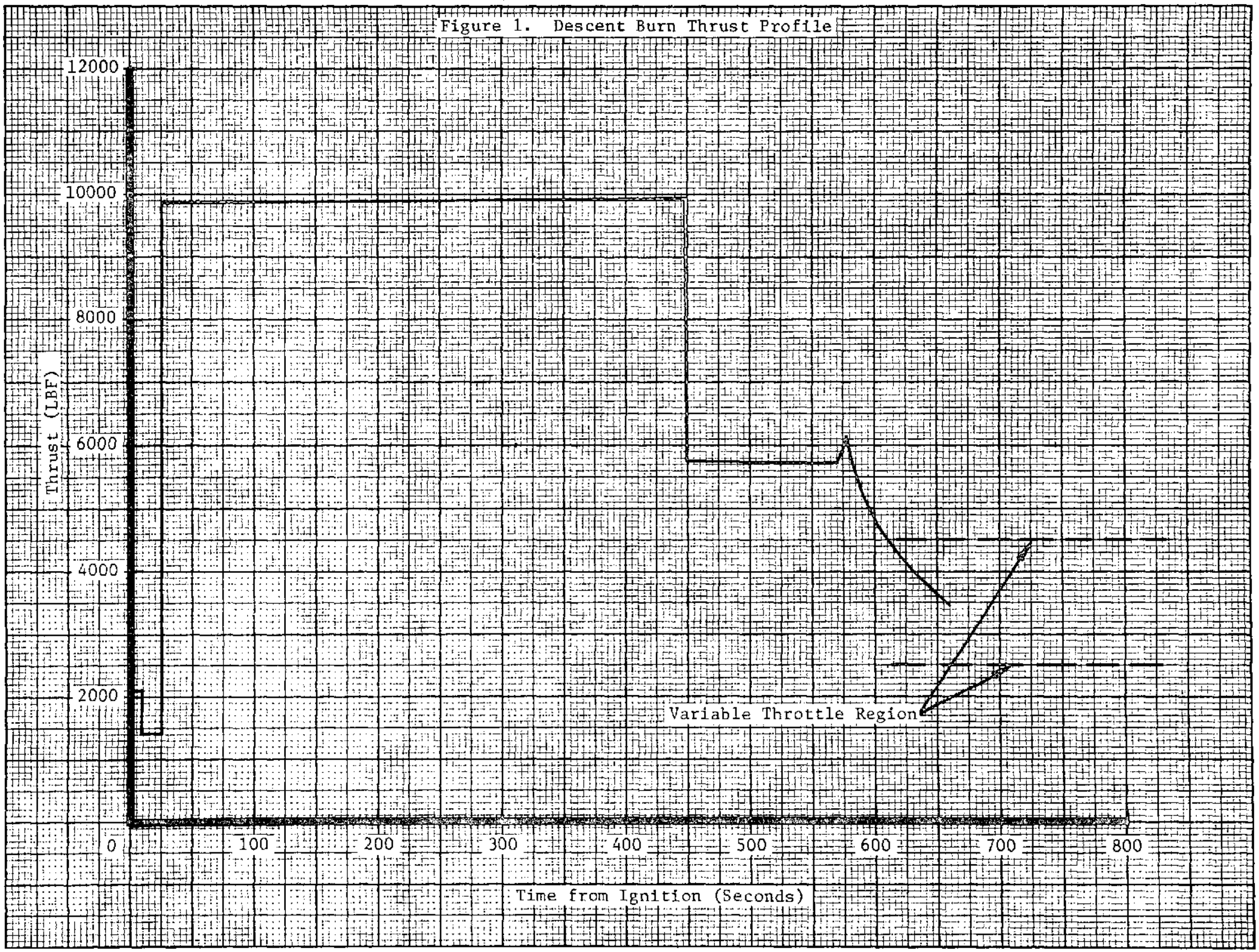
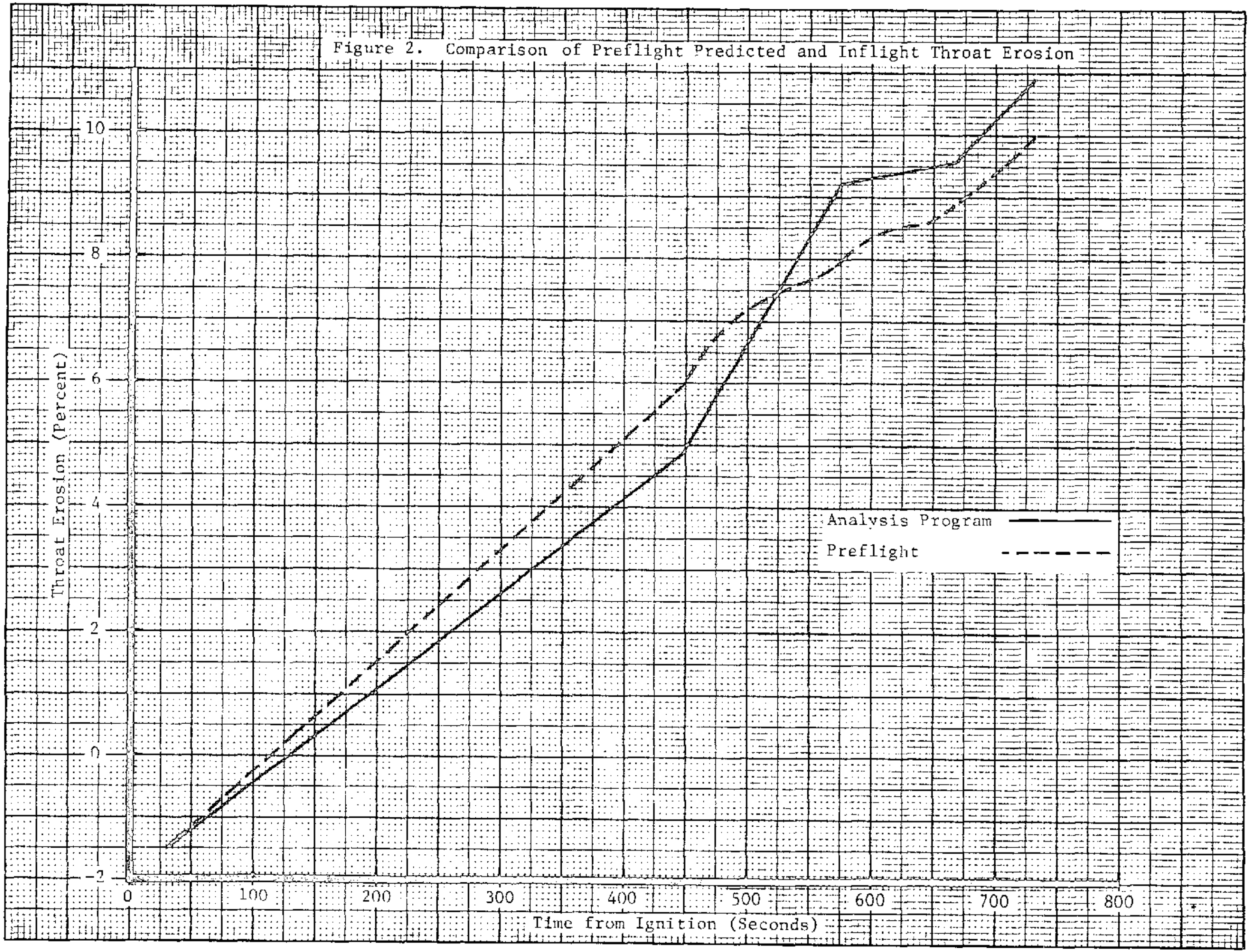


