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APOLLO 12 MISSION REPORT

SERVICE PROPULSION SYSTEM FINAL FLIGHT EVALUATION

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MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
DECEMBER 1971

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SERVICE PROPULSION SYSTEM
FINAL FLIGHT EVALUATION

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
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SERVICE 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 Service Propulsion System (SPS) performance during the Apollo 12 Mission. This report is a supplement to the Apollo 12 Mission Report. The primary objective of the analysis was to determine the steady-state performance of the SPS under the environmental conditions of actual space flight.

This report covers the additional analyses performed following the compilation of Reference 1. The following items are the major additions to the results reported in Reference 1:

1) The steady-state performance as determined from analysis of the second and sixth burns is presented.
2) The analysis techniques, problems and assumptions are discussed.
3) The flight analysis results are compared to the preflight predicted performance.
4) The propellant utilization and gaging system operation is evaluated in greater detail.

2.0 SUMMARY

CSM 108 SPS performance for the Apollo 12 Mission was evaluated and found to be satisfactory. The SPS mission duty cycle consisted of six firings for a total duration of 546.11 seconds.

SPS steady-state performance was determined primarily from the analyses of the second (LOI-1) and sixth (TEI) burns. It was determined from these analyses that the engine fuel resistance was approximately 6.6% less than its acceptance test value. Similar, though smaller, reductions in engine fuel resistance were determined from the analyses of Apollo 9 and 10 SPS performance (References 4 and 5) and a reduction of approximately the same magnitude (6.9%) was determined by the Apollo 11 postflight reconstruction. As with previous postflight analyses, it was not possible to determine from the available flight data the exact reason for the resistance variation.

Apollo 12 was the first flight for which predicted SPS performance incorporated a mixture ratio bias in order to more closely predict the decreased mixture ratio observed on recent flights. The propellant mixture ratio determined by the Apollo 12 SPS postflight analysis was essentially as predicted.

Biases in both measured oxidizer and fuel interface pressure data were defined during the postflight analysis. The oxidizer and fuel interface pressure biases were biased approximately -4 and -2.6 psi, respectively. Biases of similar magnitudes were observed from the postflight analysis of Apollo 9, 10 and 11 (References 4, 5, and 6). There is, therefore, appreciable evidence that the flight interface pressure measurements are systematically in error.

Average standard inlet condition engine performance values for the two burns analyzed are as follows: thrust - 20706 pounds; specific impulse - 315.3 seconds; and propellant mixture ratio - 1.532 units. These values are 1% greater, 0.3% greater and 1% less, respectively, than corresponding values computed from the preflight engine model. Individual standard inlet condition performance for the two burns showed good agreement with differences of only 13 pounds, 0.2 seconds and 0 units for thrust, specific impulse and mixture ratio, respectively.

Operation of the Propellant Utilization and Gaging System (PUGS) was satisfactory throughout the mission. The PUGS mode selection switch was set in the normal position for all SPS burns, therefore, only the primary system data were available. The propellant utilization (PU) valve was in the normal position for the first 7 seconds of the first burn, at which time it was moved to the increase position, where it remained for all subsequent burns. An oxidizer gaging system error was noted during propellant loading and resulted in the oxidizer sump tank gage reading correctly with the propellant level at the top of the standpipe and 0.5% high at a propellant level of 0.7%. From 0.7% to empty, the error reduced to zero. An exact cause for the error has not been determined but a strong possibility exists that the PUGS control unit circuitry was at fault. With the exception of the gaging system error, all data indicates that the PUGS operated properly and that the use of the PU valve significantly increased the propellant available for ΔV maneuvers.

3. INTRODUCTION

The primary purpose of the Apollo 12 Mission was to investigate the lunar surface environment, to emplace ALSEP I (Apollo Lunar Surface Experiments Package), to obtain lunar material samples and to enhance the capability for manned lunar exploration. Apollo 12 was the twelfth in the series of flights using Apollo hardware and was the second manned lunar landing mission.

Launch from Kennedy Space Center (KSC) occurred at 11:22 A.M. Eastern Standard Time (EST) on 14 November 1969. The launch phase was normal with the exception of an interruption in spacecraft and launch vehicle electrical power shortly after liftoff. This power loss is attributed to lightning striking the space vehicle at 36.5 seconds and again at 52 seconds after launch. The spacecraft was inserted into a near circular parking orbit of approximately 103 nautical miles by the S-IVB stage of the AS-507 launch vehicle. The S-IVB stage was restarted and performed the Translunar Injection (TLI) maneuver at approximately 2-3/4 hours Ground Elapsed Time (GET). CSM-LM docking occurred at approximately 3-1/2 GET. Separation of the docked vehicles from the S-IVB was accomplished one hour later.

During the mission there were six SPS burns with a total duration of 546.11 seconds. The first SPS burn was a mid-course correction maneuver performed approximately 31 hours after liftoff. The second, and longest, SPS burn was the Lunar Orbit Insertion (LOI-1) conducted at approximately 83-1/2 hours GET. Approximately 4-1/2 hours later, the third SPS burn, the Lunar Orbit Circularization (LOI-2) Maneuver, was performed. Two lunar orbit plane change maneuvers constituted the fourth and fifth SPS burns. They were designated LOPC-1 and LOPC-2 and were conducted at approximately 119-3/4 hours GET and 159 hours GET, respectively. The sixth SPS burn was the Trans-

Earth Injection (TEI) Maneuver. TEI ignition occurred at approximately 172-1/2 hours GET. All SPS burns were started in the single bore mode with the second valve bank being opened 3 to 5 seconds later. The third, fourth, fifth and sixth SPS burns were preceded by plus-X Reaction Control System (RCS) burns to effect propellant settling. All SPS firings were conducted under automatic control. Actual ignition time, burn duration and velocity gain for each of the six SPS firings are contained in Table 1.

The Apollo 12 Mission utilized CSM-108 which was equipped with SPS Engine S/N 61 (Injector S/N 122). The engine configuration and expected performance characteristics (Reference 2) are contained in Table 2. There were no significant configuration differences between Apollo 11 and Apollo 12.

4. STEADY-STATE PERFORMANCE ANALYSIS

Analysis Technique

The major analysis effort for this report was concentrated on determining the steady-state performance of the SPS during the second and sixth burns. The remaining four burns were of insufficient duration to warrant detailed performance analysis. The performance analysis was accomplished with the aid of the Apollo Propulsion Analysis Program (PAP) which utilizes a minimum variance technique to "best" correlate the available flight and ground test data. The program embodies error models for the various flight and ground test data that are used as inputs, and by statistical and iterative methods arrives at estimations of the system performance history, propellant weights and spacecraft weight which "best" (minimum-variance sense) reconcile the available data.

Analysis Description

The steady-state performance during the second burn was derived from the PAP analysis of a 315-second segment of the burn. The segment analyzed began approximately 21 seconds following ignition (FS-1). The first 21 seconds of the burn were not included, in order to minimize any errors resulting from data filtering spans which include transient data, and because PUGS data near the start of the burn are erroneous. The time segment analyzed was terminated approximately 17 seconds prior to SPS shutdown (FS-2) to avoid shutdown transients. The burn segment included propellant crossover (storage tank depletion) which occurred about 240 seconds after ignition. The sixth burn steady-state performance was derived from the PAP analysis of a 100 second segment of the burn. The initial 20 seconds of the burn were excluded from the segment to avoid inclusion of data from the start transient. The segment was terminated approximately 10 seconds prior to engine cutoff in order to exclude shutdown transient data. The steady-state performance analyses of both burns utilized data from the flight measurements listed in Table 3.

The initial estimated spacecraft damp weight (total spacecraft minus SPS propellant) at ignition of the second burn was 55914 lbm. The initial estimated damp weight at ignition of the sixth burn was 21524 lbm. Both values were based on the postflight weight analysis given in Reference 3.

The initial estimates of the SPS propellants onboard at the beginning of the time segment analyzed for the second burn were extrapolated from the loaded propellant weights presented in Section 5. The initial propellant estimates for the time segment analyzed for the sixth burn were extrapolated from the computed propellants remaining at the end of the time segment analyzed for the second burn. All extrapolations of propellant masses used to establish the initial estimates for a given simulation were performed in an iterative manner using derived flowrates and propellant masses from preceding simulations to ensure that the derived propellant mass history was consistent between the two burns analyzed.

The SPS engine thrust chamber throat area was input to the program as a function of time from ignition for each burn. The assumed throat area time history used in the analysis is shown in Figure 1 and was based on the characterization presented in Reference 2.

The SPS propellant densities used in the analysis were calculated from propellant sample specific gravity data obtained from KSC, flight propellant temperature data, and flight interface pressures. The temperatures used were based on data from feed-system and engine feedline temperature measurements and were input to the program as functions of time. During steady-state operation, it was assumed that respective tank bulk temperatures and engine interface temperatures were equal for both oxidizer and fuel.

The PAP simulations were performed using an "interface pressure driven" SPS model. Simply stated, this model utilizes input oxidizer and fuel engine interface pressure values, as functions of time, for the starting points in computing the pressures and flowrates throughout the system. The input interface pressures used are generally the filtered data from the flight interface pressure measurements. The program is free to bias the input pressures, if so required, to achieve a minimum variance solution, but the version used (Linear Model 0) is essentially constrained to follow the shape of the input interface pressure profiles. The shapes of the interface pressure profiles, in turn, strongly influence the computed thrust shape, and, therefore, the calculated acceleration shape. The initial simulations of both burns, using the filtered interface pressure data, yielded minor computed acceleration shape errors. Analysis of the acceleration shape errors indicated that the filtered oxidizer and fuel interface pressure data were slightly in error. Shape errors in the filtered data are not unusual and are primarily the result of the PCM quantization of the raw data, which for the interface pressures is approximately 1.2 psi/PCM count. By utilizing the noise-in-the-state version (Linear Model 2) of the program, it was possible to derive corrections to the filtered interface pressure data which significantly improved the overall data match. The corrections, which were all less than .5 psi, were then input to the Linear Model 0 version of the program for subsequent simulations.

Analysis Results

The resulting values of the more significant SPS performance parameters, as determined in the analysis, are presented in Tables 4 and 5. Table 4 contains values for the second burn as computed in the PAP simulation. Values are presented for two time slices, which were selected to show performance before and after crossover. Table 5 contains the flight performance values for the sixth burn from the PAP analysis. The values shown are for two representative time slices following FS-1. In both tables the corresponding preflight predicted values for the same time slice are also shown. All performance values, both predicted and from the PAP analysis, are at the same PU valve position, increase, and should be directly comparable.

Figure 2 shows the calculated SPS specific impulse, propellant mixture ratio, and thrust, as functions of time, for the second burn and the sixth burn. For comparison the figures also contain the predicted performance. As shown, the specific impulse was between 315.4 and 315.8 seconds throughout both burns. Based on the values computed for the two burns analyzed, and the qualitative comparison of the data from all six burns, it is concluded that the SPS steady-state performance throughout the entire mission was satisfactory. The propellant mixture ratio was essentially as predicted. It should be noted that the predicted performance for this mission incorporated a mixture ratio bias in order to more closely predict the decreased mixture ratio observed on recent flights. A more detailed comparison of the flight performance to the predicted performance is contained in a following section.

The PAP analysis of the second burn determined that the best match to the available data required that the engine fuel hydraulic resistance be adjusted from its acceptance test value. The derived fuel resistance was 811.9 lbf-sec^2/lbm/ft^5, which is approximately 6.6% less than the value determined from engine acceptance test data. Similarly, the fuel resistance derived in the sixth burn analysis was 811.6 $lbf^2/lbm\text{-}ft^5$ which agrees well with the second burn results. The Apollo 9, 10, and 11 analyses derived fuel resistances 2.1%, 4.5%, and 6.9%, less, respectively, than those determined from engine acceptance tests.

During both burns the measured oxidizer and fuel interface pressure data (SP0931P and SP0930P) appeared to be biased. The measured oxidizer interface pressure averaged approximately 4.0 psi less than the simulated pressure during both burns and the measured fuel interface pressure during both burns averaged approximately 2.6 psi less than the simulated pressure. Similar biases have been observed in previous postflight analyses (References 4, 5, and 6), and there is appreciable evidence that the flight interface pressure measurements are systematically in error. The analysis also indicated a negative biases of approximately 3 psi in the measured oxidizer tank pressure data. The measured fuel tank pressure during the sixth burn (Figure 15) exhibited a decreasing trend with time which appeared erroneous when compared to predicted and to the measured fuel interface pressure. Therefore, the measured fuel tank pressure data was not used in the sixth burn simulation.

The analysis verified that the thrust chamber throat area characterization (Figure 1) was relatively accurate, in that no changes were required to achieve a satisfactory data match for either the second or sixth burn.

Both the second and sixth burn PAP analysis indicated that relatively minor reductions in the initial estimates of the spacecraft damp weight were required. The reductions were 53 lbm and 85 lbm for the second and sixth burns, respectively.

The second burn simulation indicated that a 72 lbm reduction in the estimated oxidizer mass onboard at the start of the burn segment analyzed and a 40 lbm reduction in the corresponding estimated fuel mass onboard were required. The PAP simulation of the sixth burn required a 69 lbm increase in the estimated oxidizer mass onboard at the start of the burn segment analyzed and a 22 lbm increase in the corresponding estimated fuel mass onboard.

Early analysis results indicated an inconsistency in the amounts of propellants that were loaded and the amounts indicated by the tank gages. This discrepancy was most apparent after crossover and was therefore associated with the sump tank gages. Since the first burn was of relatively short duration, the propellant loads onboard at the beginning of the second burn should be known to almost the loading tolerances. Resolution of this problem was accomplished by considering both scale factors and biases on the sump tank gages. The best overall solution was determined to be with the use of scale factors of 0.989 and 0.983, respectively, for the oxidizer and fuel sump tank gages. These scale factors are in addition to the scale factors determined from ground test data that are discussed in the PUGS section of this report.

Shown in Figures 3 through 20 are the PAP output plots which present the residuals (differences between the filtered flight data and the program-calculated values) and filtered flight data for the segments of the second and sixth burns analyzed. The figures appear in the following order: vehicle thrust acceleration, oxidizer tank pressure, fuel tank pressure, oxidizer interface pressure, fuel interface pressure, oxidizer sump tank quantity, fuel sump tank quantity, oxidizer and fuel storage tank quantities (second burn only), and chamber pressure for the second and sixth burn, respectively.

The values for slopes and intercepts seen in the upper right hand corner of these graphs represent the slopes and intercept on the ordinate of a linear fit of the residual data. It is readily seen that the closer these numbers are to zero, the better the match.

