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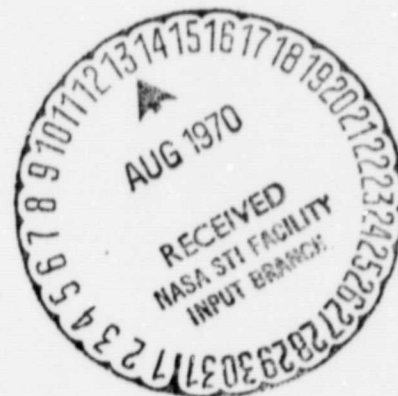
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APOLLO 7  
CSM 101  
SERVICE PROPULSION SYSTEM  
FLIGHT EVALUATION

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11 February 1969

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## SUMMARY

The performance of the Spacecraft 101 service propulsion system during the Apollo 7 Mission was evaluated and found to be satisfactory. The steady-state performance was determined by analyzing the fifth SPS burn using the Apollo Propulsion Analysis Program. From the analysis, it was determined that the best estimate of the flight performance during the fifth burn was as follows: thrust, 21180 pounds; specific impulse, 314.0 seconds; and mixture ratio, 1.605. The engine flight performance corrected to standard inlet conditions resulted in the following values: thrust, 20721 pounds; specific impulse, 313.9 seconds; and mixture ratio, 1.603. These values are 1.10%, 0.41% and 0.19% greater, respectively, than the values reported for the acceptance tests of the engine.

The Propellant Utilization and Gaging System operated nominally during all burns, however, the fuel primary gaging system accuracy was less than anticipated. As planned, two oxidizer and two fuel point sensors were uncovered during the fifth burn.

Two minimum impulse burns were performed satisfactorily, with the impulse from each burn comparing well with the present nominal performance. It is recommended that the present nominal minimum impulse performance data in the Spacecraft Operational Data Book be amended to include the statistical tolerances associated with that data.

The service propulsion thermal control system adequately maintained the required propellant line and engine valve temperatures above their minimum limits throughout the mission.

## INTRODUCTION

The Apollo 7 Mission included the first flight test of the Block II Service Propulsion System. The major difference from the Block I system was the change in the nominal propellant mixture ratio from 2.0 to 1.6 which, when coupled with other minor design improvements, results in some improvement in the engine design performance. The Apollo 7 Mission utilized CSM 101 which was equipped with the SPS engine S/N 59 (Injector S/N 129). The engine configuration and expected performance characteristics (Reference 1) are contained in Table 1.

As planned, there were eight firings of the SPS engine. The actual burn times and durations are shown in Table 2. The ignition time for the third burn was advanced 16 hours from the original flight plan to improve the margin of deorbit capability with the Service Module Reaction Control System. To insure the verification of the propellant gaging system, the firing time for the fifth burn was increased from 61 to 66 seconds to guarantee that both point sensors would be uncovered during steady-state operation. A 3-hour cold soak test was performed following the fifth burn. The engine was operated in the dual bank valve mode on all firings and the SPS PU valve was in the normal position throughout the mission.

There were four primary mission test objectives relating to the SPS. They were:

- P3.14 SPS Minimum Impulse Burn
- P3.15 SPS Performance
- P3.16 Prim/Aux Propellant Gaging System
- P3.20 SPS Propellant Thermal Control

The detailed requirements of these objectives are described in Reference 1.

## STEADY-STATE PERFORMANCE ANALYSIS

### Analysis Technique

The major analysis effort was concentrated on determining the flight steady-state performance of the SPS during the fifth burn. The fifth burn was the only SPS burn of sufficient duration to warrant detailed performance analysis. The performance analysis was accomplished by use of the Apollo Propulsion Analysis Program which utilizes a minimum variance technique to "best" correlate the available flight and ground test data. The program embodies error models for the various flight and ground test data that are used as inputs, and by iterative methods arrives at estimations of the system performance history, propellant weights and spacecraft weight which "best" (minimum-variance sense) reconcile the available data.