Figure 1. Descent Burn Thrust Profile

Figure 2. Comparison of Preflight Predicted and Inflight Throat Erosion



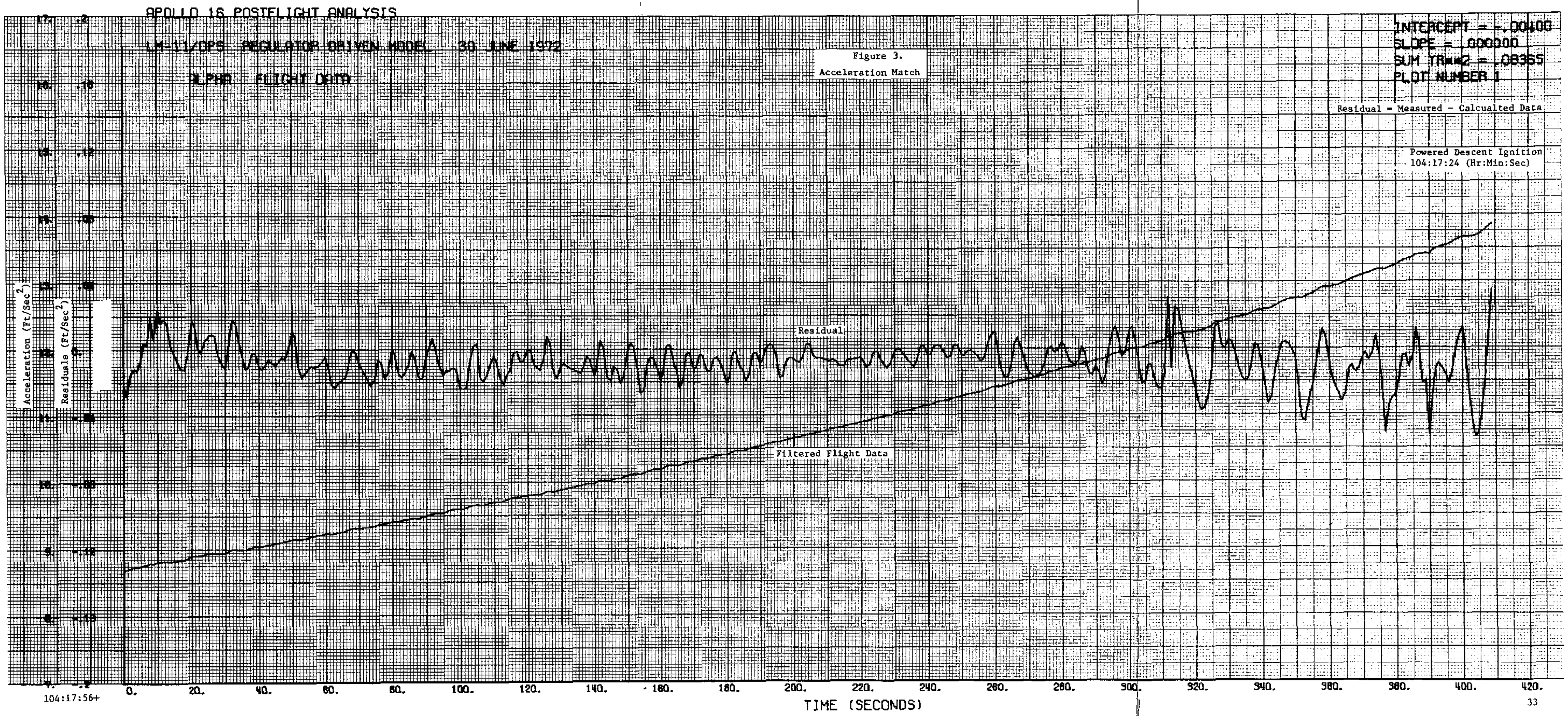


Figure 3.
Acceleration Match

FOLDOUT FRAME 1

FOLDOUT FRAME 2

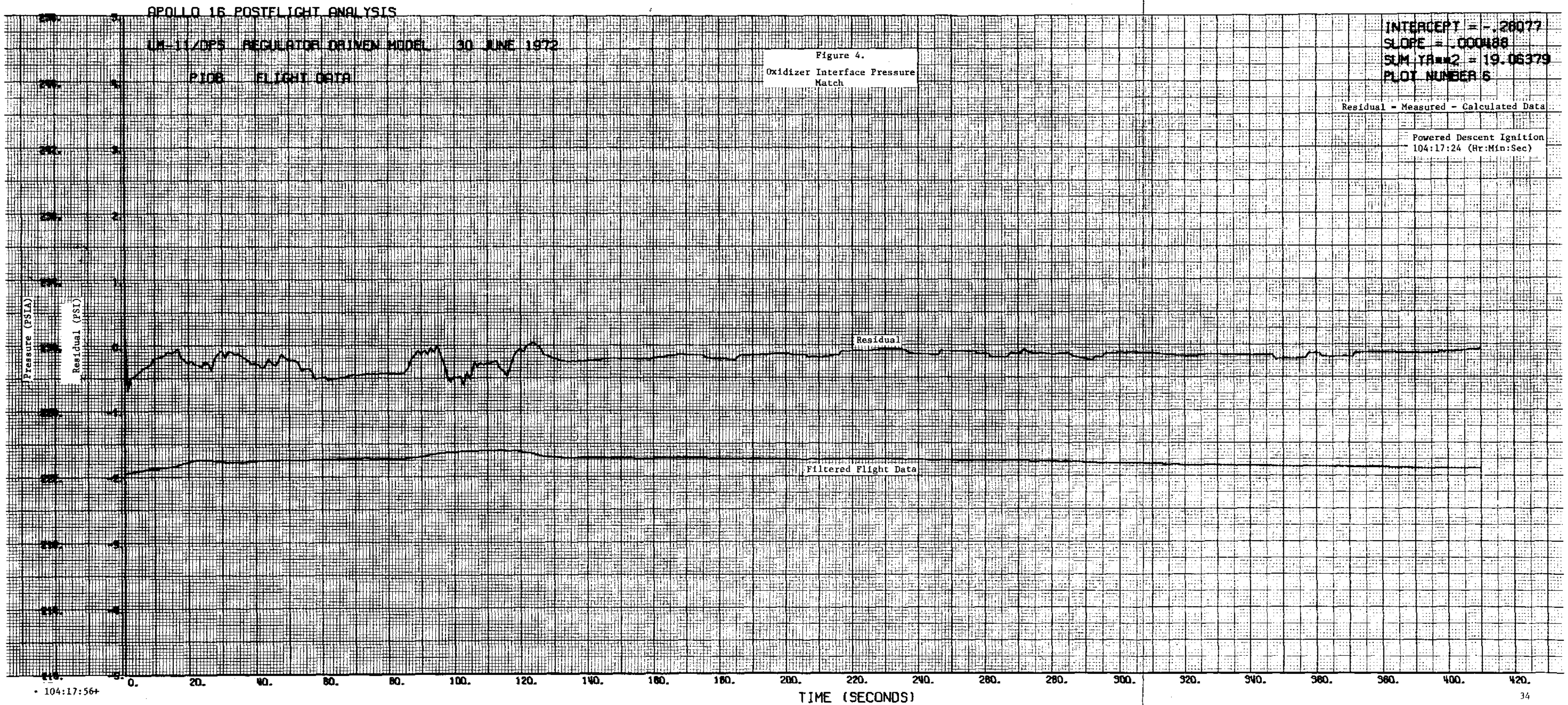


Figure 4. Oxidizer Interface Pressure Match

FOLDOUT FRAME 1

FOLDOUT FRAME 2

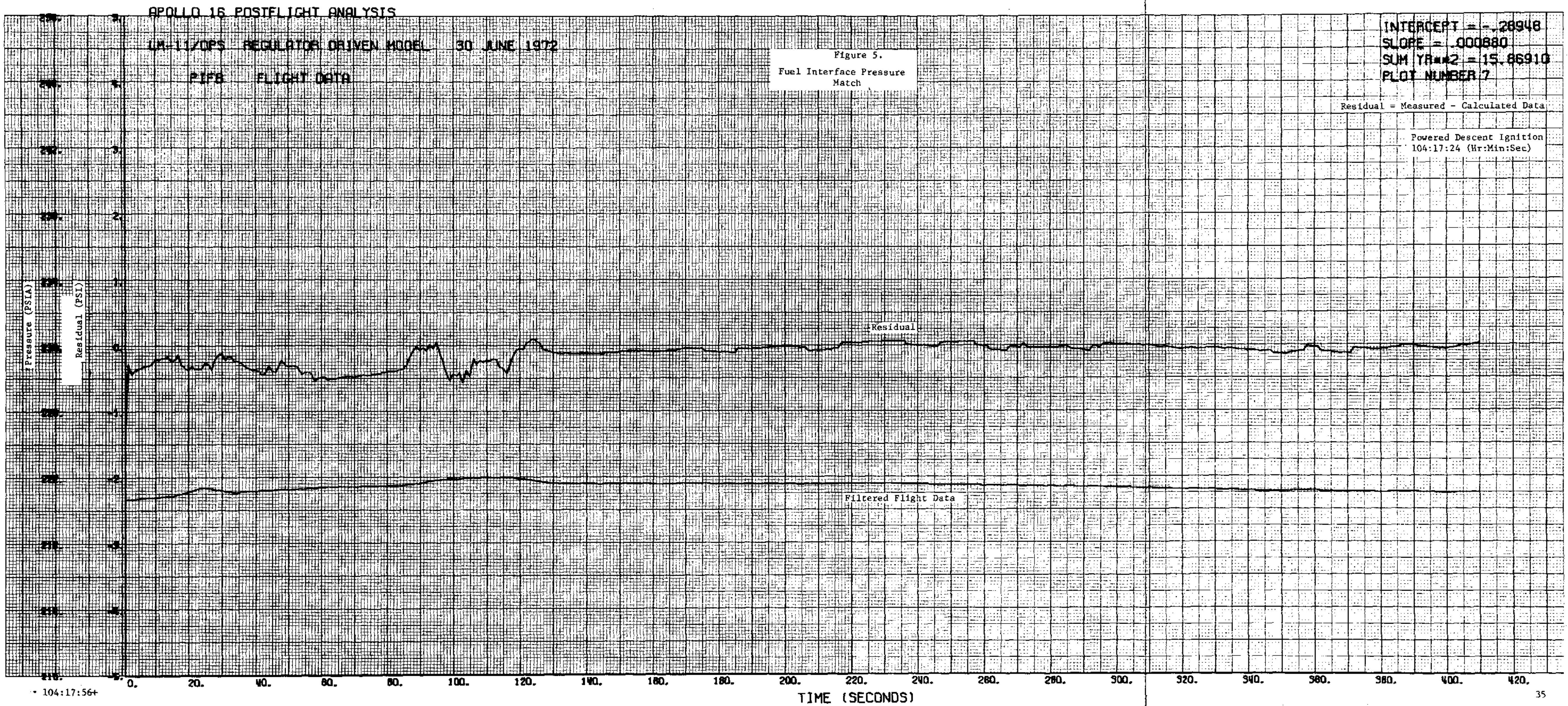
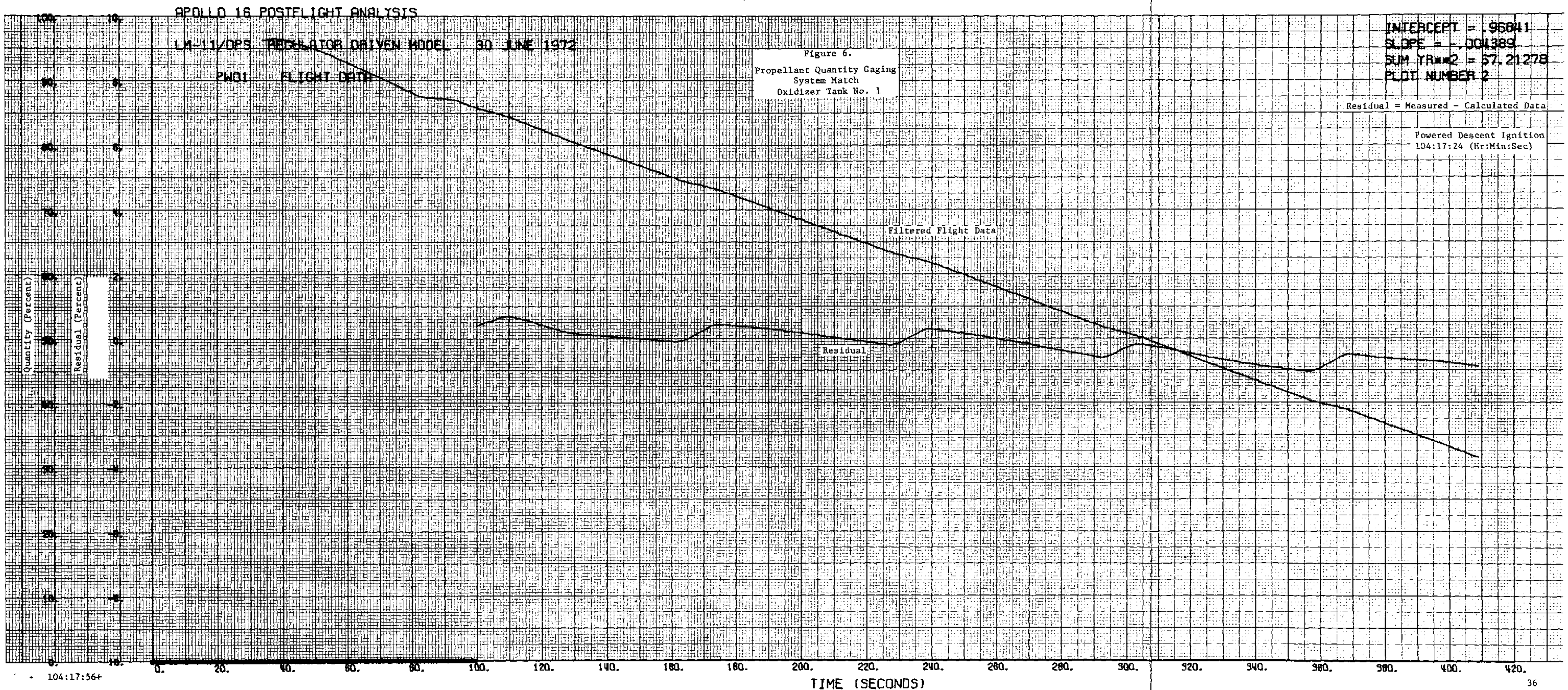