A strong indication of the validity of the PAP simulation can be obtained by comparing the thrust acceleration calculated in the simulation to that derived from the Apollo Command Module Computer (CMC) ΔV data transmitted via measurement CG0001V. This comparison is easily made in terms of the previously mentioned residual slope and intercept data. Figures 3 and 13 show the thrust acceleration during the portions of the burns analyzed, as derived from the CMC data, and the residual between the data and program calculated values. The residual time histories have essentially zero means and little, if any, discernible trend. This indicates that the simulations especially in terms of the computed specific impulse, are relatively valid, although other factors must also be considered in critiquing the simulations.

As observed on previous flights, the measured chamber pressure drifted with burn time during both burns, presumably because of thermal effects on the transducer. Although this drift has been partially modeled from knowledge obtained from past SPS flight analysis results, the existing drift model is, at best, approximate and not sufficient for detailed performance analyses where chamber pressure errors of less than 0.5 psi are significant. Because of the questionable nature of the chamber pressure data, this measurement was considered essentially useless for the detailed analysis, and was therefore not used in the simulations. The residuals plots, Figures 12 and 20 for the chamber pressure during the second and sixth burns are included for information only.. Because the chamber pressure could not be utilized, the ability of PAP to distinguish tank and interface pressure measurement errors from errors in the preflight engine model (engine resistances, thrust chamber characteristic velocity, and specific impulse) was somewhat diminished.

Several of the residual plots for the second burn show discontinuities at approximately 210 seconds from the beginning of the burn segment analyzed. These discontinuities are the result of the transients associated with propellant crossover and are not considered significant errors in the match.

Comparison with Preflight Performance Prediction

Prior to the Apollo 11 Mission, the expected performance of the SPS was presented in Reference 2. This performance prediction was for the integrated propellant feed/engine system and, wherever possible, utilized data and characteristics for the specific SPS hardware on this flight.

The predicted steady-state thrust, propellant mixture ratio, and specific impulse are shown in Figure 2 for the second and sixth burns, respectively. Also shown, for comparison, are the corresponding values for the flight as determined from the steady-state analysis. The predicted performance data is directly comparable to the flight reconstructed performance history since the PU valve logic used in the prediction resulted in maintaining the PU valve in the primary-increase position throughout the mission simulation. Flight data show that the PU valve was in the primary-increase position for all burns except the first seven seconds of the first burn.

The preflight performance prediction was biased by adjusting engine resistances to decrease the predicted mixture ratio by 2.5% at standard inlet conditions. Both the fuel and oxidizer resistances were adjusted from their acceptance values (see Table 2) in order to obtain the 2.5% mixture ratio reduction with no thrust changes. Additionally, improvements were made to propellant tank pressurization model which resulted in decreasing the predicted mixture ratio by an additional 1%. The mean difference between the actual and predicted mixture ratios for the Apollo 9, 10, and 11 Missions was -3.2%. As shown in Figure 2, the flight reconstructed mixture ratio compares quite closely with the predicted mixture ratio generated using the above technique; however, the reconstructed thrust throughout the flight was significantly greater than predicted. The reconstructed specific impulse was greater than predicted (approximately 1 second) but within the three sigma dispersions associated with the predicted specific impulse.

As noted above, the reconstructed thrust was significantly greater than predicted and was greater than would be expected for the flight propellant pressures and temperatures, as evidenced by the greater than predicted standard inlet conditions thrust (see Engine Performance at Standard Inlet Conditions). In order to account for the decreased mixture ratio and increased thrust, the fuel resistance was reduced by 6.6% from its acceptance test value, with the oxidizer resistance left equal to its acceptance test value. These results are similar to the Apollo 9, 10, and 11 postflight analysis results (References 4, 5, and 6) and substantiate the conclusion in the Apollo 11 postflight analysis report that biasing the fuel resistance will significantly improve the prediction model.

Engine Performance at Standard Inlet Conditions

The expected flight performance of the SPS engine was based on data obtained during the engine and injector acceptance tests. In order to provide a common basis for comparing engine performance, the acceptance test performance is adjusted to standard inlet conditions. This allows actual engine performance variations to be separated from performance variations which are induced by feed-system, pressurization system, and propellant temperature variations.

Based on the steady-state analysis of the second burn, the standard inlet conditions thrust, specific impulse and propellant mixture ratio were 20699 pounds, 315.2 seconds and 1.532, respectively. These values are 1% greater, 0.3% greater and 1% less, respectively, than the corresponding values computed from the engine model used in the preflight prediction.

The sixth burn analysis yielded standard inlet conditions thrust, specific impulse and propellant mixture ratio of 20712 pounds, 315.4 seconds and 1.532 units, respectively. These values are 1% greater, 0.4% greater and 1% less, respectively, than the corresponding values computed from the preflight engine model.

The standard inlet conditions performance values for the two burns agree well with each other, with the thrust, specific impulse, and propellant mixture values being only 13 pounds, 0.2 seconds, and 0.0 units different, respectively. The average standard inlet conditions thrust, specific impulse and propellant mixture ratio for the two burns were 20706 pounds, 315.3 seconds, and 1.532 units, respectively. These values are 1% greater, 0.3% greater, and 1% less, respectively, than the corresponding values computed from the preflight engine model.

As previously discussed, the engine resistances used in the preflight prediction were adjusted from their acceptance test values in an attempt to improve the mixture ratio prediction. If the average standard inlet conditions thrust, specific impulse and mixture ratio from the flight are compared to their corresponding values computed from an engine model based on the unadjusted acceptance test resistances the flight values are found to be 1% greater, 0.3% greater and 3.5% less, respectively, than the values from the unadjusted model.

The standard inlet conditions performance values reported herein were calculated for the following conditions.

STANDARD INLET CONDITIONS

Oxidizer interface pressure, psia	162
Fuel interface pressure, psia	169
Oxidizer interface temperature, °F	70
Fuel interface temperature, °F	70
Oxidizer density, lbm/ft^3	90.15
Fuel density, lbm/ft^3	56.31
Thrust acceleration, lbf/lbm	1.0
Throat area (initial value), in^2	121.700

Of primary concern in the flight analysis of all Block II engines is the verification of the present methods of extrapolating the specific impulse for the actual flight environment from data obtained during ground acceptance tests at sea level conditions. Since the SPS engine is not altitude tested during the acceptance tests, the expected specific impulse is calculated from the data obtained from the injector sea level acceptance tests using conversion factors determined from Arnold Engineering Developing Center (AEDC) simulated altitude qualification testing. As previously discussed, the average standard inlet conditions specific impulse determined from analyses of the second and sixth burns was 315.3 seconds. The predicted specific impulse at standard inlet conditions, as extrapolated from the ground test data was 314.3 seconds. The expected tolerance associated with the predicted standard inlet condition value of 314.3 seconds (Reference 2) was ±1.593 seconds (3-sigma). The flight value was well within this tolerance.

5. PUGS EVALUATION AND PROPELLANT LOADING

Propellant Loading

The oxidizer tanks were loaded to CM display readout of 100.9% at a tank pressure of 111 psia and an oxidizer temperature of 67.3^{O}F. The fuel tanks were loaded at 111 psia and 68.2^{O}F to a display readout of 101.0%. The SPS propellant loads calculated from these data, and propellant sample density data, are shown in Table 6. When the tank pressure was increased to 193 psia for oxidizer and 191 psia for fuel during the leak test, the CM displays read 100.59% for oxidizer and 100.45% for fuel. A decrease in the readings is expected when the tank pressures are increased because of tank stretch and propellant density changes. With the tank pressures at the higher values, the storage tank primary gage readings recorded through the ground equipment showed that the oxidizer sump tank level was at 56.8% which is quite close to the maximum gageable (57%) in the sump tank. The fuel sump tank level was at 56.6%. These readings indicate the oxidizer level in the sump tank would be close to the maximum gageable at flight pressures of 175-180 psia. As planned, the oxidizer storage tank primary gage was zero adjusted with an approximate -0.4% bias. This zero adjustment bias was incorporated for Apollo 10 and subs to prevent erroneous storage tank readings after crossover as experienced during the Apollo 9 Mission (Reference 3). The zero adjustment bias causes a small, but known, time varying error in the readings from the storage tank primary gage prior to crossover.

PUGS Operation in Flight

The propellant utilization gaging system (PUGS) operated satisfactorily throughout the mission. The PUGS mode selection switch was set in the normal position for all SPS burns, therefore, only the primary system data were available. The propellant utilization (PU) valve was in the normal position for the first 7 seconds of the first burn, at which time it was moved to the increase position, where it remained for all subsequent burns.

Figures 21 and 22 show the T/M PUGS data for all Apollo 12 SPS burns. Figure 23 shows the indicated propellant unbalance history; as computed from the T/M data. The indicated unbalance history should reflect the CM display unbalance history, within T/M accuracy. The propellant unbalance history shows a small initial increase reading at the beginning of the second SPS burn. The first burn was too short to yield truly meaningful gaging system data. A decreasing trend is noted during the second burn until crossover at which time an expected shift to increase occurred. The magnitude of the shift was approximately 80 pounds overall, from a decrease reading of 40 pounds to an increase reading of 40 pounds. As previously stated, the shift to an increase reading is expected and results from the elimination of two known errors, a 0.4% zero gage adjustment and the ungageable oxidizer above the top of the sump tank probe. After crossover, the indicated unbalance begins the increase slightly due primarily to an oxidizer sump tank gaging error. The gage error was noted during propellant loading and resulted in the gage reading correctly with the propellant level at the top of the standpipe and 0.5% high at a propellant level of 0.7%. From 0.7% to empty, the error reduced to zero. An exact cause for the error has not been determined, but a strong possibility exists that PUGS control unit circuitry was at fault since the sump tank probe itself was acceptance tested and operated linearly. This error also caused the indicated unbalance for the sixth burn to increase, however, in both this instance and the period of the second burn after crossover, the actual balance history (after correcting for the error) would show a decreasing trend. At the end of the sixth burn, the indicated unbalance shows 50 lbm increase.

By using the propellant loading data and the final T/M gaging data, the average mixture ratio with the PU valve in the increase position was computed to be 1.598, which agrees well with the predicted data. It should be noted that this is the first flight for which the SPS performance prediction incorporated the results of post postflight analyses the regarding mixture ratio bias.

All data indicates the PUGS functioned properly and that the use of the PU valve significantly increased the propellant available for ΔV maneuvers. The positioning of the PU valve on this flight was the result of experience and information gained from past flight analyses concerning system operation. Data from this flight substantiates these analyses.

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TABLE 1

SPS DUTY CYCLE

Maneuver	FS1 (1) (G.E.T.)	FS2 (1) (G.E.T.)	Burn Duration (Sec)
MCC-2	30:52:44.37	30:52:53.55	9.18
LOI-1	83:25:23.36	83:31:15.61	352.25
LOI-2	87:48:48.08	87:49:04.99	16.91
LOPC-1	119:47:13.23	119:47:31.46	18.23
LOPC-2	159:04:45.47	159:05:04.72	19.25
TEI	172:27:16.81	172:29:27.13	130.32
		Total	546.11

(1) Times are from Command Module Computer Downlink Data - CG0001V.

TABLE 2

PREDICTED CSM 108 SPS ENGINE AND FEED SYSTEM CHARACTERISTICS

Engine No.	61
Injector No.	122
Chamber No.	339
Initial Chamber Throat Area (in^2)	121.700

Engine and System Fluid Resistances ($lbf\text{-}sec^2/lbm\text{-}ft^5$)

	Acceptance Test	Predicted (1)
Fuel Engine Feedline	869.6	841.3
Oxidizer Engine Feedline	492.8	501.6
Fuel System Feedline	36.02	
Oxidizer System Feedline		
PU Valve in Pri-normal Position	97.72	
PU Valve in Pri-increase Position	49.45	
PU Valve in Pri-decrease Position	167.55	

Characteristic Equation for C*:

$$C^* = C^*_{S.C.} + 870.5\,(MR - 1.6) - 273.83(MR^2 - 2.56) - 0.31878(P_C - 99)$$
$$+ 12.953(TP - 70) - 0.07414(TP^2 - 4900) - 5.466(MR \cdot TP - 112)$$
$$+ 0.03119(MR \cdot TP^2 - 7840);\ \text{where } C^*_{S.C.}\ \text{(Engine No. 61)} = 6006\ \text{ft/sec}$$

Characteristic Equation for I_{SP}:

$$I_{SP} = I_{SP_{vac}} - 96.954(1.6 - MR) - 0.0487(99 - P_C) - 0.06276(70 - TP)$$
$$+ 30.409(2.56 - MR^2) + 0.0004483(4900 - TP^2);$$
$$\text{where } I_{SP_{vac}}\ \text{(Engine No. 61)} = 314.3\ \text{lbf-sec/lbm}$$

(1) To account for the decrease mixture ratio observed during past flights, predicted engine resistances were adjusted so as to result in a mixture ratio reduction of 2.5% at standard inlet conditions.

TABLE 3

FLIGHT DATA USED IN STEADY STATE ANALYSIS

Measurement Number	Description	Range	Sample Rate Samples/Sec
SP0930 P	Pressure, Engine Fuel Interface	0-300 psia	10
SP0931 P	Pressure, Engine Oxidizer Interface	0-300 psia	10
SP0661 P	Pressure, Engine Chamber	0-150 psia	100
SP0003 P	Pressure, Oxidizer Tanks	0-250 psia	10
SP0006 P	Pressure, Fuel Tanks	0-250 psia	10
SP0048 T	Temperature, Engine Fuel Feed Line	0-200^{o}F	1
SP0049 T	Temperature, Engine Oxidizer Feed Line	0-200^{o}F	1
SP0054 T	Temperature, 1 Oxidizer Distribution Line	0-200^{o}F	1
SP0057 T	Temperature, 1 Fuel Distribution Line	0-200^{o}F	1
SP0655 Q	Quantity, Oxidizer Tank 1 Primary - Total Auxiliary	0-50%	1
SP0656 Q	Quantity, Oxidizer Tank 2	0-60%	1
SP0657 Q	Quantity, Fuel Tank 1 Primary - Total Auxiliary	0-50%	1
SP0658 Q	Quantity, Fuel Tank 2	0-60%	1
CG0001 V	Computer Digital Data	40 Bits	1/2

TABLE 4

SERVICE PROPULSION SYSTEM STEADY-STATE PERFORMANCE
SECOND SPS BURN

PARAMETER	INSTRUMENTED					
	Before Crossover			After Crossover		
	FS-1 + 50 Sec.			FS-1 + 300 Sec.		
	Predicted	PAP	Meas.	Predicted	PAP	Meas.
PU Valve Position	Increase	Increase	Increase	Increase	Increase	Increase
Oxidizer Tank Pressure, psia	174	177	174	175	177	174
Fuel Tank Pressure, psia	175	175	175	175	176	176
Ox Interface Pressure, psia	167	169	164	170	168	169
Fuel Interface Pressure, psia	171	171	168	173	171	172
Engine Chamber Pressure, psia	102	104	102	104	105	105
	DERIVED					
Oxidizer Flowrate, lbm/sec	40.8	41.4	---	41.3	42.0	---
Fuel Flowrate, lbm/sec	25.7	26.0	---	25.8	26.2	---
Propellant Mixture Ratio	1.586	1.593	---	1.596	1.602	---
Vacuum Specific Impulse, sec	314.4	315.5	---	314.5	315.6	---
Vacuum Thrust, lbf	20920	21252	---	21120	21504	---

Notes:
(1) Predicted values from Reference 2.
(2) Calculated values from Propulsion Analysis Program
(3) Measured data are as recorded and are not corrected for biases and errors in text.