### Analysis Program Results

The SPS steady-state performance was determined from the analysis of a 44-second segment of the fifth SPS burn. The segment of the burn analyzed commenced approximately 11.5 seconds after SPS ignition (FS-1), and included the flight time between 594012 and 594056 seconds G.E.T. The first 11.5 seconds of the burn were not included, to reduce any errors resulting from data filtering spans which include transient data. The time segment analyzed was terminated approximately 11.4 seconds prior to SPS shutdown (FS-2) for similar reasons. Because the resulting 44-second data segment is somewhat shorter than desirable for detailed performance analysis, larger than usual tolerances must be placed on the program results.

The Apollo Propulsion Analysis Program results presented in this report were based on simulations using data from the flight measurements listed in Table 3. The propellant densities were calculated from sample specific gravity data from KSC, a flight temperature of 61°F, and flight interface pressures. The 61°F temperature was an estimate based on all the feed-system temperature measurements. The estimated spacecraft damp weight (CSM minus SPS propellants) at SPS 5 ignition was obtained from the Apollo Spacecraft Program Office, and was assumed constant throughout the burn. The initial estimates of the SPS propellants onboard at the beginning of the time segment analyzed were extrapolated from the loaded propellant weights.

The results of the propulsion analysis program simulation are contained in Table 4 along with the preflight predicted values. The values presented are for a time slice approximately 56 seconds following FS-1, and are considered representative of the actual flight values throughout the segment of burn analyzed. The computed flight performance values shown in Table 4 indicate the SPS performance was satisfactory and within the expected tolerances.

Several apparent instrumentation discrepancies were indicated by the post flight simulation of the fifth burn. These discrepancies were determined by calculating the values which would best correlate all the data. The measured oxidizer interface pressure exhibited a slight decay (approximately 0.5 psi) over the portion of the burn from FS-1 + 12 seconds to FS-1 + 56 seconds. This trend did not agree with the increasing oxidizer tank pressure, which indicated the oxidizer interface pressure should have risen approximately 0.8 psi over the time

interval in question. Furthermore, use of the measured oxidizer interface pressure profile in the analysis program resulted in a poor match to the acceleration profile. The analysis also indicated a bias of 2 to 3 psi existed in the oxidizer interface pressure data. This was partially substantiated by the fact that prior to each SPS burn the oxidizer interface pressure read 1 to 3 psi lower than the oxidizer tank pressure. Prior to SPS thrusting the two measurements should be equal.

The analysis indicated the fuel interface pressure was reading approximately 3 psi low throughout the burn. The time history of the measured chamber pressure during the fifth burn is shown in Figure 1. The measured chamber pressure appeared to be approximately 1.0 psi low early in the burn and exhibited a positive drift of approximately 1.3 psi over the burn. Similar chamber pressure drifts and biases were evident on the other Apollo 7 burns and have been observed consistently on previous Block I SPS flights. The Propellant Utilization and Gaging System discrepancies are discussed separately in a later section.

Shown in Figures 2 through 5 are analysis program output plots which represent the residual errors, or differences between the filtered flight data and program-calculated values. The figures represent thrust acceleration, oxidizer sump tank primary PUGS readings, oxidizer tank pressure, and fuel tank pressure, respectively. Because of the questionable nature of the chamber pressure data (discussed previously) and of the fuel sump tank PUGS primary data (discussed in the PUGS Evaluation Section), data from these measurements were not matched directly in the program. However, point sensor data from the fuel auxiliary PUGS were utilized in the program by inputting the fuel quantity indicated by

the uncover time of the 15th point sensor.