Figure 5.
Fuel Interface Pressure Match

FOLDOUT FRAME /

FOLDOUT FRAME 2



APOLLO 16 POSTFLIGHT ANALYSIS
LM-11/DPS REGULATOR DRIVEN MODEL 30 JUNE 1972
PWD1 FLIGHT DATA
Figure 6.
Propellant Quantity Gaging System Match Oxidizer Tank No. 1
INTERCEPT = .96841
SLOPE = -.004389
SUM YR**2 = 57.21278
PLOT NUMBER 2
Residual = Measured - Calculated Data
Powered Descent Ignition 104:17:24 (Hr:Min:Sec)
Filtered Flight Data
Residual
Quantity (Percent)
Residual (Percent)
104:17:56+
TIME (SECONDS)

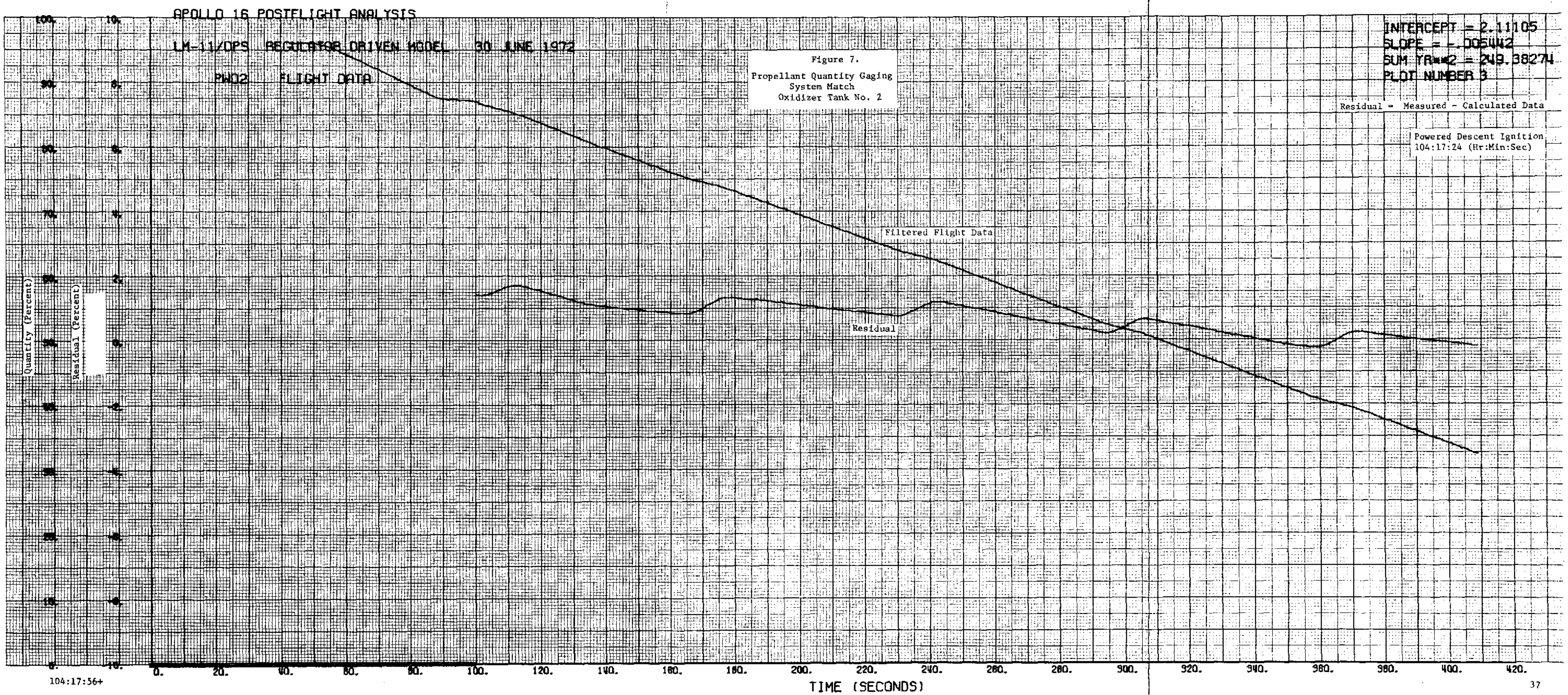


Figure 7.
Propellant Quantity Gaging System Match
Oxidizer Tank No. 2

FOLDOUT FRAME 1

FOLDOUT FRAME 1

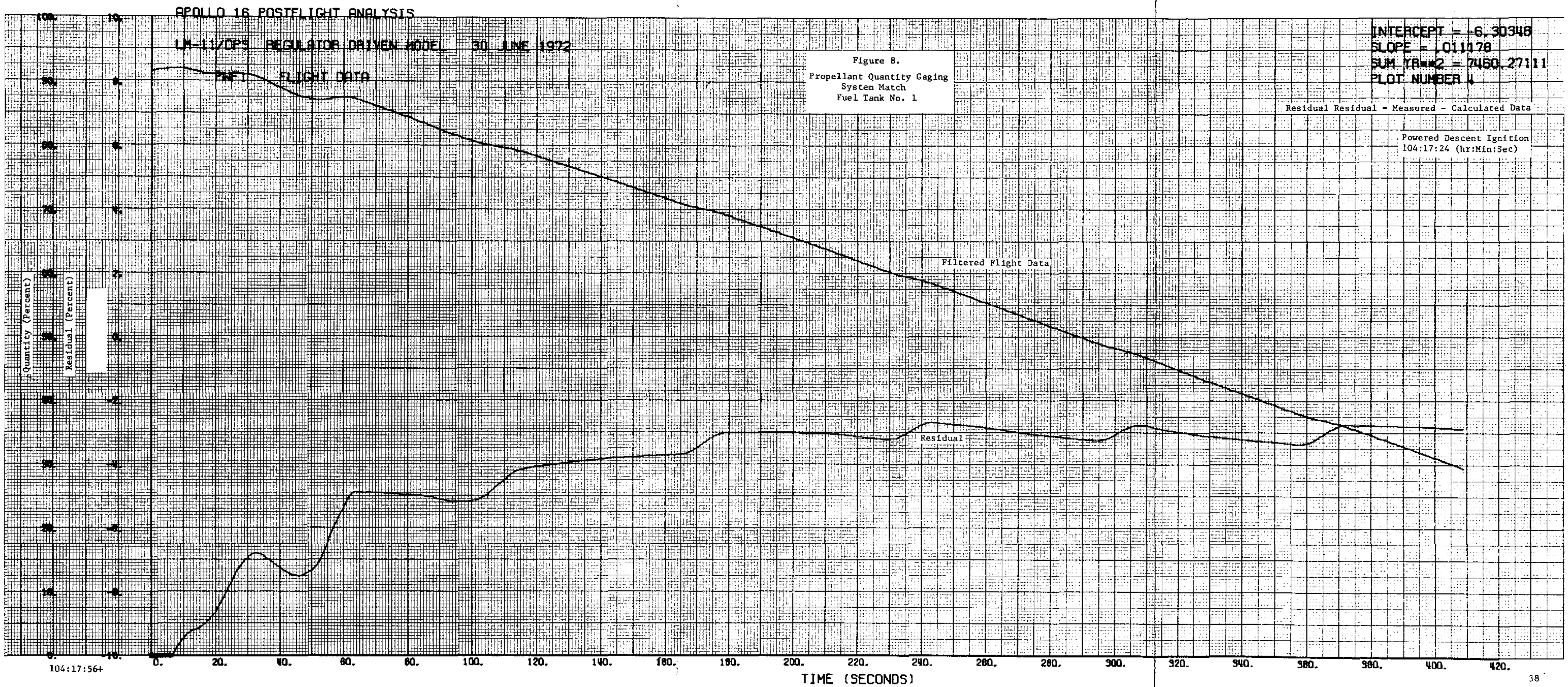


Figure 8.
Propellant Quantity Gaging System Match
Fuel Tank No. 1

FOLDOUT FRAME 1

FOLDOUT FRAME 2

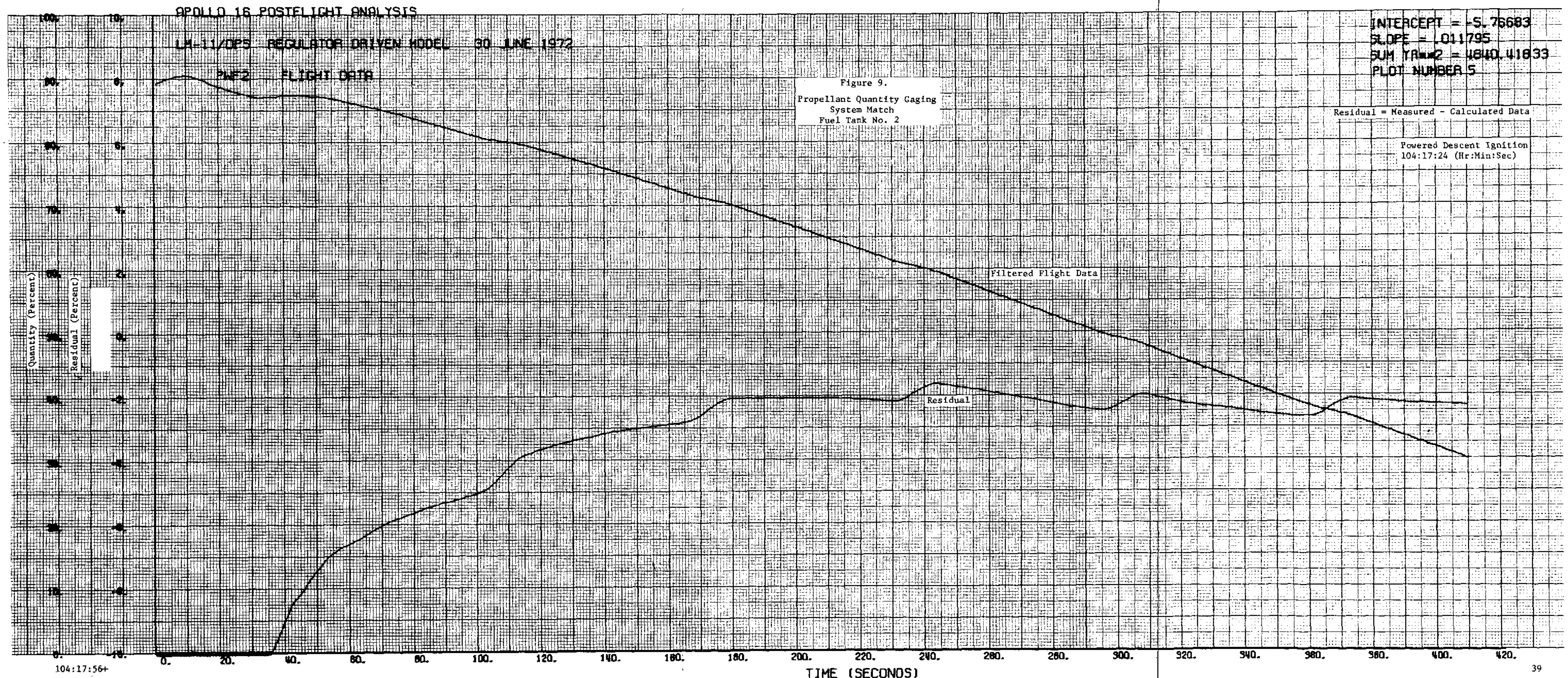


Figure 9.
Propellant Quantity Gaging System Match Fuel Tank No. 2

FOLDOUT FRAME 1

FOLDOUT FRAME 2

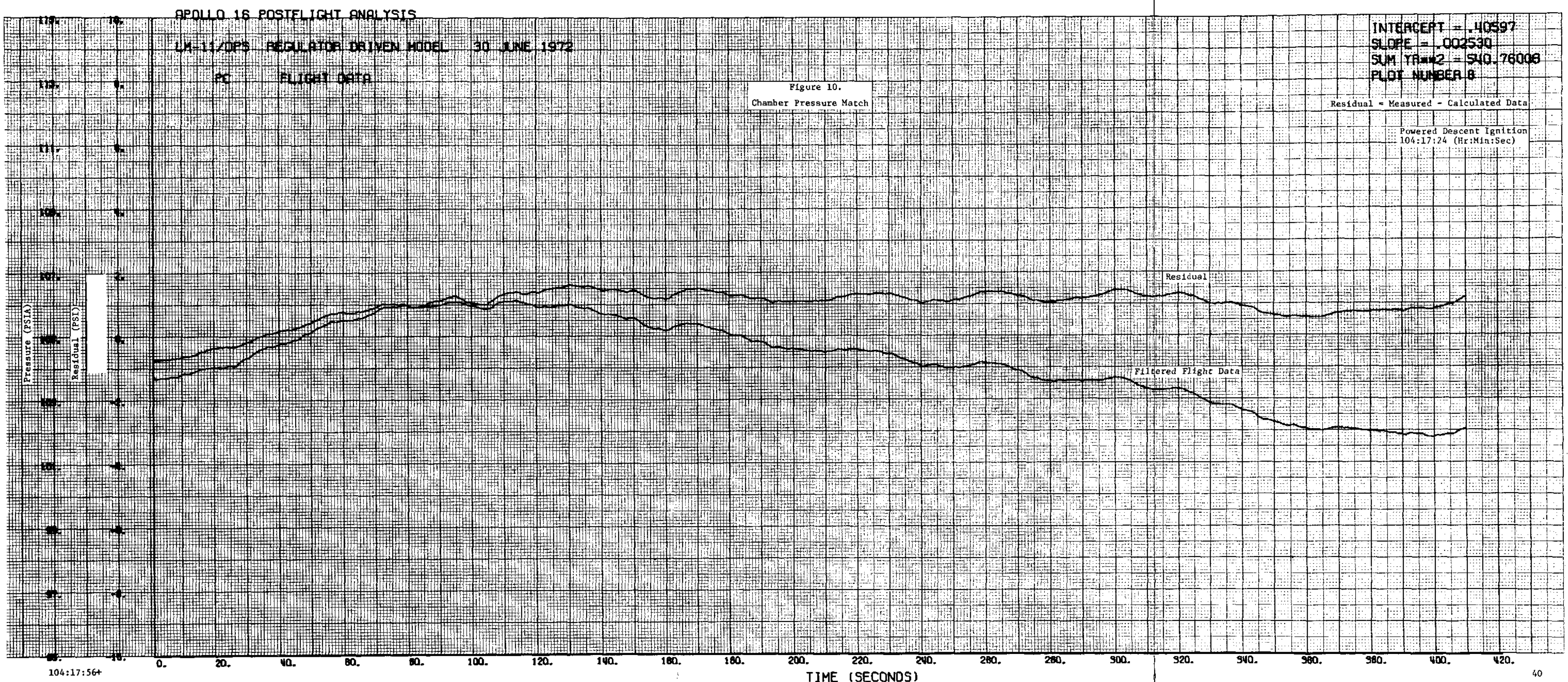
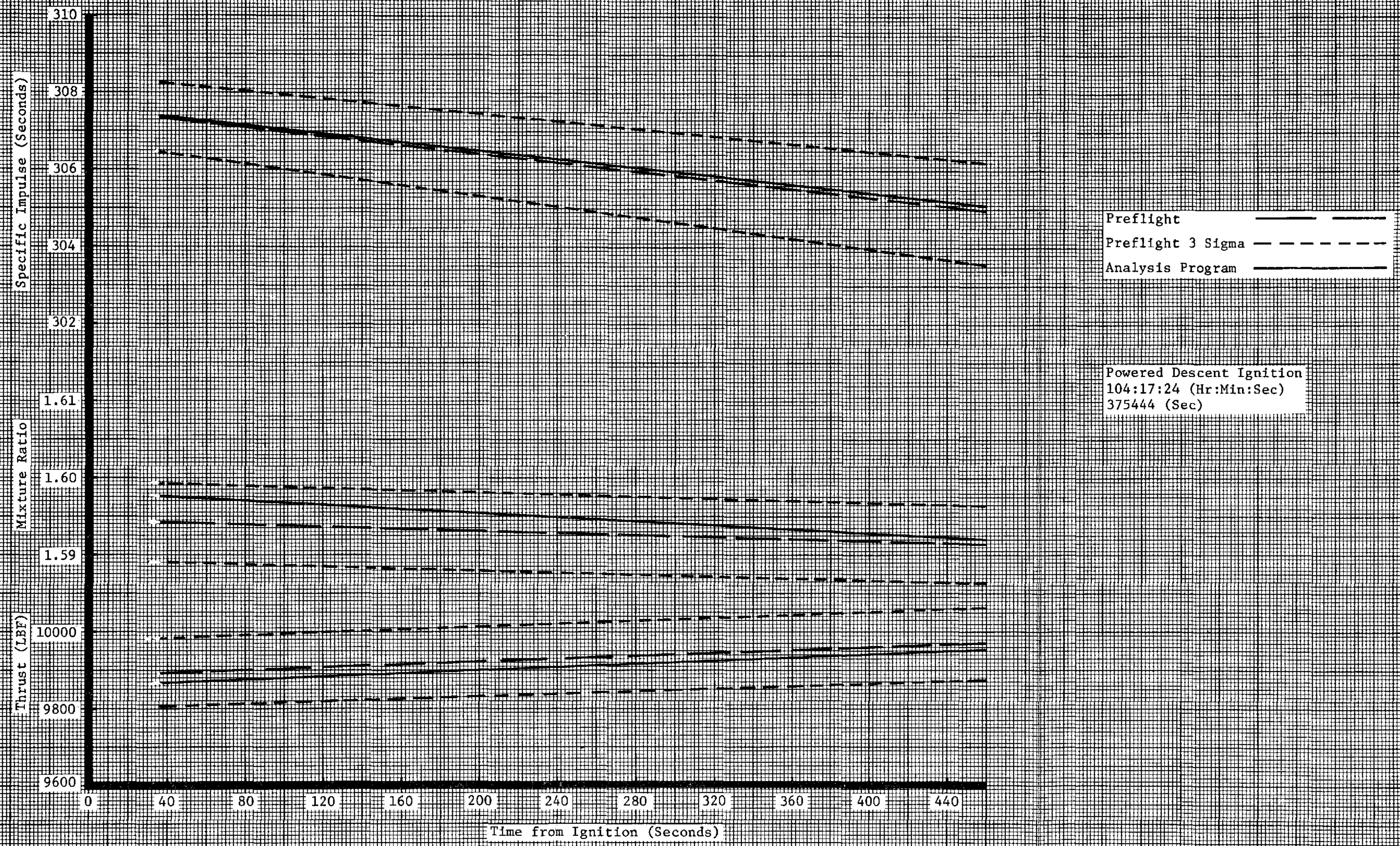