TABLE 5

SERVICE PROPULSION SYSTEM STEADY-STATE PERFORMANCE
SIXTH SPS BURN

PARAMETER	INSTRUMENTED					
	FS-1 + 50 Sec.			FS-1 + 100 Sec.		
	Predicted	PAP	Measured	Predicted	PAP	Measured
PU Valve Position	Increase	Increase	Increase	Increase	Increase	Increase
Oxidizer Tank Pressure, psia	176	177	172	175	177	173
Fuel Tank Pressure, psia	175	176	178	175	176	177
Oxidizer Interface Pressure, psia	171	172	168	170	172	168
Fuel Interface Pressure,psia	174	174	171	173	174	171
Engine Chamber Pressure,psia	104	105	103	103	105	104
	DERIVED					
Oxidizer Flowrate, lbm/sec	41.4	41.8	-	41.2	41.8	-
Fuel Flowrate, lbm/sec	25.8	26.0	-	25.8	26.0	-
Propellant Mixture Ratio	1.599	1.605	-	1.593	1.605	-
Vacuum Specific Impulse, sec	314.5	315.8	-	314.5	315.8	-
Vacuum Thrust, lbf	21200	21423	-	21100	21398	-

Notes:
(1) Predicted values from Reference 2
(2) Calculated values from Propulsion Analysis Program
(3) Measured data are as recorded and are not corrected for biases and errors dicussed in text.

TABLE 6

SPS PROPELLANT LOADING DATA

Propellant	Total Mass Loaded (lbm) Computed From Loading Data	Planned
Oxidizer	25089	25092
Fuel	15728	15704
TOTAL	40817	40796

FIGURE 1

SPS CHAMBER THROAT AREA

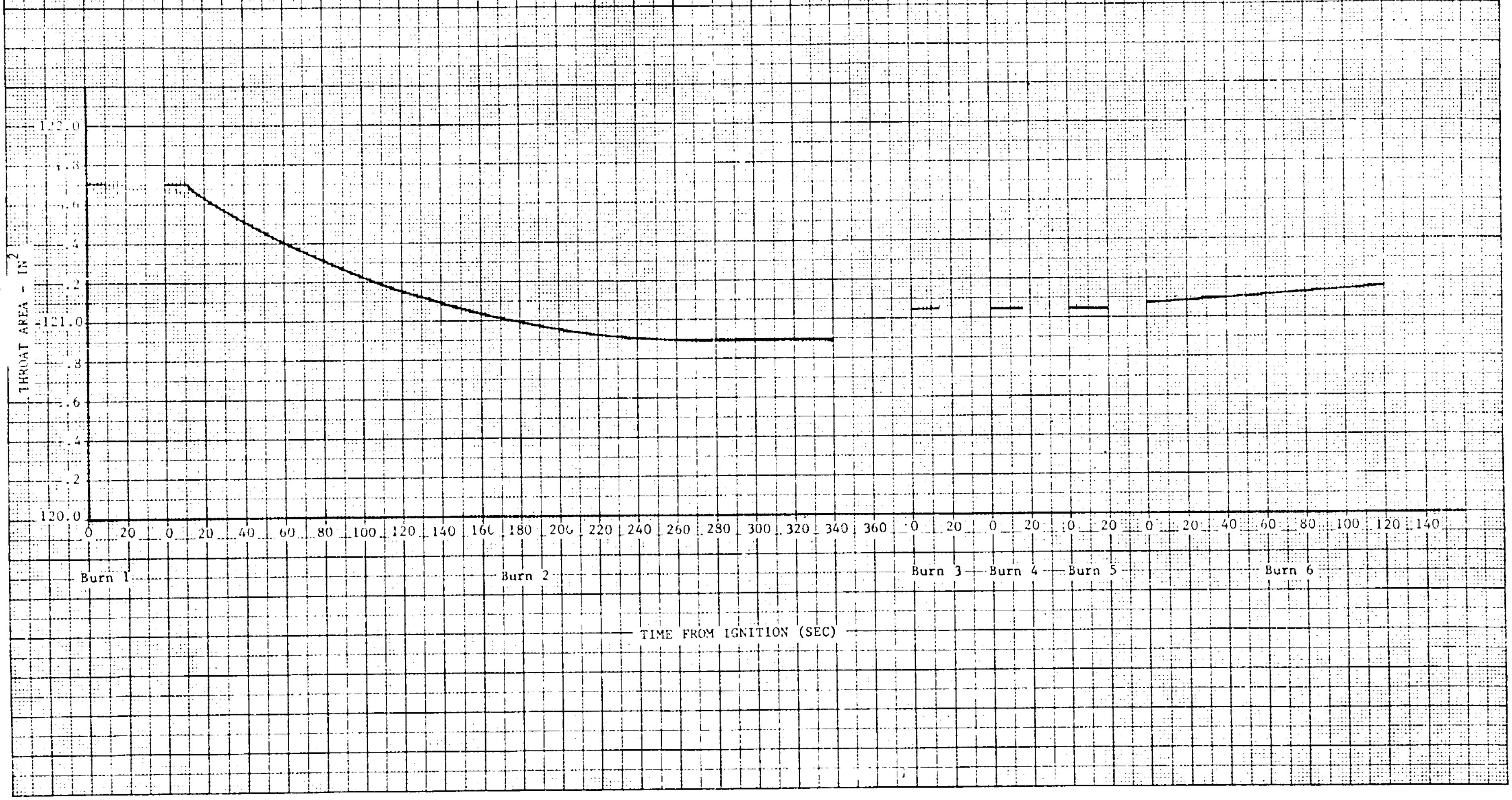
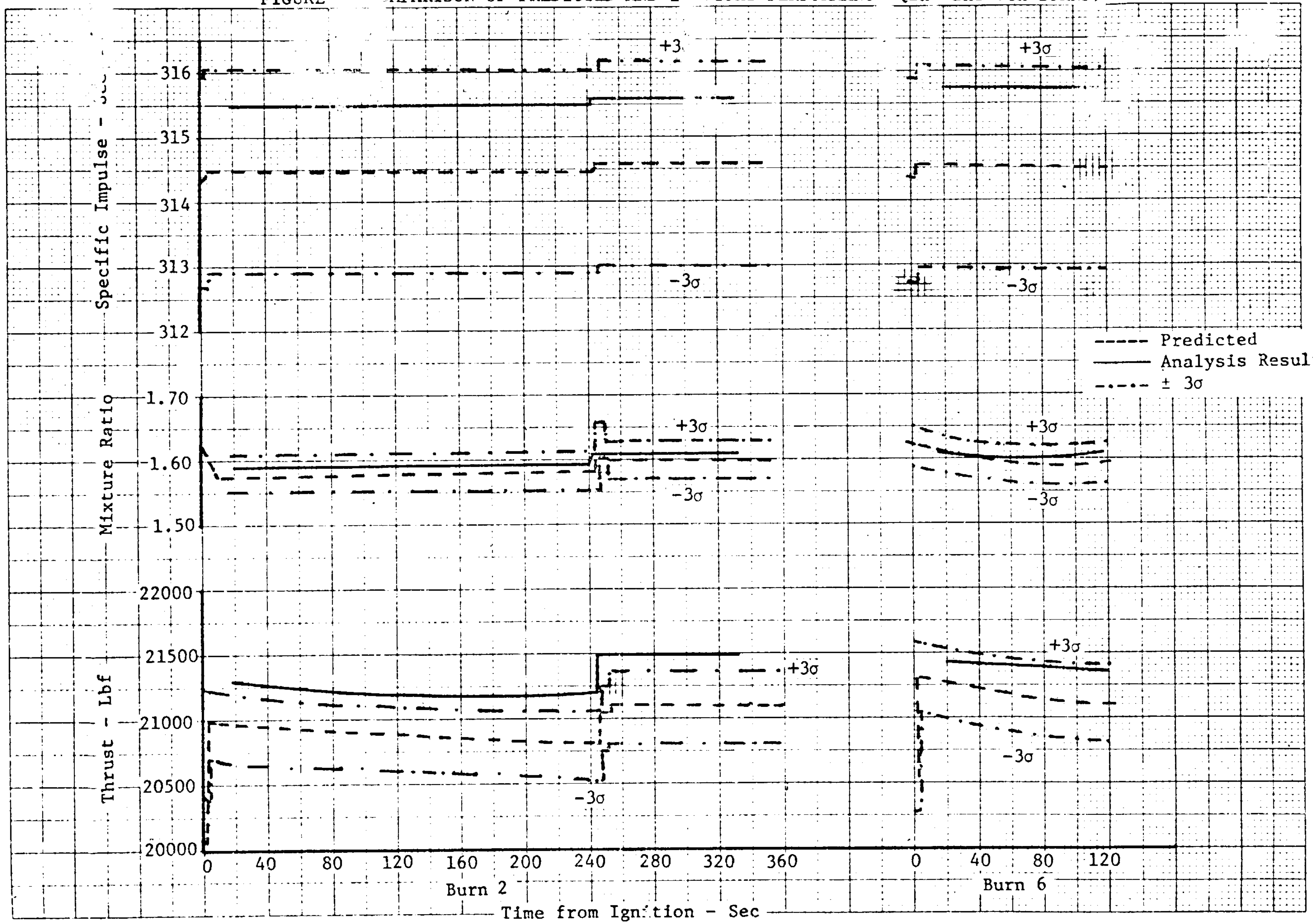


FIGURE)MPARISON OF PREDICTED AND I ⁼.IGHT PERFORMANC (2n. and 6th BURNS)



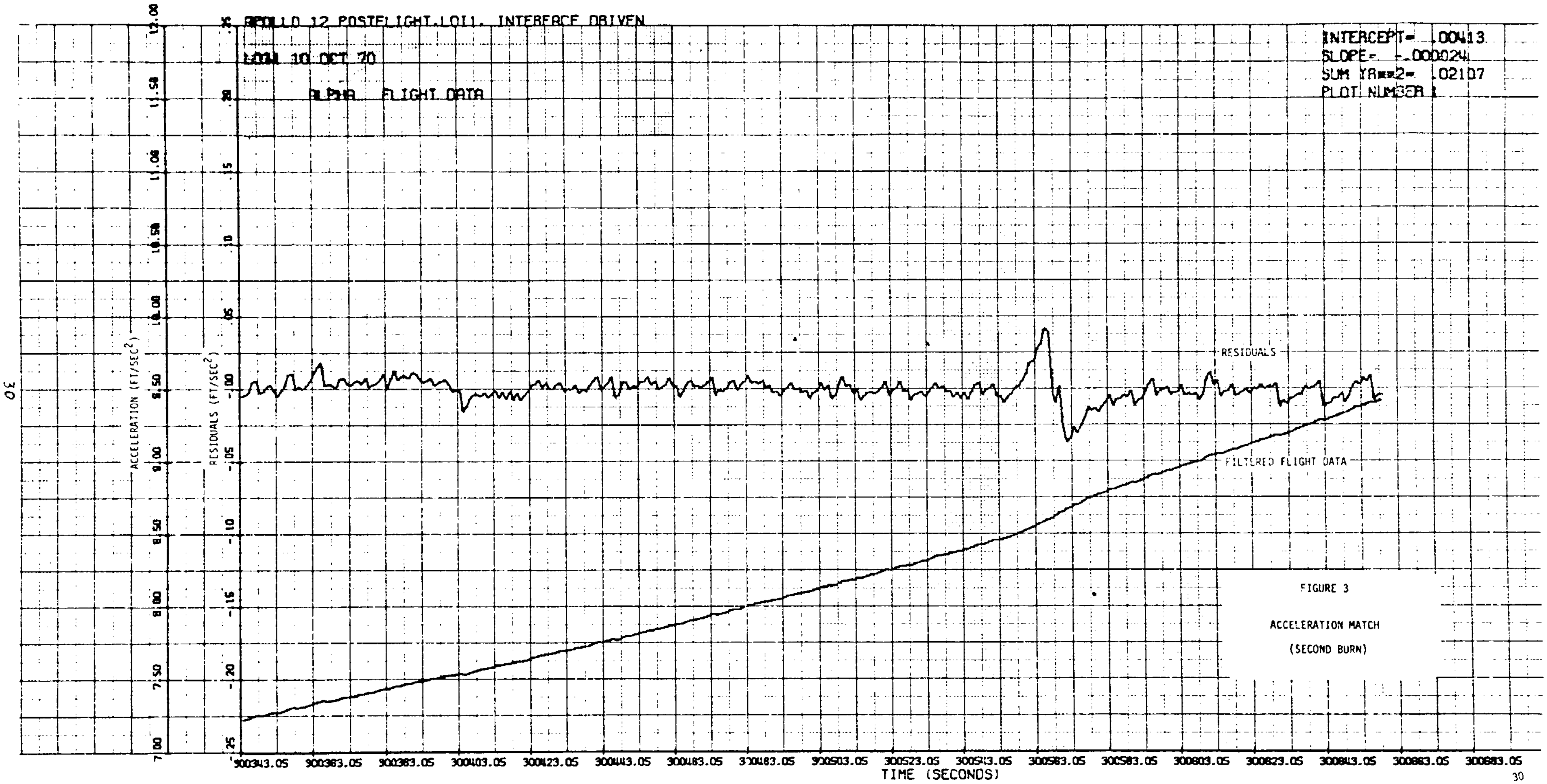


FIGURE 3

ACCELERATION MATCH

(SECOND BURN)