A strong indication of the validity of the analysis program simulation can be obtained by comparing the thrust acceleration calculated in the simulation to that derived from the Apollo Command Module Computer (CMC)  $\Delta V$  data transmitted via measurement CG0001V. Figure 2 shows the thrust acceleration during the portion of the burn analyzed, as derived from the CMC data, and the residual error between the CMC and program calculated values. The residual error time history has essentially a zero mean, and little, if any, discernable trend. This indicates the simulation is relatively valid, although other factors must also be considered in critiquing the simulation. The previously discussed instrumentation discrepancies and the short burn time made the performance analysis quite difficult. Because of these apparent instrumentation discrepancies it was not possible to complete a simulation which gave a consistent match of all the measured data. Further, the short burn time meant that the total data sample available was insufficient for the program to completely resolve the inconsistencies. In attempting to resolve these problems several program simulations were run during the evaluation in which the various propulsion system measurements were given more or less weight in the solution. The values reported herein are from the simulation giving the best match to the Command Module Computer (CMC) acceleration data, and are therefore considered to give the best estimate of the specific impulse, 314.0 seconds. The specific impulse values from other simulations which also gave reasonable data matches were between 313.2 and 314.1 seconds. Therefore, although certain inconsistencies were not fully resolved, it is felt that the performance values quoted herein

are relatively accurate and that the conclusion that the flight performance was nominal is valid.

#### Comparison with Preflight Performance Prediction

Prior to the Apollo 7 Mission the expected performance of the SPS was presented in Reference 1. This performance prediction was for the integrated propellant feed/engine system and was characteristic of the SPS hardware on this flight. Thus, it was a preflight estimate of the propulsion system performance under space flight conditions, with no restrictions placed upon the conditions at the inlets to the engine.

The predicted thrust, specific impulse, and mixture ratio for the portion of the fifth burn analyzed are compared to their corresponding analysis program-calculated inflight time histories in Figure 6. As shown in Figure 6, and previously shown in Table 4, the preflight predicted and inflight computed performance agree favorably near the end of the time segment and are within the expected tolerances contained in Reference 1. During the initial portion of the time segment analyzed, and near the start of the burn, the inflight computed thrust is somewhat greater than the 3 sigma predicted value. The initial inflight calculated mixture ratio is also greater than the 3 sigma predicted value. Both of these differences from predicted are attributed mainly to the fact that the oxidizer tank pressure at the start of the burn was about 3 psi higher than expected in the preflight prediction. After sufficient burn time had elapsed for the tank ullage pressures to reach equilibrium with the helium regulator setting, the inflight computed and preflight predicted performance agreed satisfactorily.

### Engine Performance at Standard Inlet Conditions

The expected flight performance of the SPS engine was based on the 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. The Apollo 7 SPS flight performance corrected to standard inlet conditions yielded a thrust of 20721 pounds, a specific impulse of 313.9 seconds, and a propellant mixture ratio of 1.603. These values are 1.10% greater, 0.41% greater and 0.19% greater, respectively, than the values reported for the acceptance tests of the engine. The standard inlet conditions thrust value from flight is slightly greater than the upper specified value of 20705 lbs. However, considering the tolerances in the flight analysis value, this is acceptable. 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 <sup>3</sup>        | 90.15   |
| Fuel density, lbm/ft <sup>3</sup>            | 56.31   |
| Thrust acceleration, lbf/lbm                 | 1.0     |
| Throat area (initial value), in <sup>2</sup> | 121.641 |

Of primary concern in the flight analysis of all Block II engines will be 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 in the injector sea level acceptance tests using conversion factors determined from AEDC qualification testing. Although the specific impulse determined for this flight is somewhat greater than the expected value, it is within the stated uncertainty, and when the tolerances on the flight value are also considered, there appears to be no evidence that the present method of calculating vacuum specific impulse should be changed. The validity of this conclusion should be continually verified on each subsequent flight.

#### Comparison of all Steady State Data

The steady state data for SPS burns 1, 2, 3, 7, and 8 were also reviewed and compared with burn 5. Because these burns were relatively short, no detailed performance analysis was attempted. The minimum impulse burns, 4 and 6, are covered in the later transient discussions. Table 5 summarizes the basic flight data. The pressure values shown are representative of the middle of each burn. The data indicate that all burns were repeatable and considered nominal.