Figure 10.
Chamber Pressure Match

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Figure 11. Comparison of FTP Preflight Predicted and Actual Performance



FOLDOUT FRAME 1

FOLDOUT FRAME 2

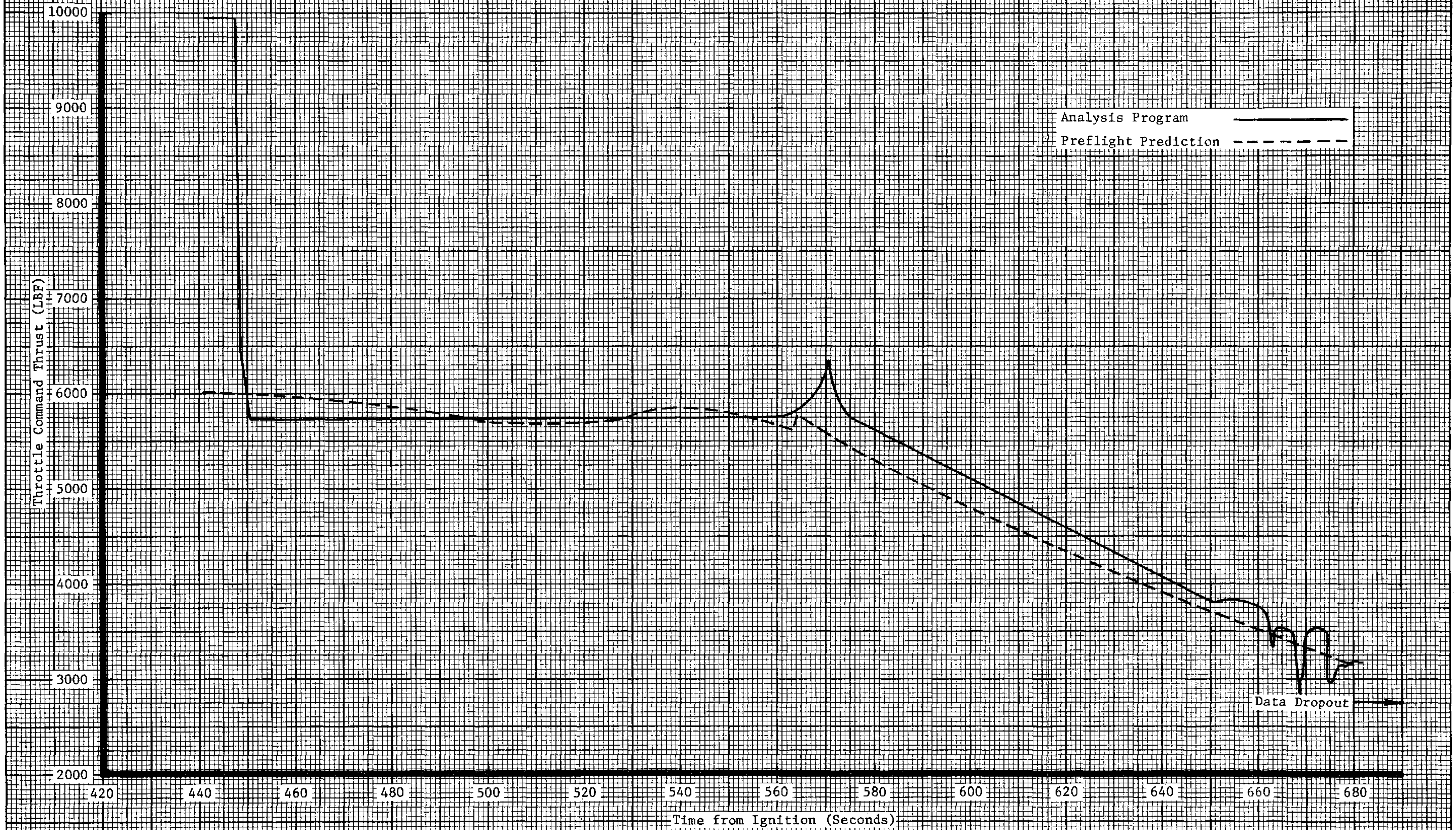


Figure 12. Comparison of Preflight Predicted and Analysis Program Simulated Throttle Command Thrust

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Figure 13. Comparison of Preflight Predicted and Analysis Program Simulated Engine Mixture Ratio (Throttled Region)

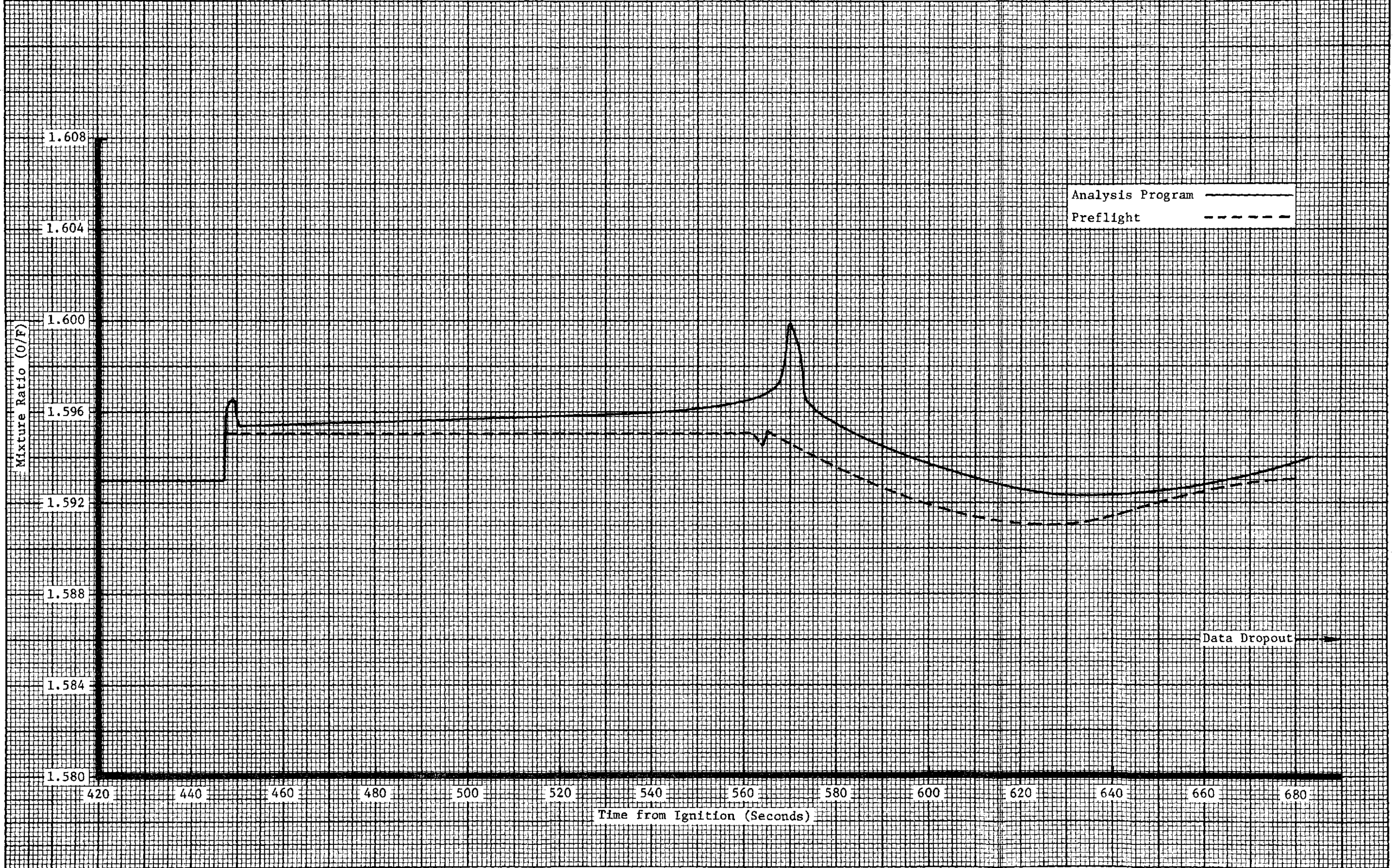
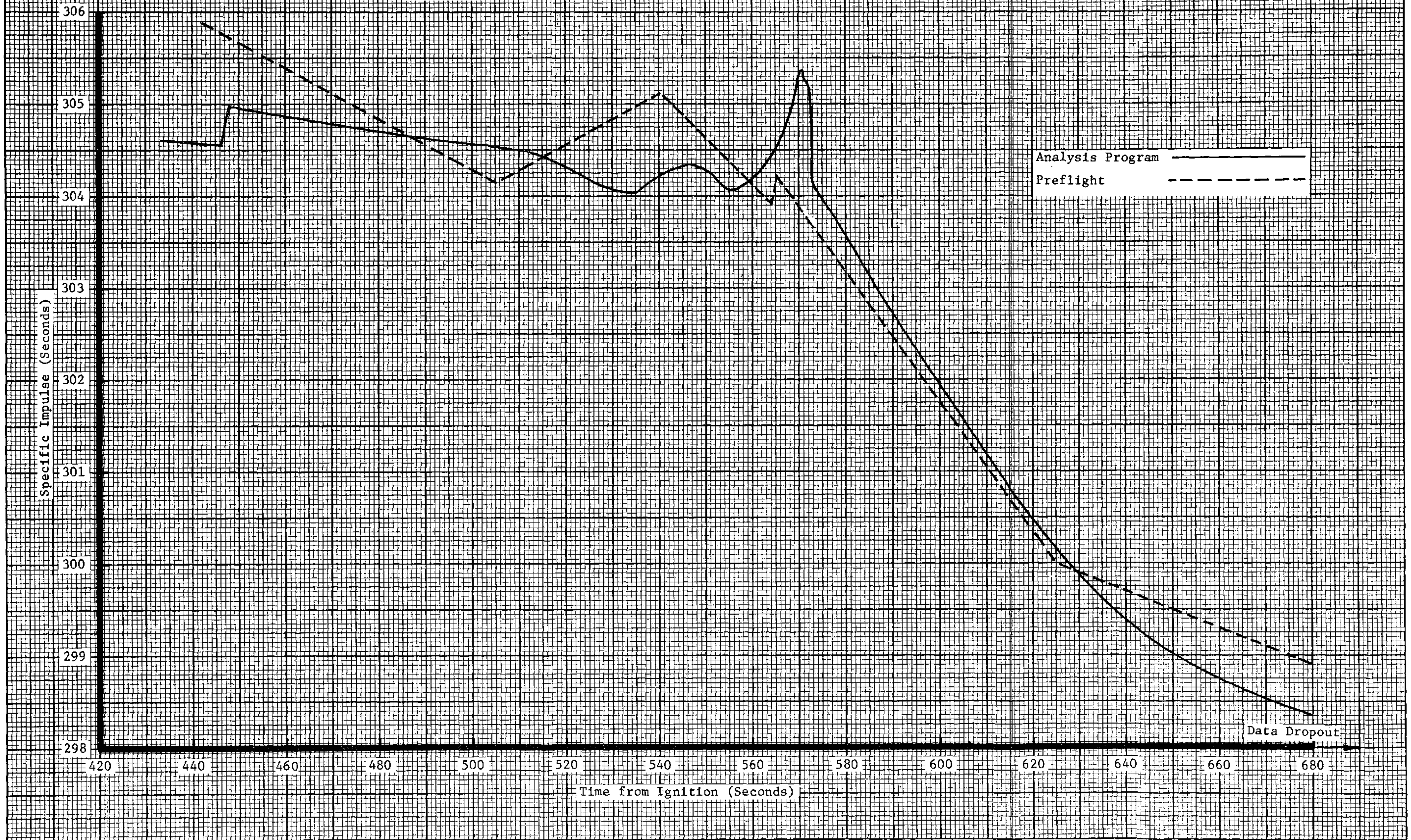