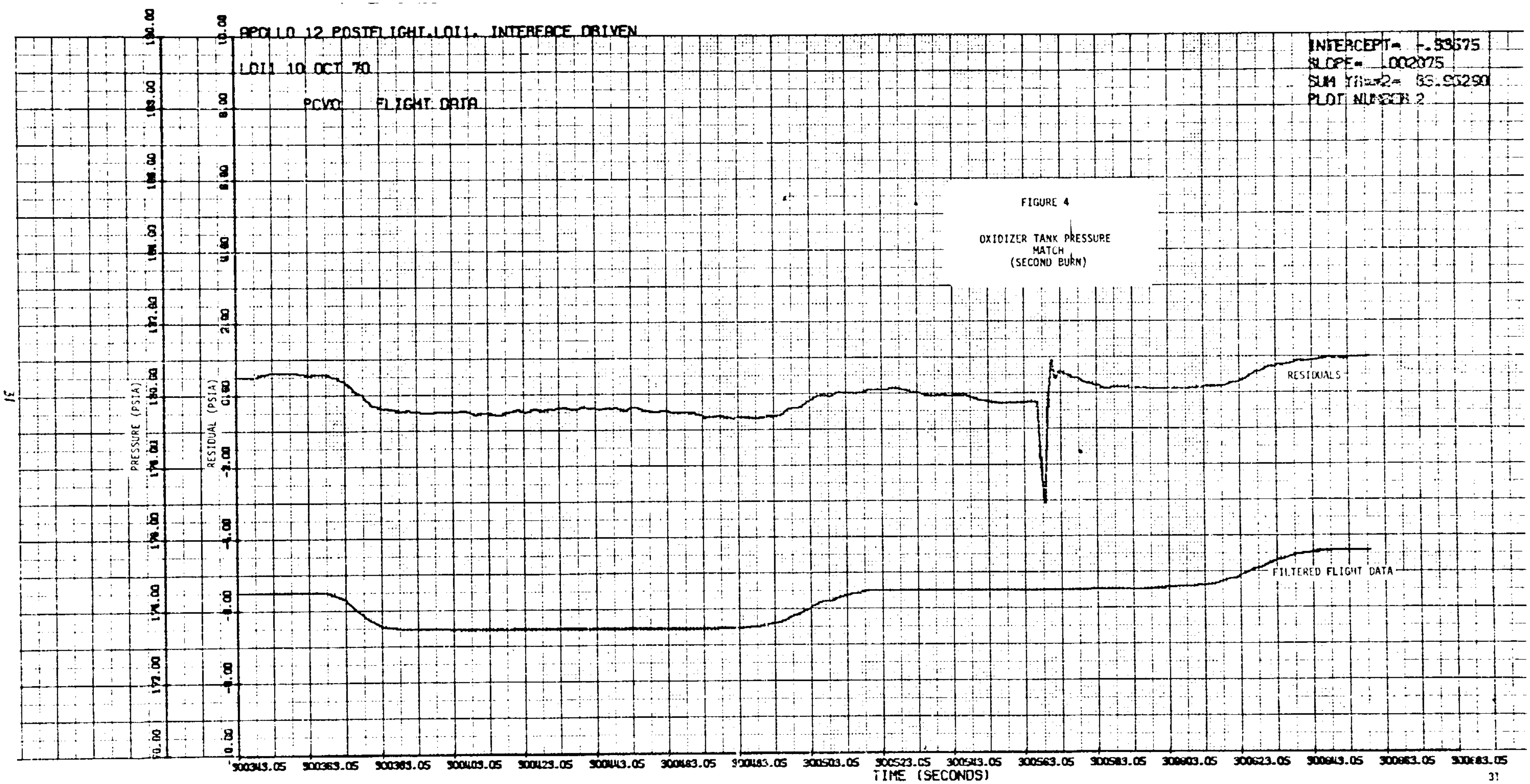


FIGURE 4

OXIDIZER TANK PRESSURE MATCH (SECOND BURN)

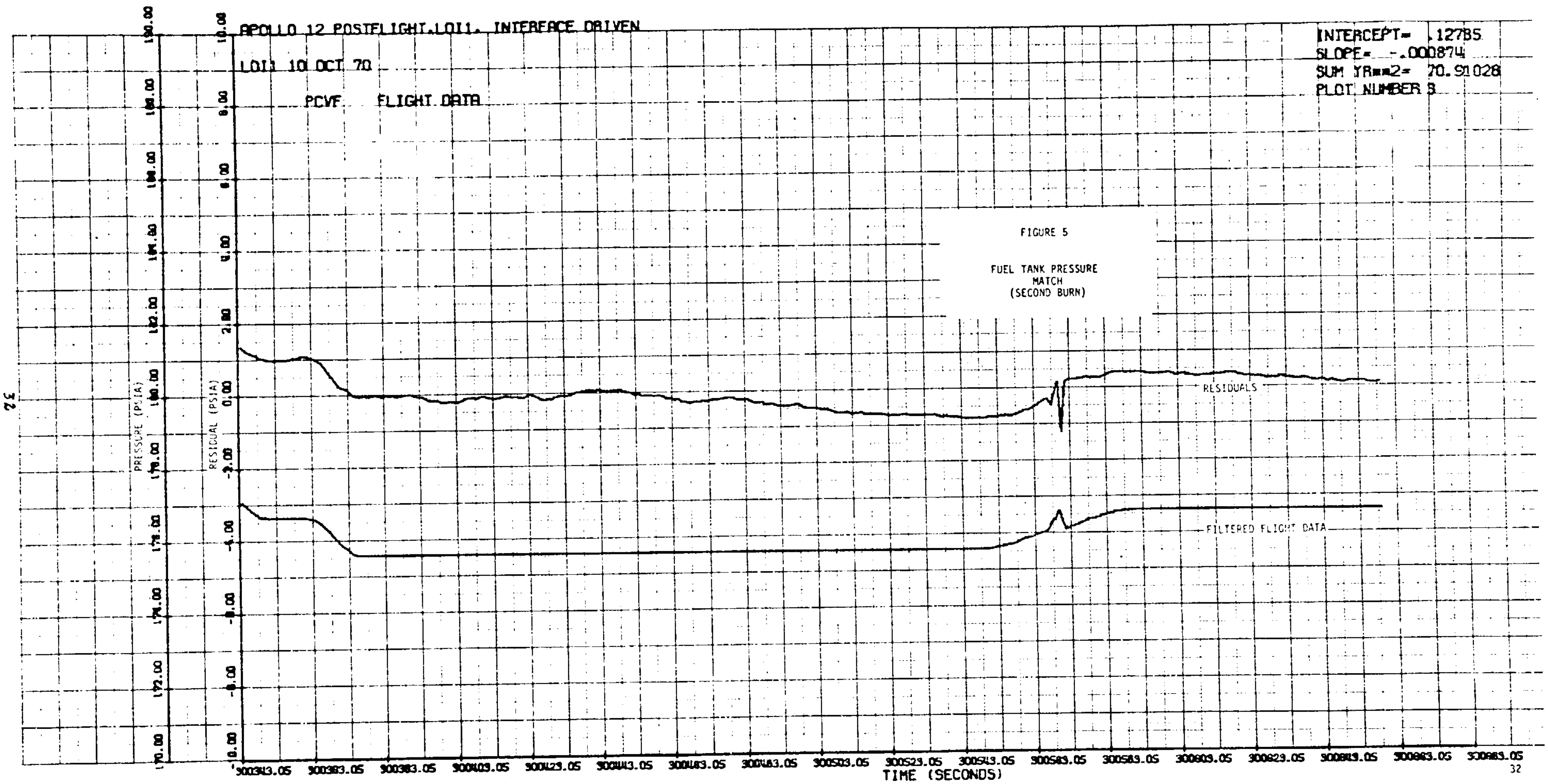


FIGURE 5

FUEL TANK PRESSURE MATCH (SECOND BURN)

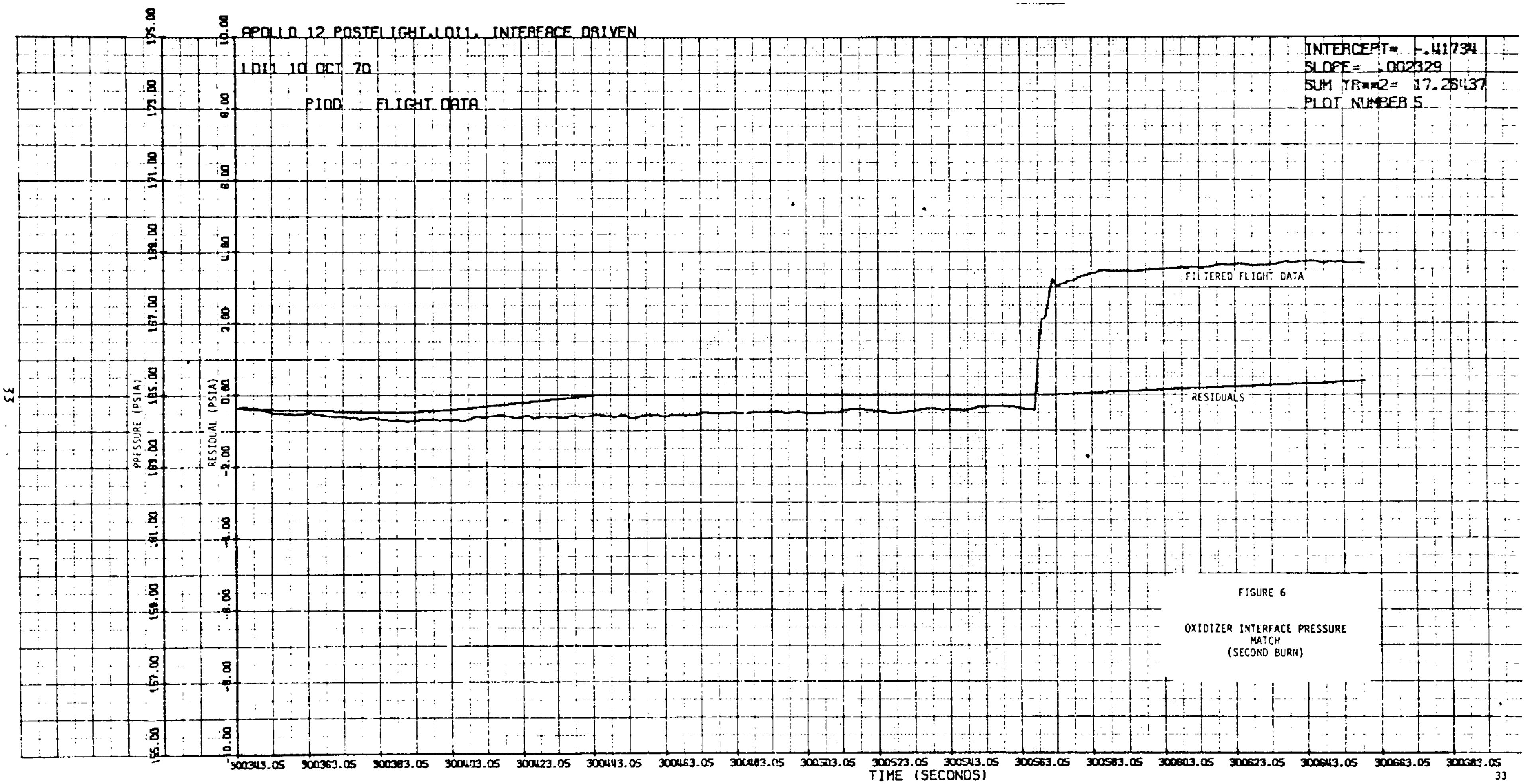


FIGURE 6

OXIDIZER INTERFACE PRESSURE MATCH (SECOND BURN)

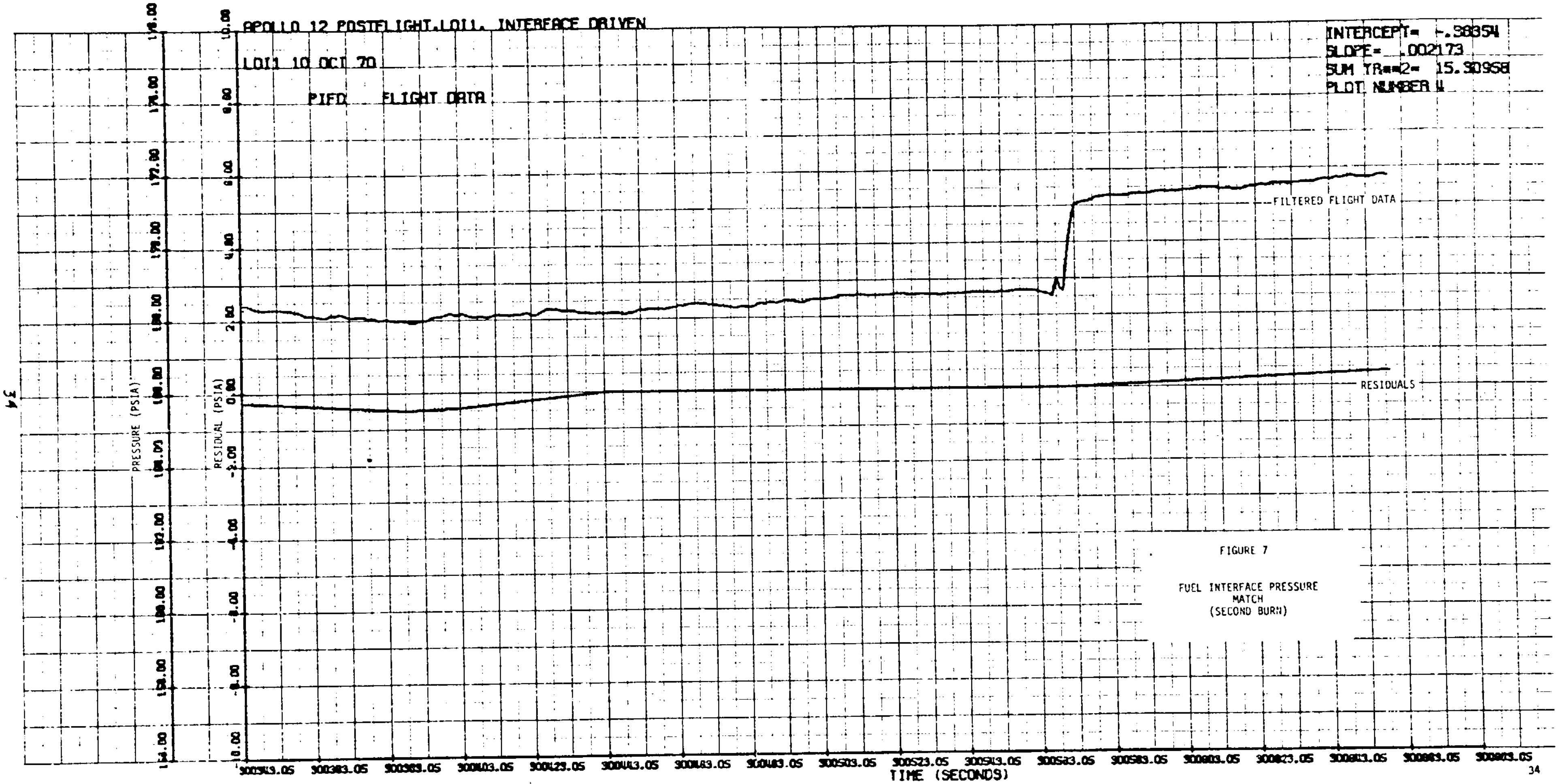


FIGURE 7

FUEL INTERFACE PRESSURE
MATCH
(SECOND BURN)