## PUGS EVALUATION

### Propellant Loading

Oxidizer - The service propulsion oxidizer sump tank was loaded with nitrogen tetroxide on 2 October, 1968. The oxidizer sump tank was filled to the top of the crossover line standpipe, and the sump tank primary gaging system probe was calibrated. Oxidizer was then drained from the sump tank until the required flight load was indicated on the CM oxidizer display. At that time, the CM oxidizer display indicated 22.4% and the tank ullage pressure was 110 psia. When the oxidizer tank was pressurized to 190 psia for leak check the display indicated 22.3%. The oxidizer temperature was 71°F.

Fuel - On 3 October, 1968, the fuel sump tank was loaded with Aerozine-50 in a manner similar to the oxidizer tank. With the flight load onboard and a tank ullage pressure of 110 psia, the CM fuel display indicated 22.3%. When the fuel tank ullage pressure was increased to 190 psia, the CM fuel display read 22.2%. The fuel temperature was 72°F. Prior to launch, a small amount of fuel (estimated at 20 lbs) was inadvertently drained overboard.

Propellant Densities - Density measurements were made on one oxidizer sample and one fuel sample. The analysis yielded an oxidizer density of 90.16 lbm/ft<sup>3</sup> at the loaded temperature of 71°F and under a pressure of 190 psia. At 72°F and under a pressure of 190 psia, the fuel density was 56.42 lbm/ft<sup>3</sup>.

The total propellant loads, as calculated from measured densities and the CM display readings, were as follows:

# PROPELLANT LOADS

| Propellant            | Total Mass Loaded, lbm |         |
|-----------------------|------------------------|---------|
|                       | Actual <sup>b</sup>    | Planned |
| Oxidizer <sup>a</sup> | 6027                   | 6029    |
| Fuel <sup>a</sup>     | 3711 <sup>c</sup>      | 3728    |
| Total                 | 9738                   | 9757    |

<sup>a</sup>Includes gageable, ungageable, and vapor-loaded quantities.

<sup>b</sup>Reported by KSC in Spacecraft Operational Data Book.

<sup>c</sup>Assumes 20 lbm drained overboard prior to launch.

## PUGS PERFORMANCE

The Propellant Utilization and Gaging System (PUGS) was operated in the primary mode for all SPS firings except the fifth. For the fifth burn, the PUGS was switched to the auxiliary mode which provided primary sump tank and total auxiliary (point sensors) T/M propellant mass readings. Since the storage tanks should be essentially empty, the total auxiliary reading should equal the primary sump tank values during the fifth burn. A known bias exists in the sump tank gage readings, because of the difference in liquid levels in the sump tanks and inside the gaging system stillwells. The stillwell is a manometer that balances the pressure at the bottom of the stillwell with a fluid head. Under nonflow conditions, this fluid head is equivalent to the level of propellant in the tank. However, when propellant is flowing, the fluid head in the stillwell is reduced by the dynamic head of the propellant flowing by the bottom of the stillwell through the zero-gravity retention reservoir. The size of the liquid level bias decreases as the vehicle acceleration increases with burn time. S/C 101 is the last vehicle to have this bias. The bias has not been well defined for Block II, and the value used was extrapolated from the Block I bias. Because the initial bias is not accurately known and the bias changes with burn time (or acceleration) there is an added uncertainty in the PUGS reading beyond the typical PUGS tolerances.

The oxidizer primary gaging system operated nominally throughout the mission. Following the 4.5-second lockout period after ignition of the first SPS burn, the oxidizer reading was approximately 0.7 percent greater than expected based on the initial oxidizer load. The Apollo

Propulsion Analysis Program simulation of the fifth burn, along with the auxiliary point sensor data during the fifth burn, also indicated an oxidizer primary probe bias of approximately + 1.0 percent. During the third SPS burn, the storage tank oxidizer primary probe was showing a small reading of about 0.5 percent. Oxidizer primary gaging shifts of approximately 0.4 percent in the positive direction have been observed at WSTF and during the S/C 102 flight readiness-firings. These shifts have been attributed to propellant vapor in the tanks. The oxidizer primary probes are zero-point calibrated prior to loading oxidizer into the tanks.