Figure 14. Comparison of Preflight Predicted and Analysis Program Simulated Engine Specific Impulse (Throttled Region)



FOLDOUT FRAME 1

FOLDOUT FRAME 2

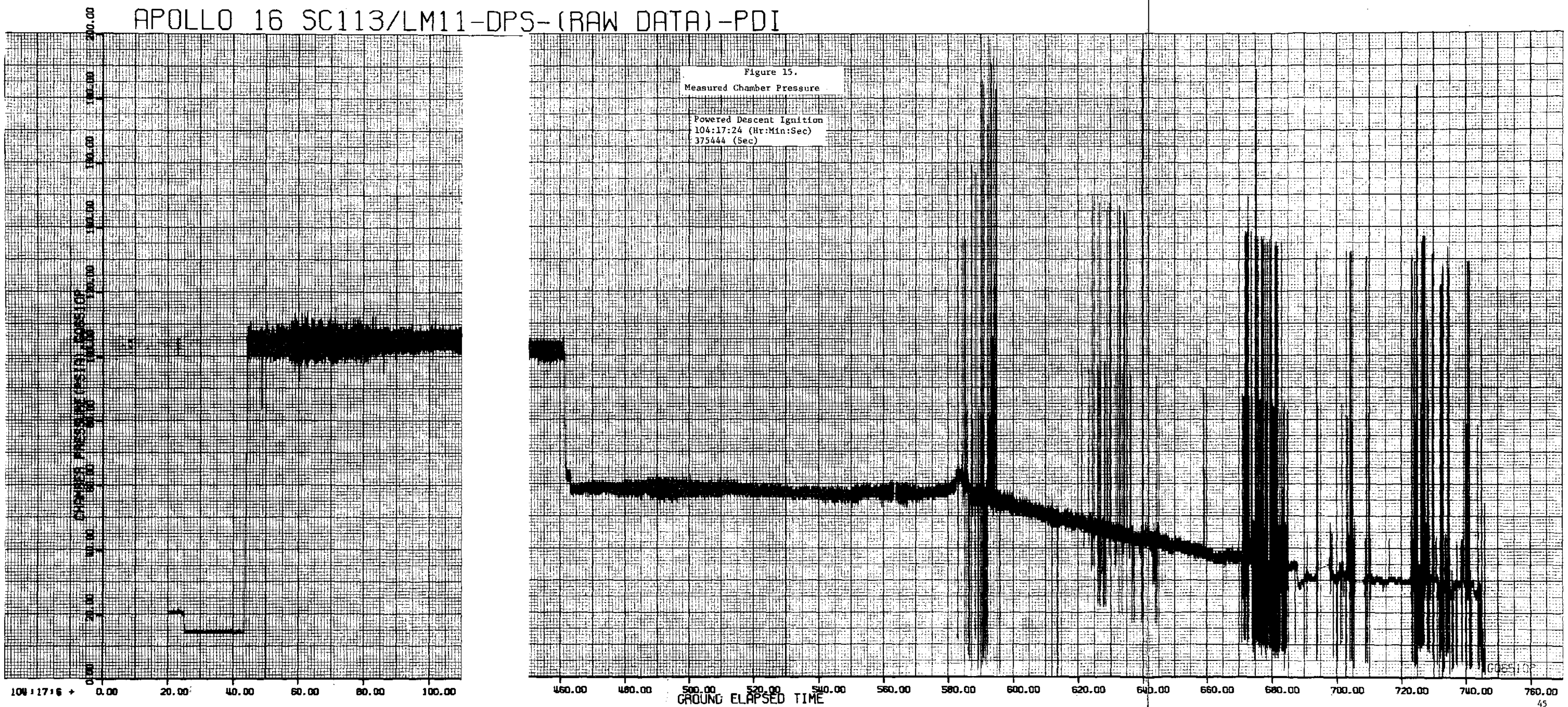
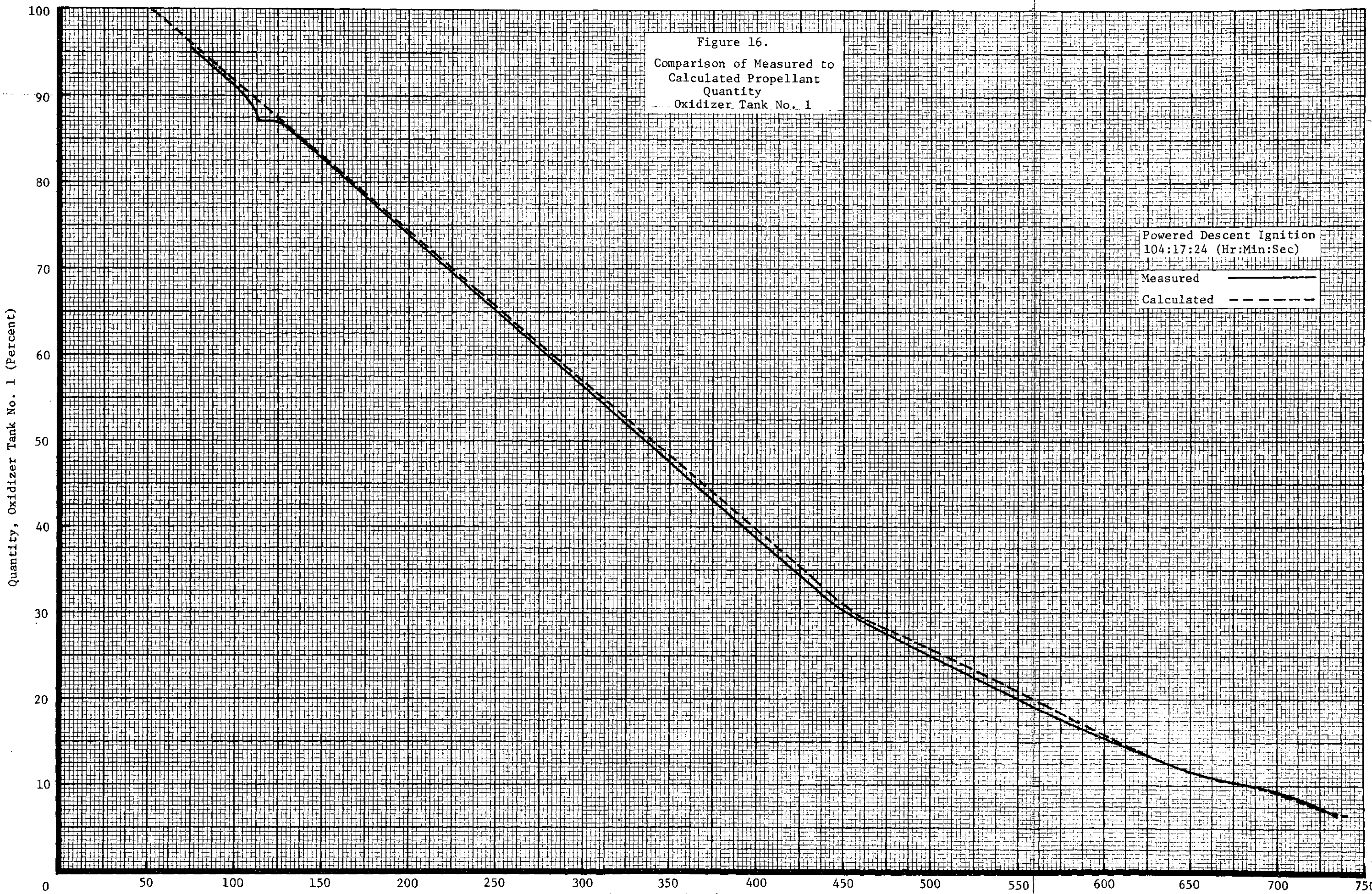