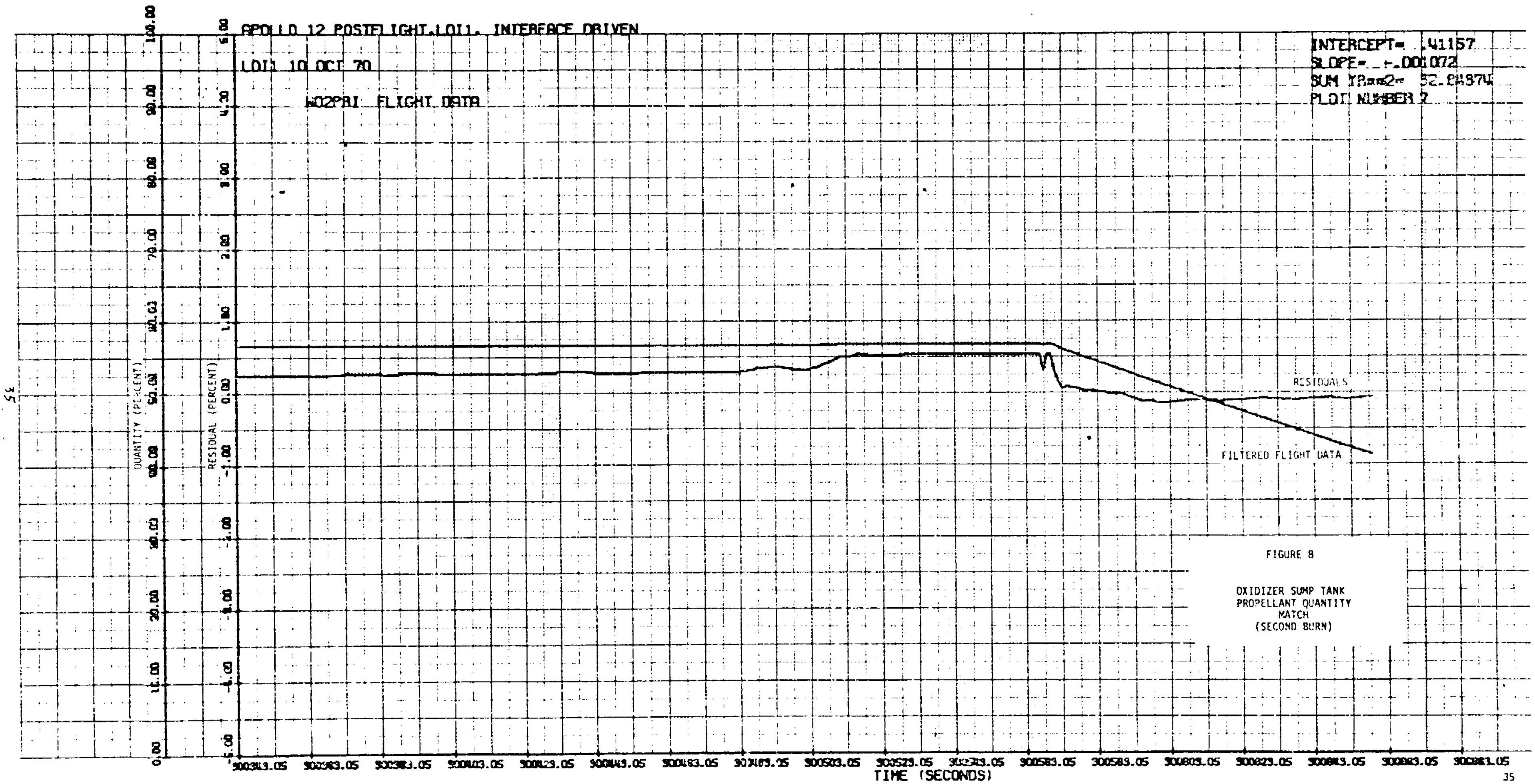


FIGURE 8

OXIDIZER SUMP TANK PROPELLANT QUANTITY MATCH (SECOND BURN)

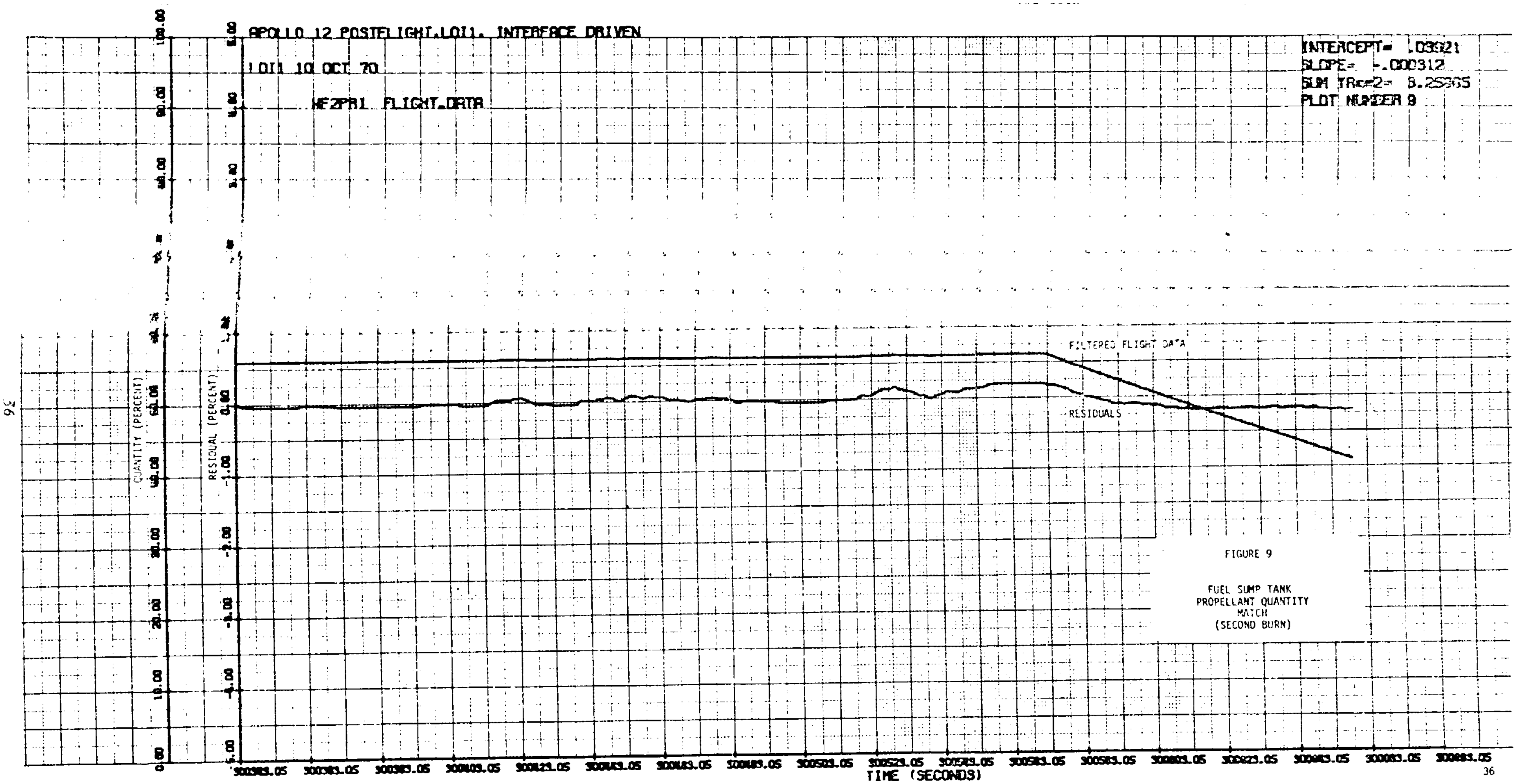


FIGURE 9

FUEL SUMP TANK
PROPELLANT QUANTITY
MATCH
(SECOND BURN)

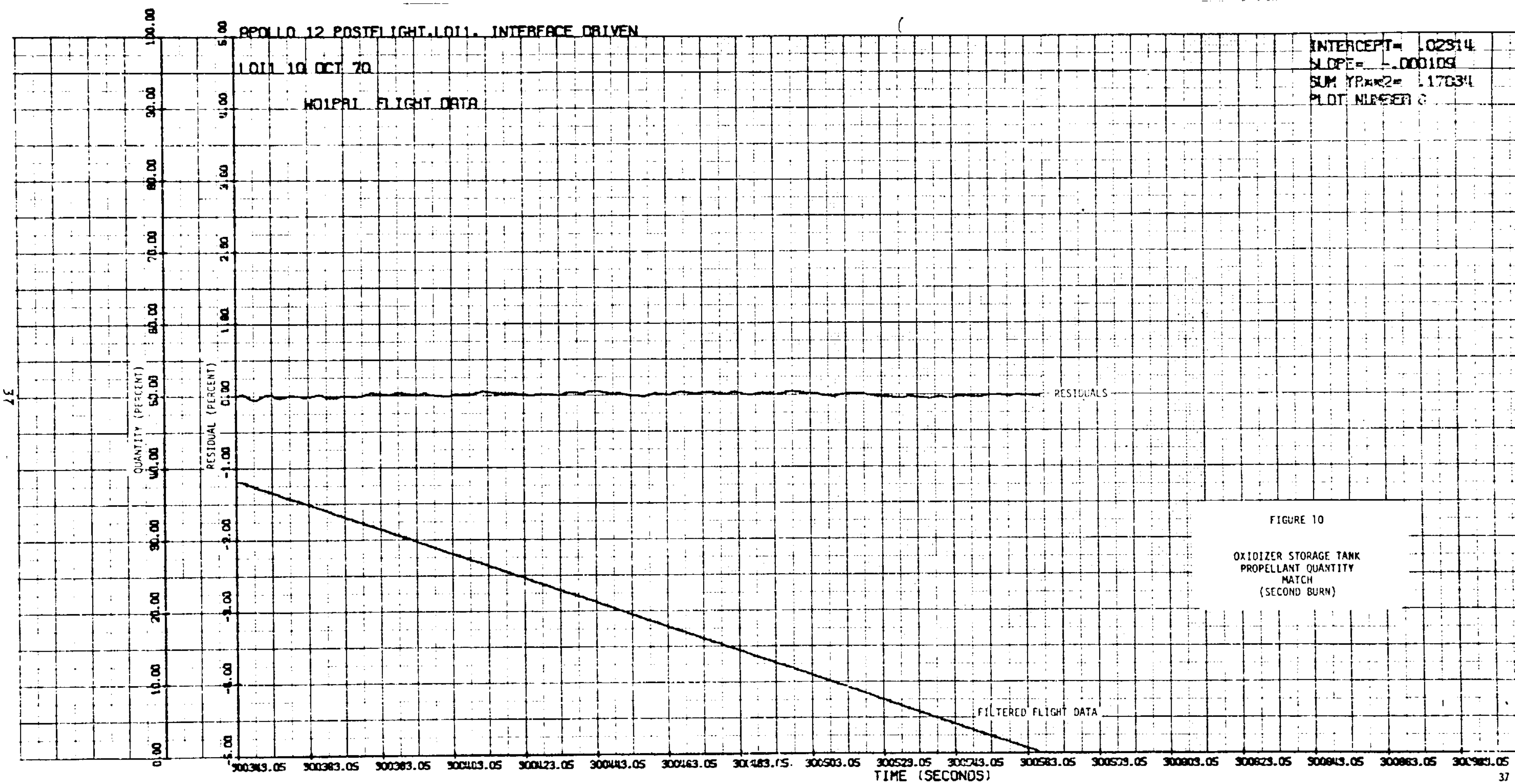
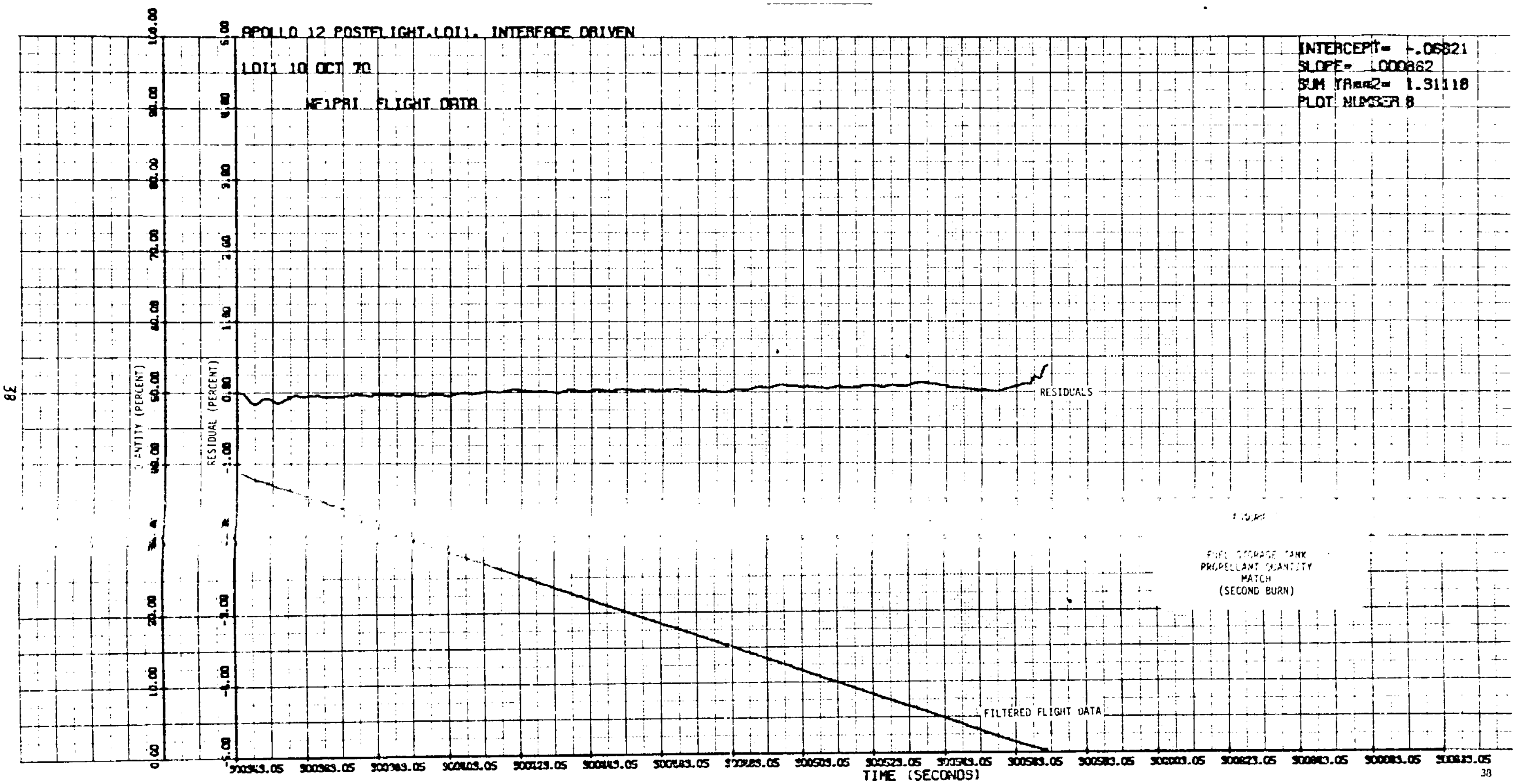