The fuel primary gaging system displayed several characteristics that are not fully explained at this time. Between burns 2 and 3, burns 3 and 5, and burns 5 and 7, small shifts of approximately 0.5 percent were noted in the T/M readings. On all burns, the fuel primary reading either moved upward or failed to show the expected decrease after the end of the lockout period. Following the upward shifts, the fuel primary probe indicated a high flow rate but appeared to be approaching a more realistic reading toward the end of the burns. During burns 3, 7, and 8, the storage tank fuel primary probe was indicating positive readings as large as 1.5 percent.

The crew reported that following the third SPS burn, the CM displays showed approximately 17.15 percent for oxidizer and 18.20 percent for fuel. When the PUGS switch is in the primary position, the CM display shows the sum of the sump and storage tank primary readings both for oxidizer and fuel. It appears that the higher fuel reading was the result of (1) the upward fuel shift after lockout

and (2) the fact that the fuel storage tank primary probe was indicating higher values than the oxidizer storage tank probe. During the fifth burn, the CM displays show oxidizer readings higher than the fuel readings. This change from fuel high in burn 3 to oxidizer high in burn 5 was due to switching from the primary to the auxiliary (point sensors) system on the CM display.

During the fifth burn, the fourteenth and fifteenth point sensors in both the oxidizer and fuel sump tanks were uncovered, as planned. As noted previously, the oxidizer point sensor data verified that the oxidizer primary probe was reading high by approximately 1.0 percent. The fuel point sensor data agreed well with the fuel primary data near the end of burn, but showed the primary to be somewhat high early in the burn. This high reading on the fuel primary probe early in the burn appeared to be associated with the upward shifts following the lockout period which were discussed previously. Because of these questionable shifts in the fuel primary data, only the fuel point sensor data was used in the performance analysis. Both the oxidizer point sensor and primary probe data were used, however.

Summarizing the performance of the PUGS:

1. The oxidizer primary gaging system operated as expected but exhibited an initial upward shift of about 0.7 percent and remained high throughout the mission.
2. The fuel primary gaging system exhibited shifts between burns and also abnormal upward shifts after lock-out. In general, the accuracy of the fuel primary gaging system for the short burns and the initial portion of the fifth burn

was less than expected.

3. The auxiliary system operated normally during burn 5.

## PRESSURIZATION SYSTEM EVALUATION

The SPS pressurization system operated nominally throughout the mission. There was no indication of leakage. The helium supply pressure and the propellant ullage pressures indicated a nominal helium usage for each of the eight SPS burns.

## ENGINE TRANSIENT ANALYSIS

An analysis of the start and shutdown transients of all eight firings was completed. A summary of results for the six full burns is presented in Table 6 along with appropriate specifications and acceptance test data. For comparative purposes similar data for Apollo 6 (Block I engine) are included.

Representative curves of the first start (burn 1), a restart (burn 2), a restart and shutdown (burn 5), and both minimum impulse burns (burns 4 and 6) are shown in Figures 7a, 7b, 8, 9a, and 9b.

The analysis of the engine transient operation is significantly affected by the fact that the flight chamber pressure measurement does not exhibit good transient response characteristics. This fact has been clearly demonstrated in recent altitude testing at AEDC where direct comparisons were made between flight instrumentation and close coupled, high response instrumentation. The results of the tests are still being reviewed, but the test data did indicate that the flight chamber pressure measurement can be grossly and randomly inaccurate during the start transient. This was particularly evident in the chamber pressure overshoot where the magnitude indicated by the flight instrument was as much as 50 psi higher than the corresponding close coupled instrumentation. The overshoot duration showed a similar disagreement. These effects are believed to be due to thermal effects on the flight transducer diaphragm. In view of this, the transient analysis of the engine is difficult and the results are, at best, questionable.

The pressure profiles during the start and overshoot varied significantly from burn to burn. Some of this variability is undoubtedly due

to thermal effects on the flight pressure unit, and the sample rate of one sample every 10 milliseconds. However, similar start characteristics have been observed in previous flights and ground tests, and are not considered unusual.