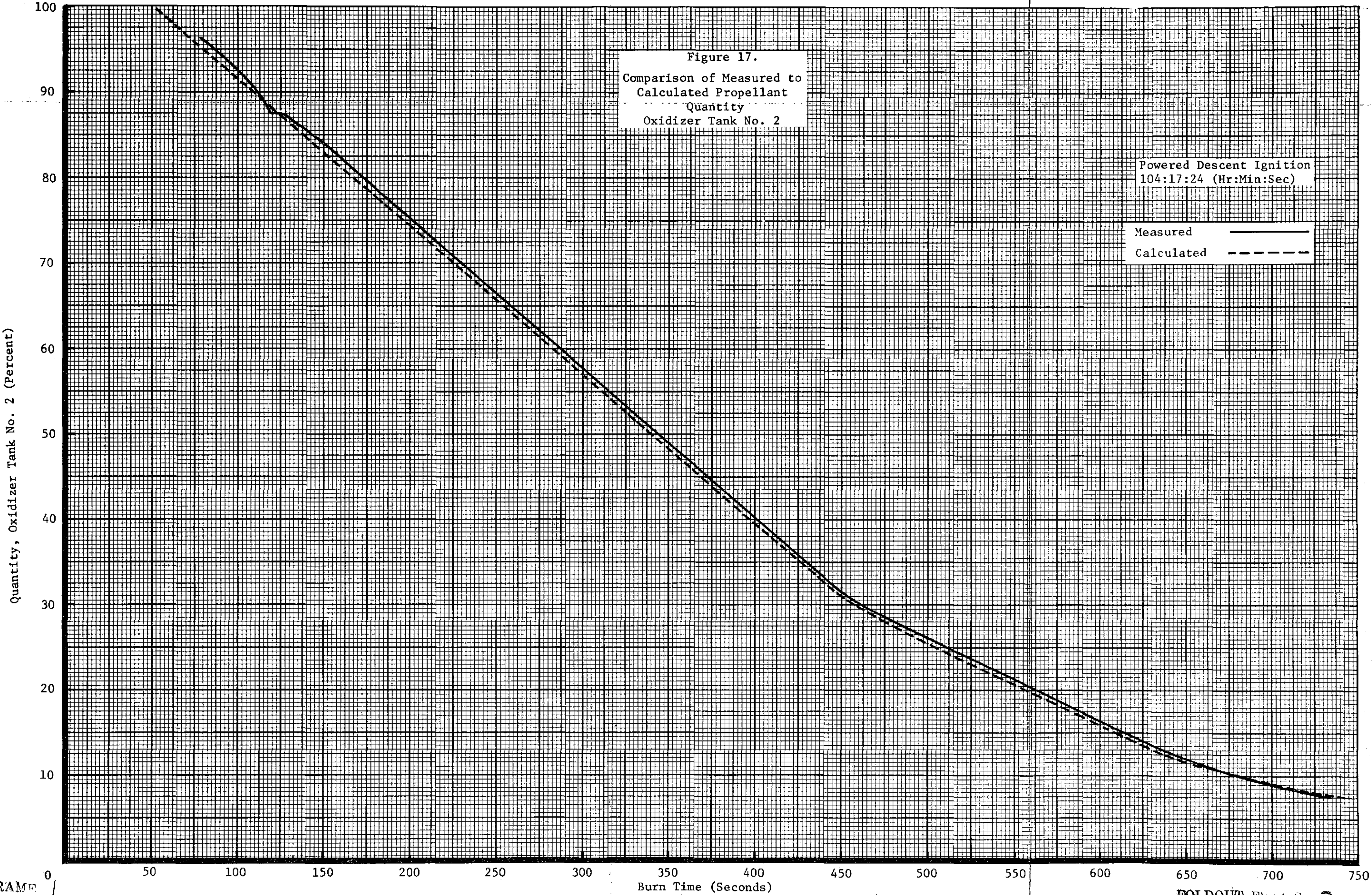
Figure 15.
Measured Chamber Pressure

FOLDOUT FRAME 1

FOLDOUT FRAME 2

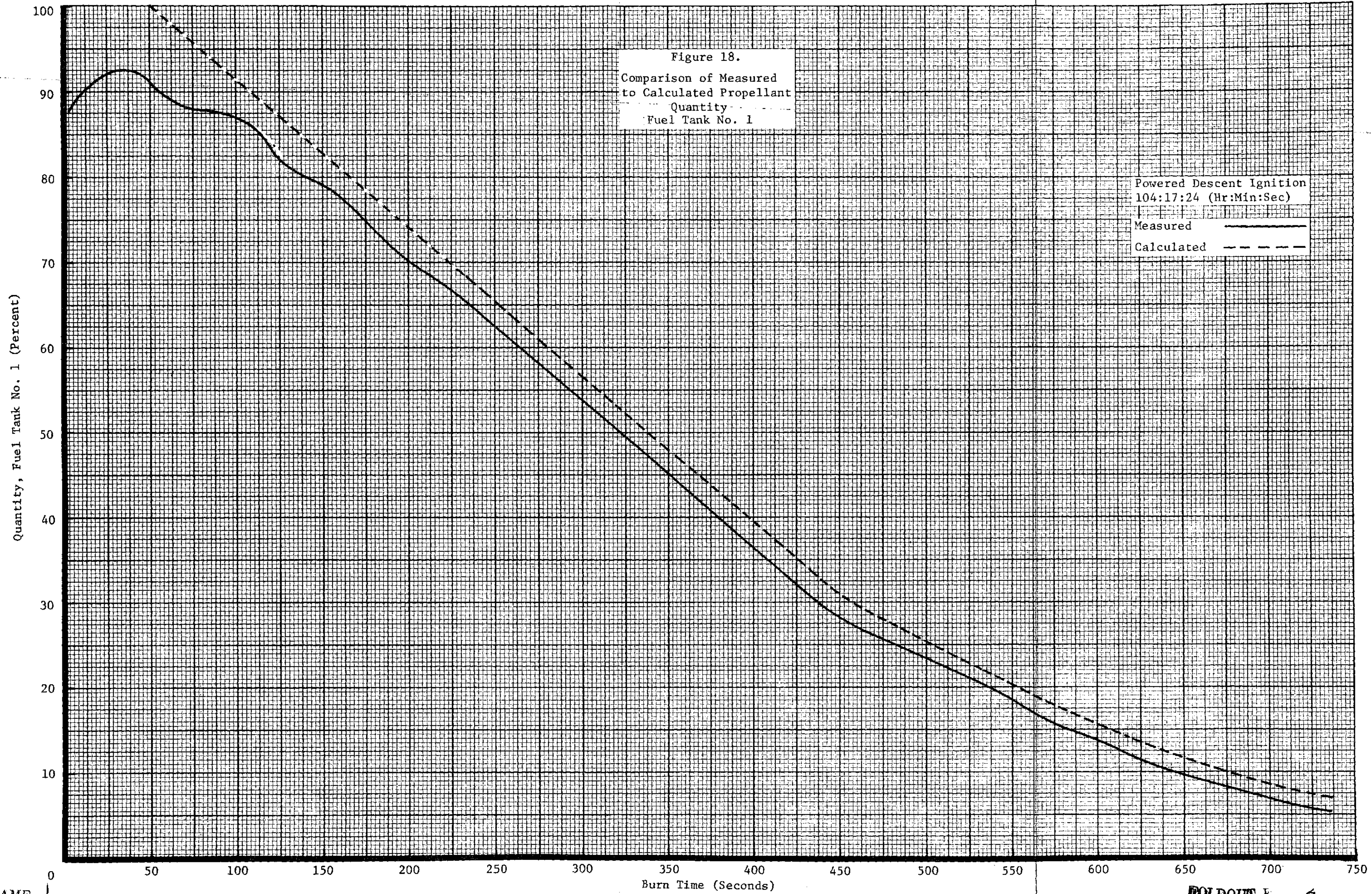


FOLDOUT FRAME 1
FOLDOUT FRAME 2



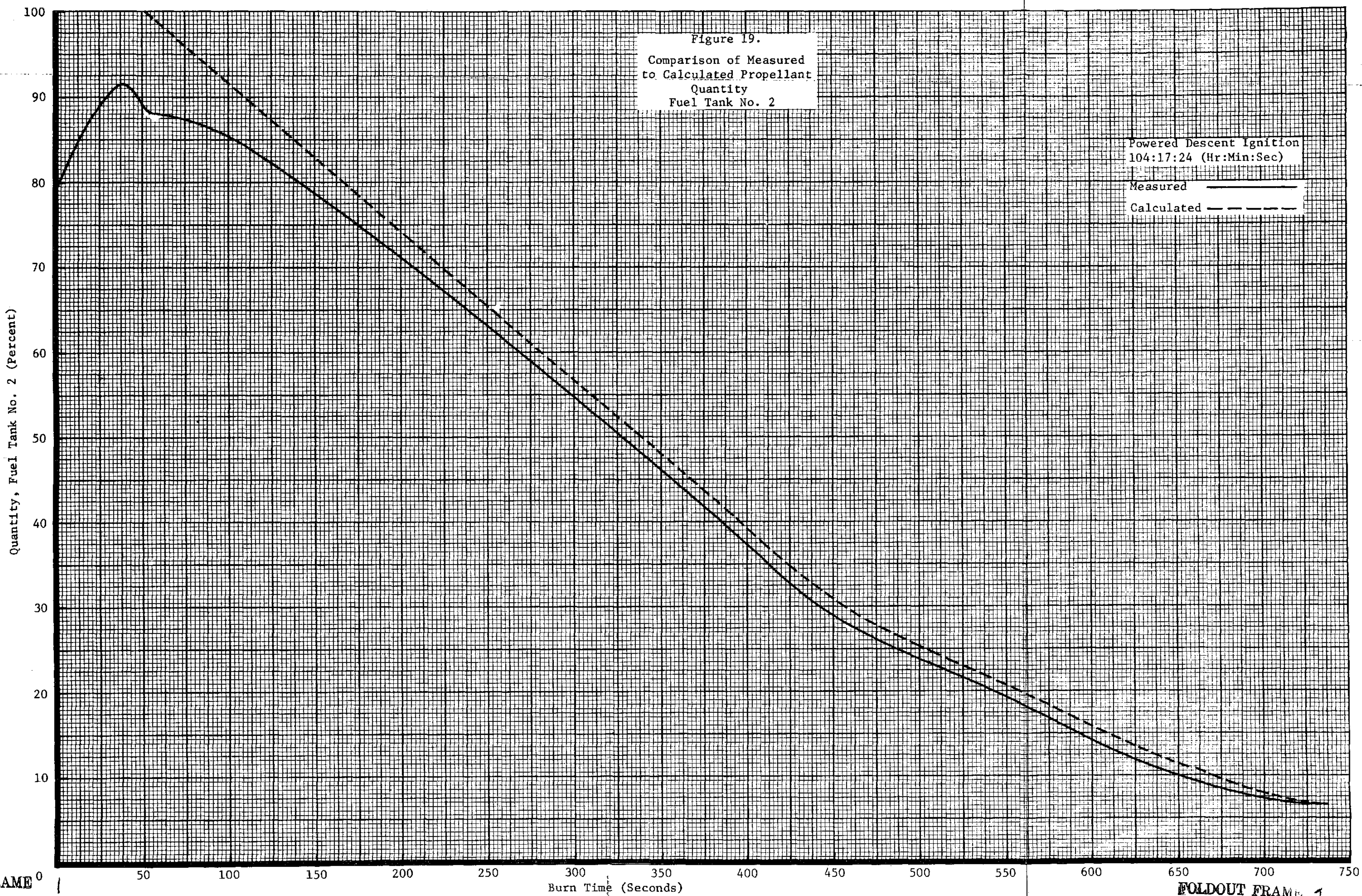
FOLDOUT FRAME 1

FOLDOUT FRAME 2



FOLDOUT FRAME

FOLDOUT FRAME 2



FOLDOUT FRAME

FOLDOUT FRAME 1