FIGURE 10

OXIDIZER STORAGE TANK
PROPELLANT QUANTITY
MATCH
(SECOND BURN)


APOLLO 12 POSTFLIGHT.LOI1. INTERFACE DRIVEN
LOI1 10 OCT 70
WF1PRI FLIGHT DATA
INTERCEPT= -.06821
SLOPE= .000862
SUM YR**2= 1.31118
PLOT NUMBER 8
RESIDUALS
FILTERED FLIGHT DATA
FUEL STORAGE TANK
PROPELLANT QUANTITY
MATCH
(SECOND BURN)
TIME (SECONDS)
RESIDUAL (PERCENT)
QUANTITY (PERCENT)

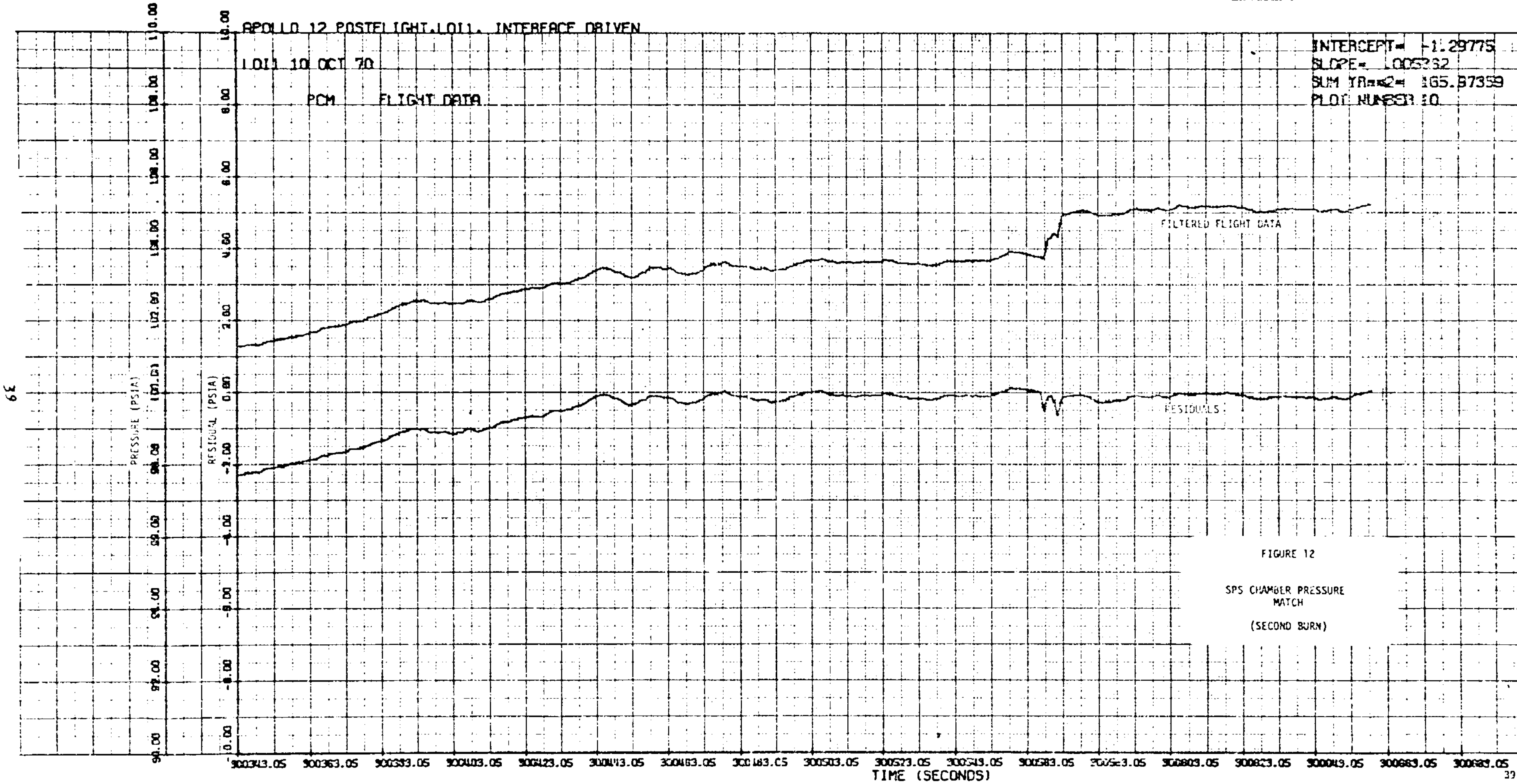
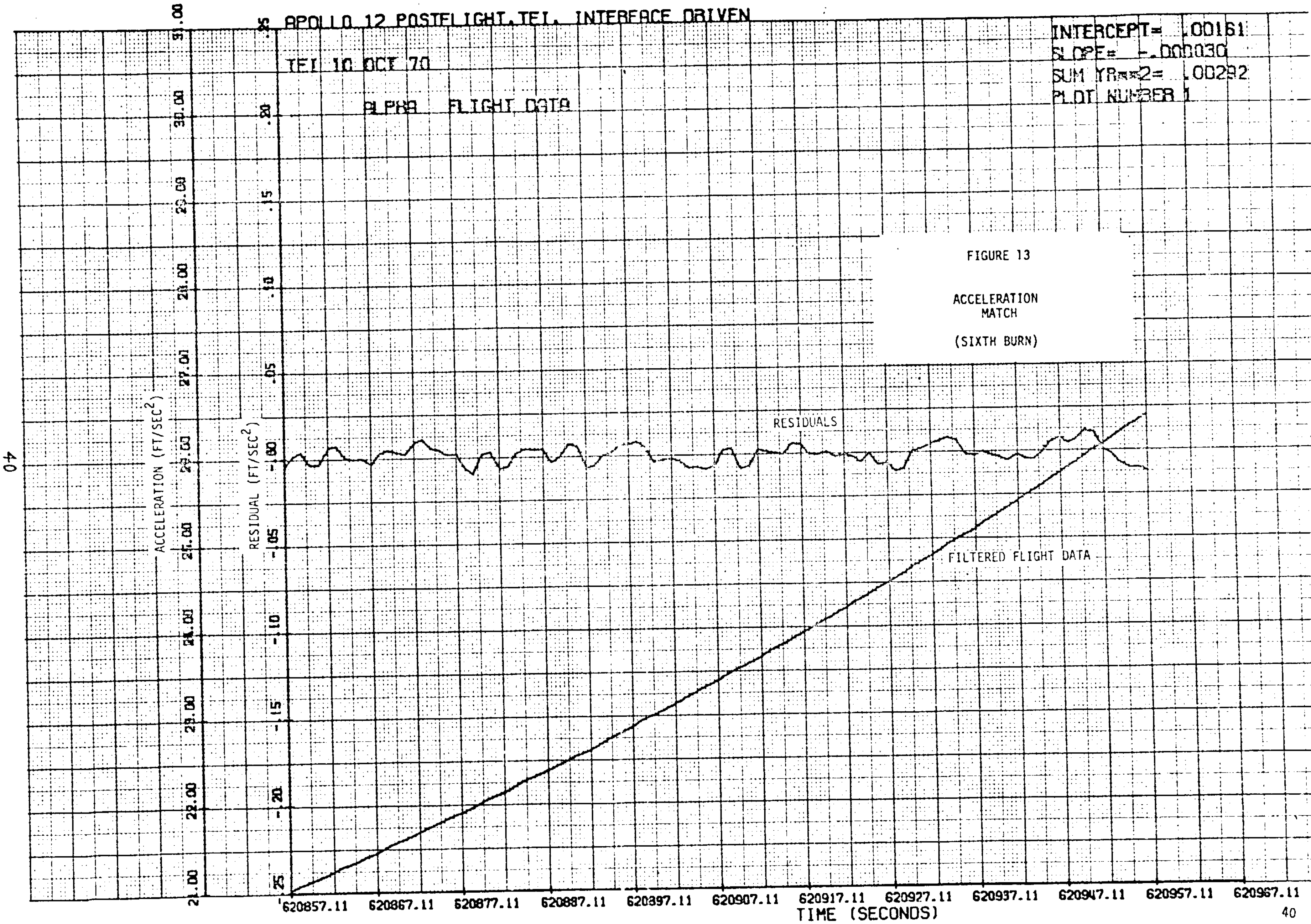


FIGURE 12

SPS CHAMBER PRESSURE MATCH

(SECOND BURN)


APOLLO 12 POSTFLIGHT.TEI. INTERFACE DRIVEN
TEI 10 OCT 70
ALPHA FLIGHT DATA
INTERCEPT= .00161
SLOPE= -.000030
SUM YR**2= .00292
PLOT NUMBER 1
FIGURE 13
ACCELERATION MATCH
(SIXTH BURN)
RESIDUALS
FILTERED FLIGHT DATA
ACCELERATION (FT/SEC^2)
RESIDUAL (FT/SEC^2)
31.00
30.00
29.00
28.00
27.00
26.00
25.00
24.00
23.00
22.00
21.00
.25
.20
.15
.10
.05
-.00
-.05
-.10
-.15
-.20
-.25
620857.11
620867.11
620877.11
620887.11
620897.11
620907.11
620917.11
620927.11
620937.11
620947.11
620957.11
620967.11
TIME (SECONDS)
40

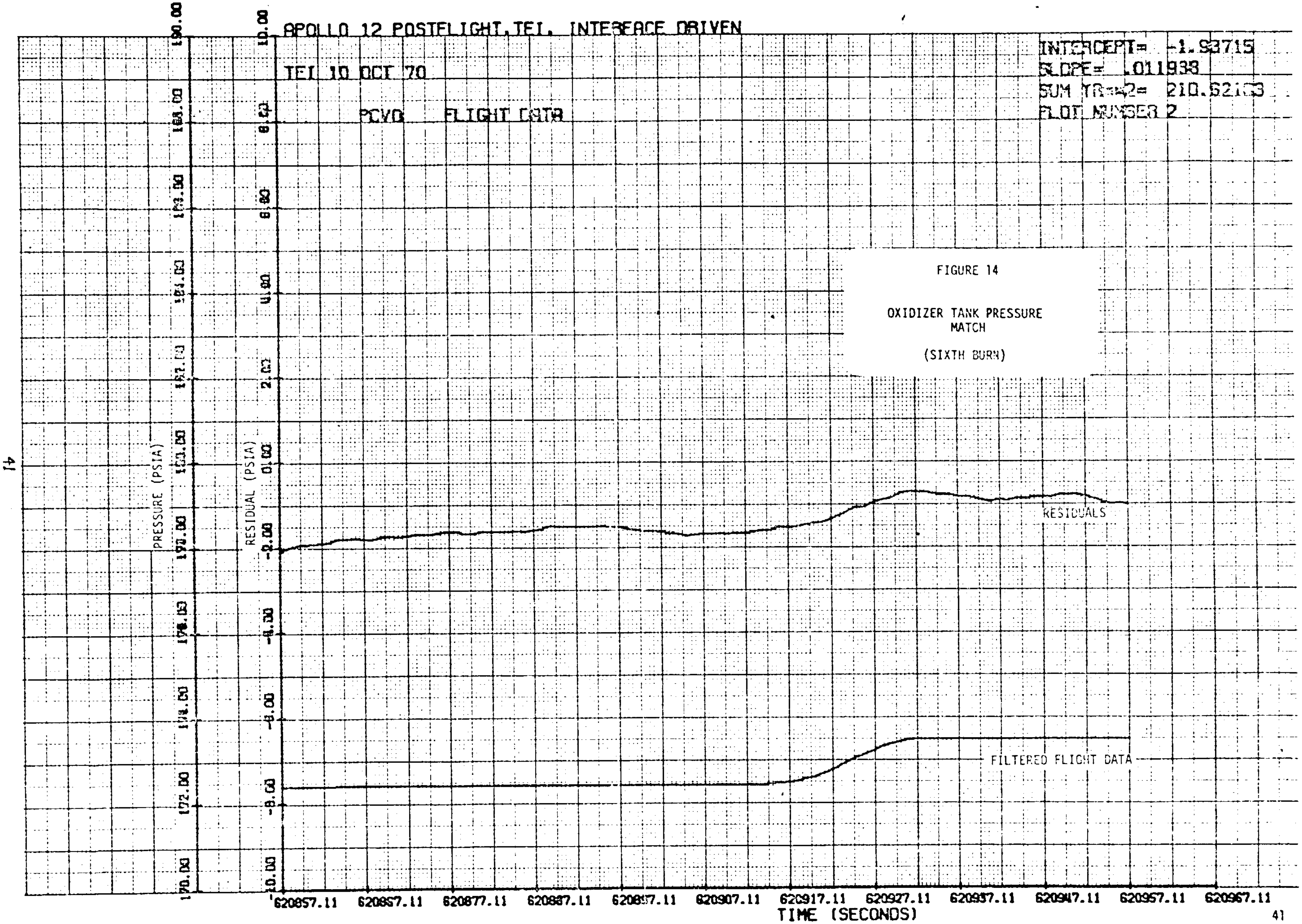


FIGURE 14

OXIDIZER TANK PRESSURE MATCH

(SIXTH BURN)