The pertinent engine transient characteristics are discussed below. Start Impulse - One important item to note is the technique used to compute the start impulse to 90% chamber pressure. The accepted technique involves taking the chamber pressure integral from FS-1 to the first time that chamber pressure reaches 90% of the steady state value.

Since a mean chamber pressure line is not used, this value may be significantly affected by the normal fluctuations in chamber pressure during the transient. An example of this is shown in Figure 8. The pressure fluctuations in Figure 8 move the 90% point to the left, which decreases the impulse to 90% value. It is believed that this results in excessive variability in the computed impulse values. The technique for determining start impulse is being reviewed to determine methods of reducing this error.

Although the impulse values were admittedly affected by the previously mentioned pressure transducer characteristics and the measurement technique, with the exception of the seventh burn all start impulse values were within specification requirements.

Start Time - Table 6 shows that the time from ignition to 90% steady state thrust was within specification for all six full burns.

Chamber Pressure Overshoot - As noted in Table 6 the overshoot of the second burn exceeded the chamber pressure measurement range of 150 psia. This is not unusual since overshoots of 158 psia have been observed during specially instrumented ground test. The measured overshoot peaks and

durations (1000 ms) were as expected. Ground tests indicate actual overshoots are only of 20-50 milliseconds duration and the flight measurement tends to greatly amplify them. Thus, it is believed that the 1000 milliseconds times are not indicative of actual durations.

Shutdown Impulse - The total impulse as calculated from cutoff signal to 0% steady-state thrust was very repeatable and within specification for all six full burns (Table 6). All shutdowns appeared to be normal.

Shutdown Time - The time measured from cutoff signal to 10% of steady state thrust was repeatable and within specification (Table 6).

Minimum Impulse Firings - Total impulse during the minimum impulse firings (burns 4 and 6) was calculated from both chamber pressure and incremental velocity data (Table 7) with the latter values 1800 to 2800 pound seconds less than the former. Because of the lack of confidence in the chamber pressure measurement, the impulse values determined from velocity data are believed to be more indicative of actual performance. The impulse values, as determined from the velocity data, are plotted in Figure 10, along with the typical minimum impulse performance curves presently contained in the Spacecraft Operational Data Book (Reference 2). Both the fourth burn and sixth burn values are seen to agree well with the nominal impulse for a dual-bore, 0.5 second burn. The minimum impulse data presented in Reference 2 show only the nominal impulse values, and do not present the associated tolerances for those values. Considering the fact that the minimum impulse data contained in Reference 2 may be used for both mission planning and flight control purposes, it is recommended that the minimum impulse performance curves in Reference 2 be amended to include the statistical tolerances associated with that data.

Transient Spiking and Pops - No unusual transient chamber pressure spikes were noted. During the first burn an oxidizer interface pressure spike occurred at start. The maximum recorded pressure was 250 psia. However, this has been observed during ground tests and is considered normal for a dry start (no propellant initially between the ball valves-first burn). No chamber pressure "POPS" (low magnitude-short duration spikes) were observed during the flight. However, the capability of the flight chamber pressure instrumentation to record short duration pops or spikes is severely limited.

## SPS PROPELLANT THERMAL CONTROL

The SPS Thermal Control System was more than adequate to maintain the SPS propellant lines and the engine valve above their minimum temperature limits for the earth orbital mission. Thermal characteristics of the propellant lines were better than anticipated for random, drifting flight in that the rate of temperature decrease was less than predicted. Both the oxidizer and fuel engine feedlines had minimum temperatures of 55°F, while the system feedline minimum temperatures were 50°F for oxidizer and 52°F for fuel. The majority of the propellant line temperature decrease during the mission resulted from the colder tank propellant moving into the lines with each SPS burn. These tank propellant temperatures had decreased continually throughout the mission as expected.