FIGURE 20

APOLLO 16 PREFLIGHT TO FLIGHT DATA COMPARISON, SHe SYSTEM

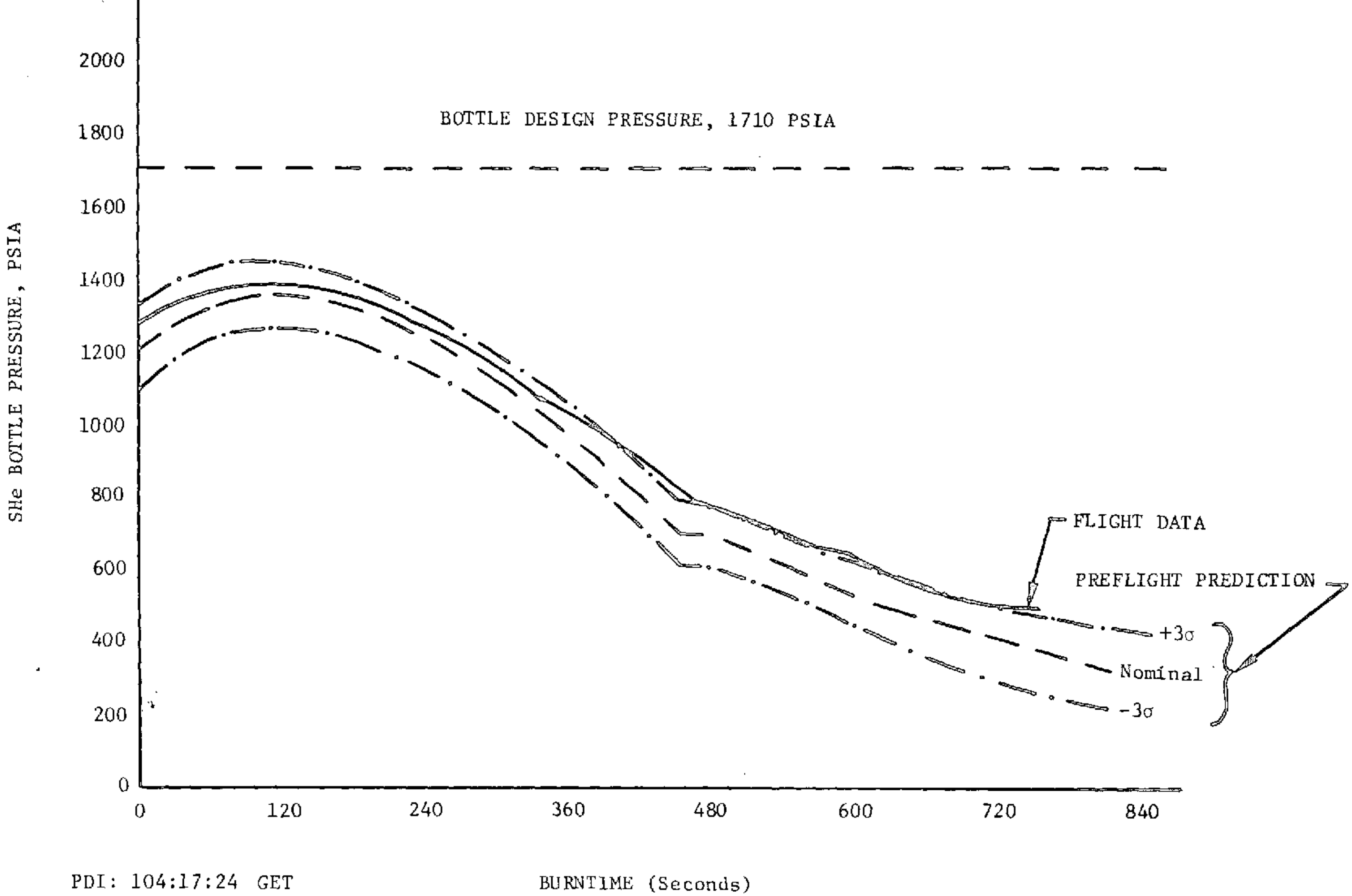


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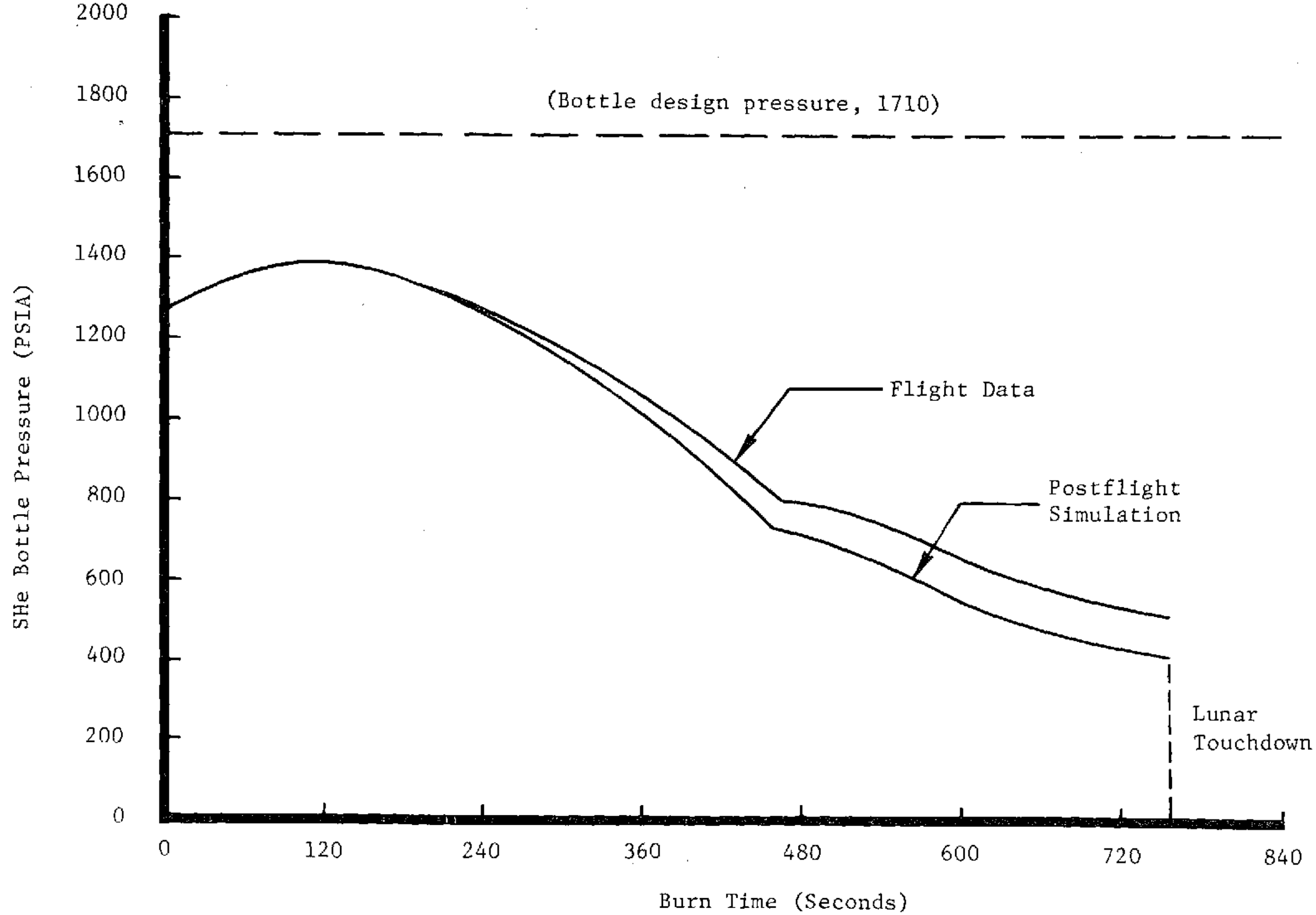
Figure 21.

Apollo 16 Postflight Simulation to Flight Data Comparison, SHe System



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