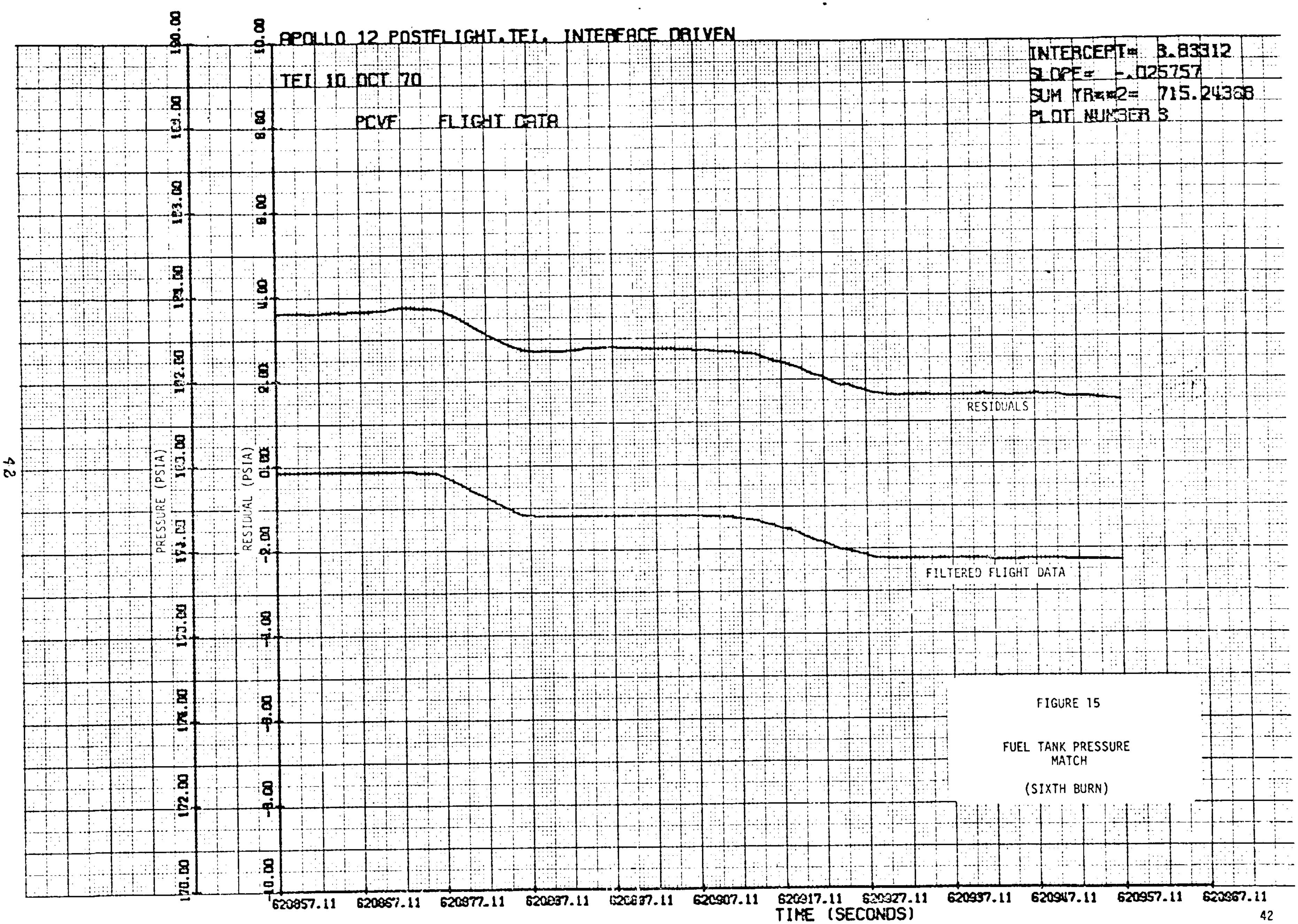


FIGURE 15

FUEL TANK PRESSURE MATCH

(SIXTH BURN)

42

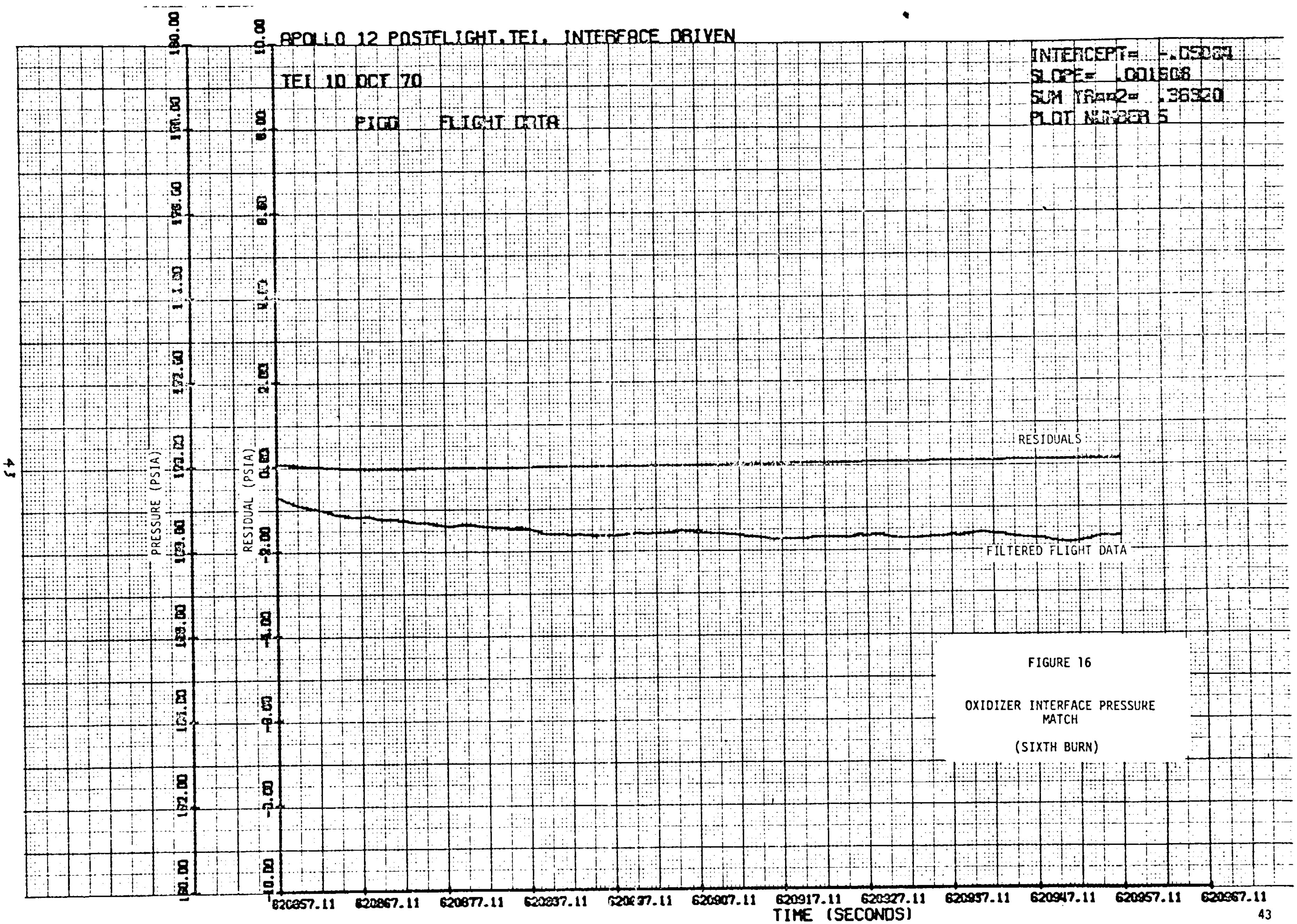


FIGURE 16

OXIDIZER INTERFACE PRESSURE MATCH

(SIXTH BURN)

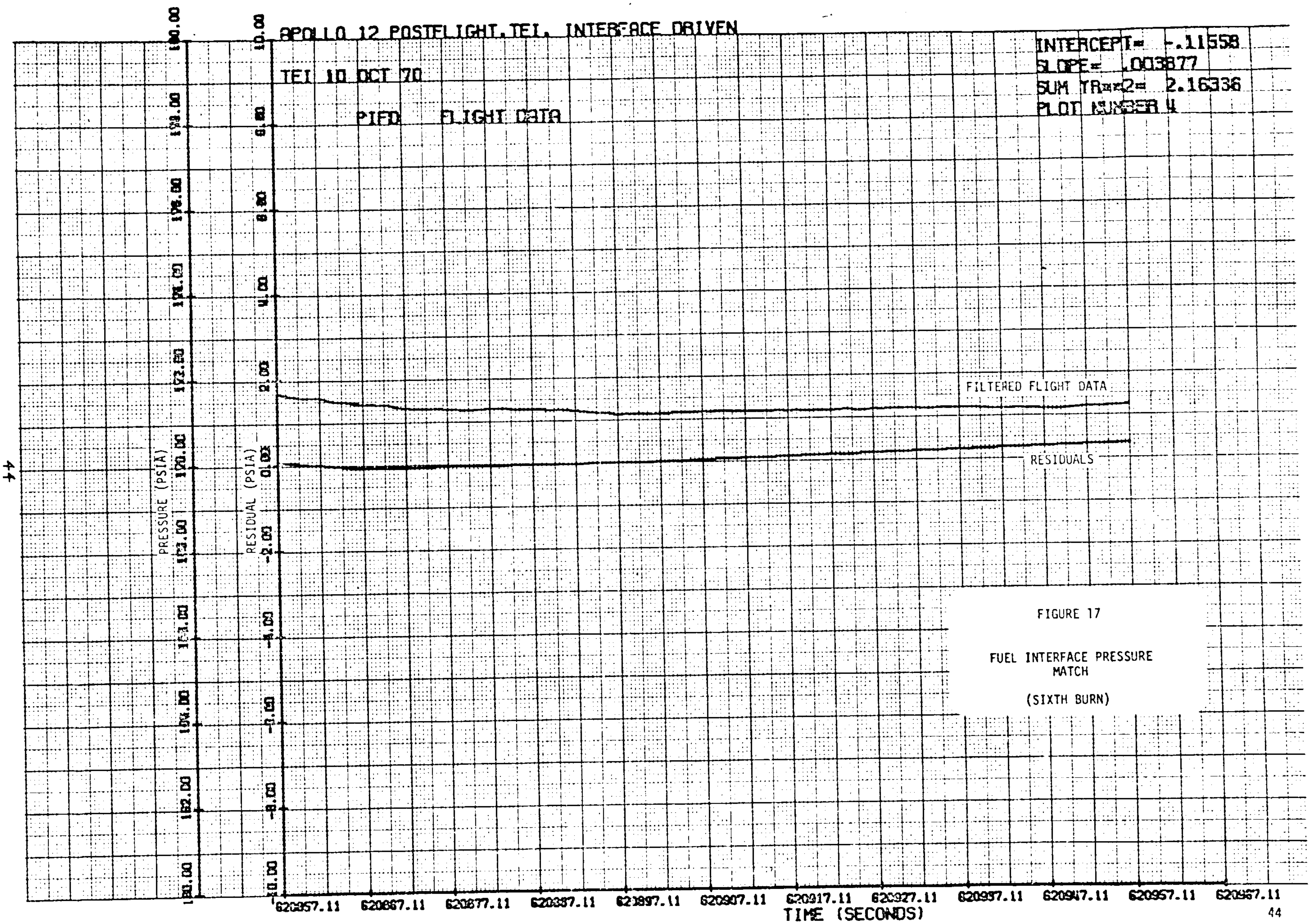


FIGURE 17

FUEL INTERFACE PRESSURE MATCH

(SIXTH BURN)

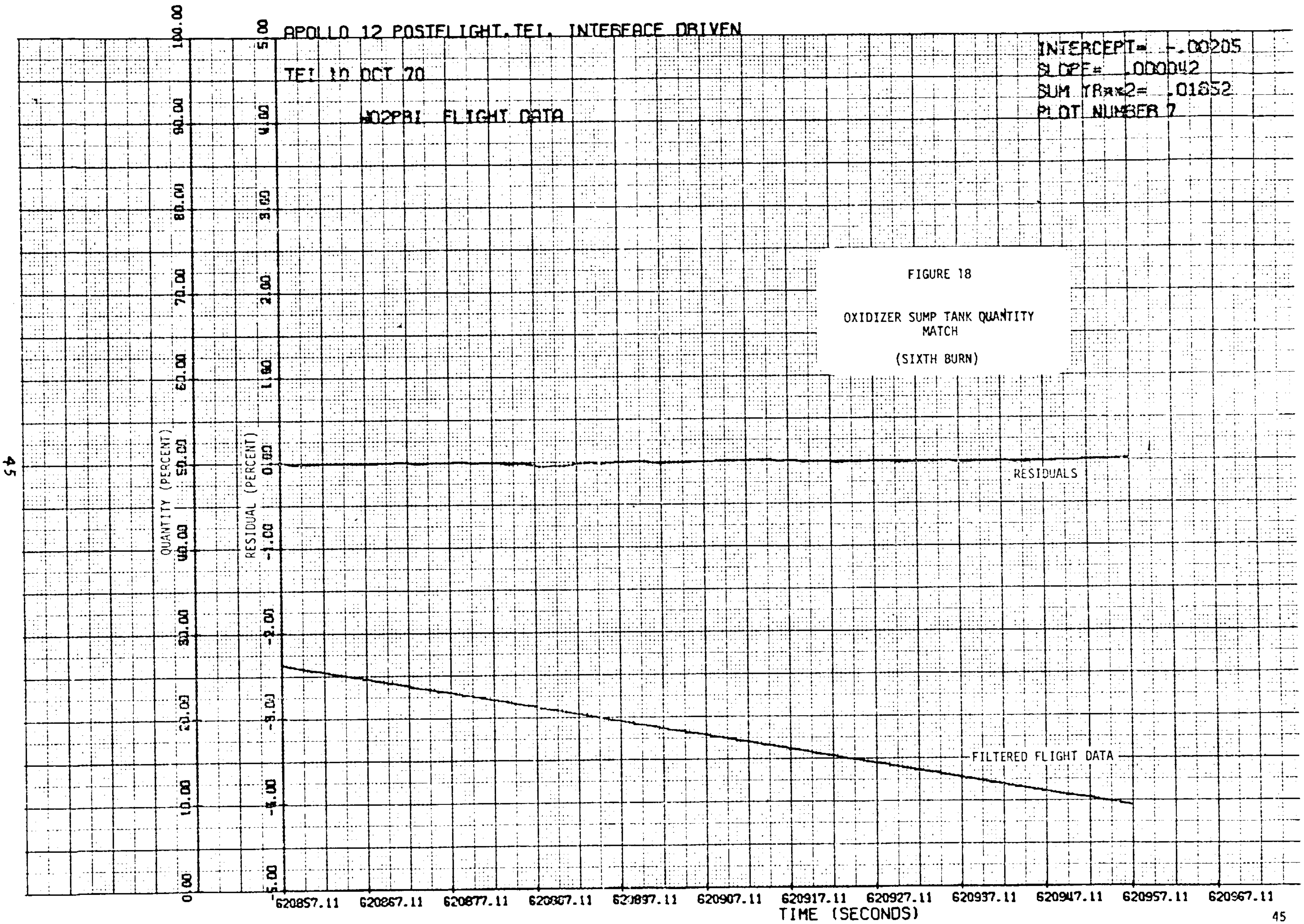


FIGURE 18

OXIDIZER SUMP TANK QUANTITY MATCH

(SIXTH BURN)

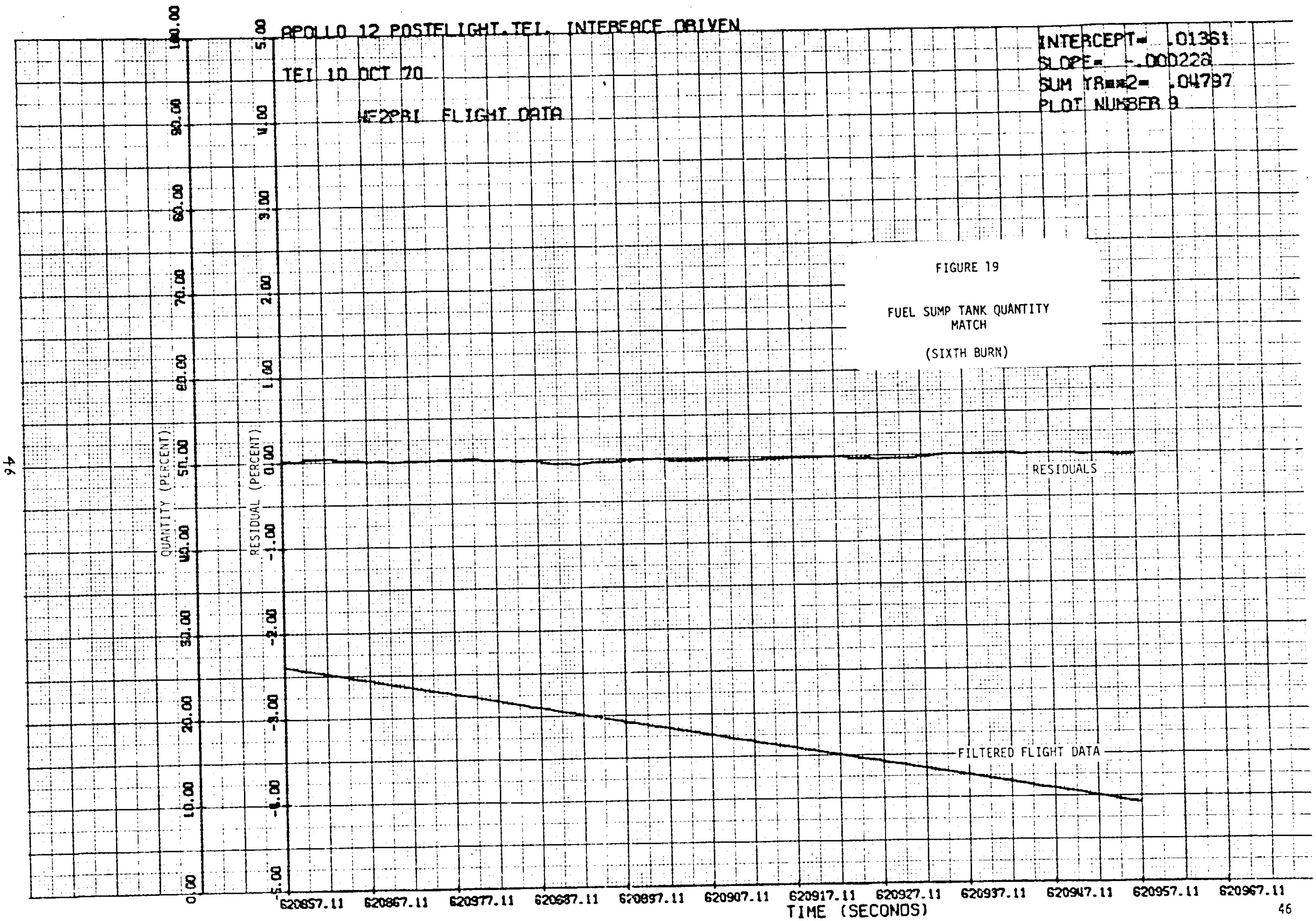


FIGURE 19

FUEL SUMP TANK QUANTITY MATCH

(SIXTH BURN)

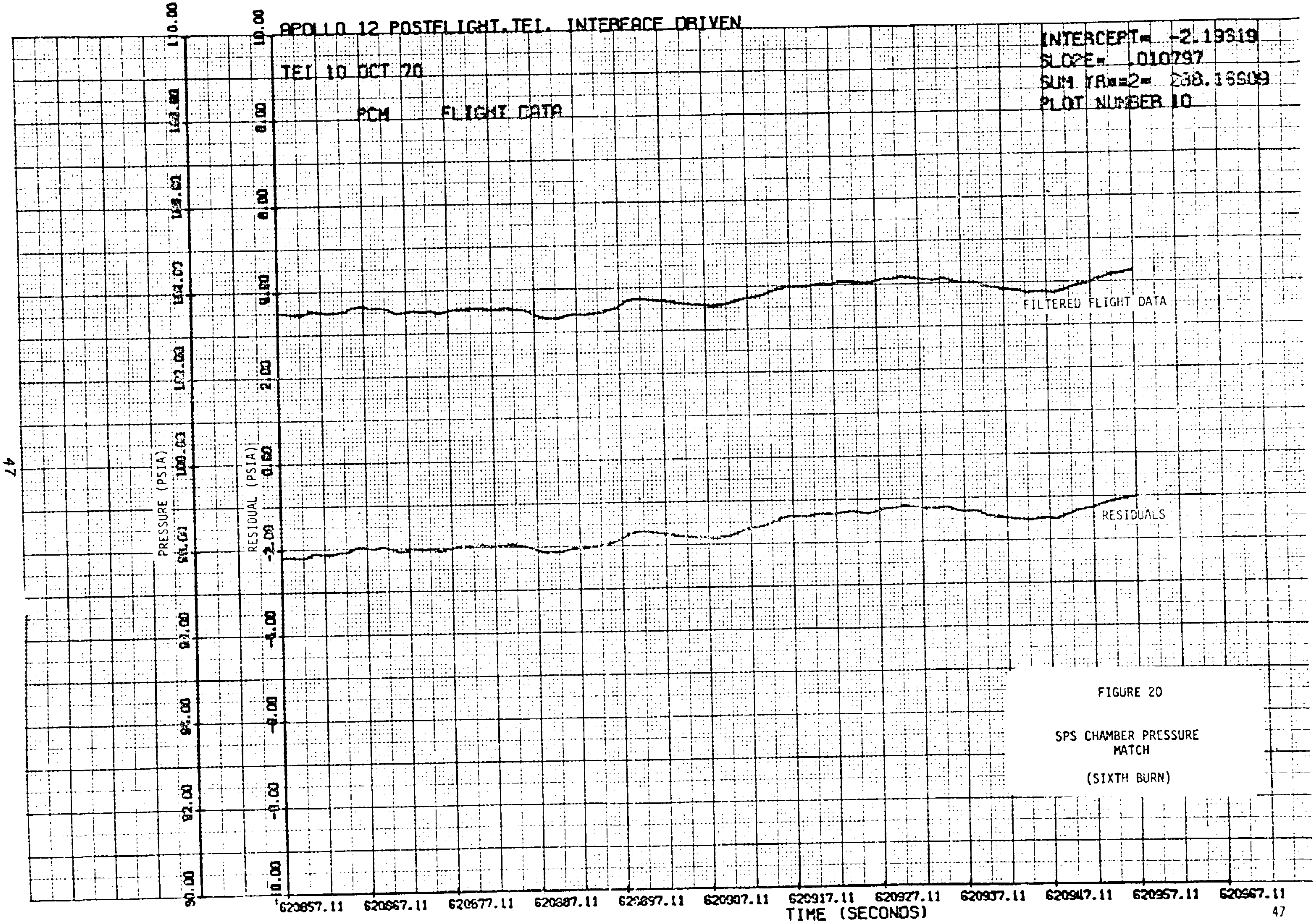


FIGURE 20

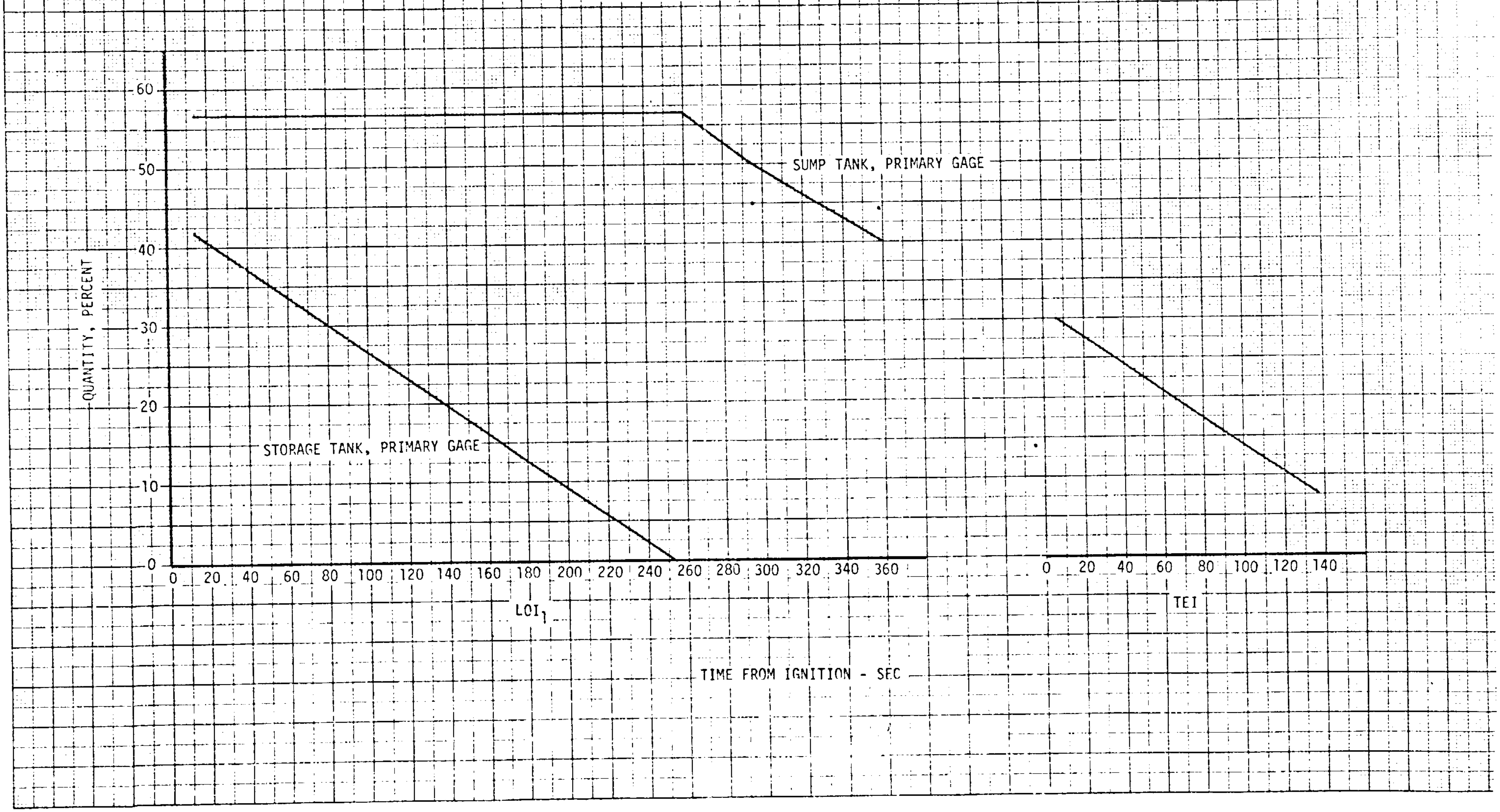
SPS CHAMBER PRESSURE MATCH

(SIXTH BURN)

FIGURE 21

TELEMETERED GAGING QUANTITIES

OXIDIZER



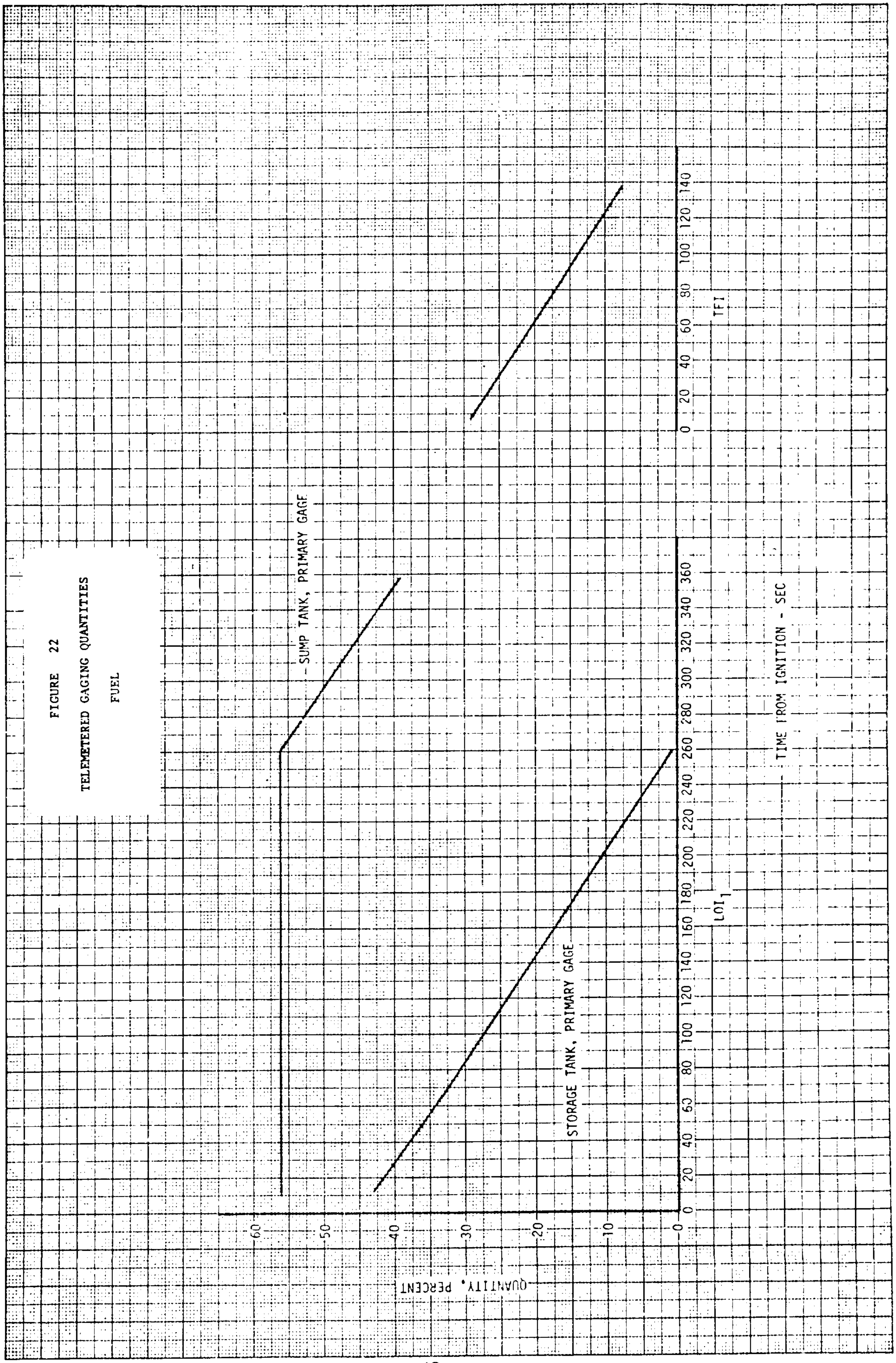


FIGURE 22
TELEMETERED GAGING QUANTITIES
FUEL

FIGURE 23

SPS INDICATED PROPELLANT UNBALANCE