The SPS bipropellant valve temperature was maintained above 50°F prior to each burn with SPS heater operation necessary before the sixth and eighth burns. The SPS heaters were activated one additional time for a 6-hour test of their heating rates; 3 hours in the single bank A position followed by 3 hours in the dual bank A/B position. The Heater System A operation showed engine line temperature increases of less than 1°F/hr and engine valve temperature increases of approximately 2°F/hr. The A/B bank operation showed approximate temperature increases of 2°F/hr for the engine lines and 3°F/hr on the engine valve. After the fifth burn, a 3-hour cold soak test was performed with no notable decrease in propellant line or engine valve temperatures.

TABLE 1

## S/C 101 SPS ENGINE AND FEED SYSTEM CHARACTERISTICS

|                                                                                                               |         |
|---------------------------------------------------------------------------------------------------------------|---------|
| Engine No.                                                                                                    | 59      |
| Injector No.                                                                                                  | 129     |
| Chamber No.                                                                                                   | 326     |
| Initial Chamber Throat Area (in. <sup>2</sup> )                                                               | 121.641 |
| Oxidizer Engine Feedline Resistance<br>(lb <sub>f</sub> -sec <sup>2</sup> /lb <sub>m</sub> -ft <sup>5</sup> ) | 476.    |
| Fuel Engine Feedline Resistance<br>(lb <sub>f</sub> -sec <sup>2</sup> /lb <sub>m</sub> -ft <sup>5</sup> )     | 857.7   |
| Oxidizer System Feedline Resistance<br>(lb <sub>f</sub> -sec <sup>2</sup> /lb <sub>m</sub> -ft <sup>5</sup> ) | 97.72   |
| Fuel System Feedline Resistance<br>(lb <sub>f</sub> -sec <sup>2</sup> /lb <sub>m</sub> -ft <sup>5</sup> )     | 36.02   |

Characteristic Equation for C\*:

$$C^* = C^*_{S.C.} + 870.5 (MR - 1.6) - 273.83 (MR^2 - 2.56) - 0.31878 (P_C - 99) + 12.953 (TP - 70) - 0.07414 (TP^2 - 4900) - 5.466 (MR \cdot TP - 112) + 0.03119 (MR \cdot TP^2 - 7840.); \text{ where } C^*_{S.C.} (\text{Engine No. 59}) = 5986.6 \text{ ft/sec}$$

Characteristic Equation for I<sub>SP</sub>:

$$I_{SP} = I_{SP_{vac}} - 96.954 (1.6 - MR) - 0.0487 (99 - P_C) - 0.06276 (70 - TP) + 30.409 (2.54 - MR^2) + 0.0004483 (4900 - TP^2); \text{ where } I_{SP_{vac}} (\text{Engine No. 59}) = 312.4 \text{ lb}_f\text{-sec/lb}_m$$

TABLE 2

## SPS DUTY CYCLE

|       | <u>FS1</u> <sup>(1)</sup>   | <u>FS2</u> <sup>(1)</sup>    | <u>BURN DURATION</u> |
|-------|-----------------------------|------------------------------|----------------------|
| SPS 1 | 26:24:55.659                | 26:25:05.019                 | 9.360                |
| SPS 2 | 28:00:56.469                | 28:01:04.228                 | 7.759                |
| SPS 3 | 75:48:00.274 <sup>(2)</sup> | 75:48:09.270                 | 8.996                |
| SPS 4 | 120:43:00.410               | 120:43:00.920                | 0.510                |
| SPS 5 | 165:00:00.420               | 165:01:07.367 <sup>(3)</sup> | 66.947               |
| SPS 6 | 210:08:00.410               | 210:08:00.910                | 0.500                |
| SPS 7 | 239:06:11.966               | 239:06:20.186                | 8.220                |
| SPS 8 | 259:39:16.360               | 259:39:28.150                | 11.790               |

(1) Times from Command Module Computer Downlink Data - CG000IV

(2) SCS Mode - Times are from solenoid drivers - CH3604X

(3) MTVC mode after 30 seconds - Time from solenoid drivers - CH3604X

TABLE 3

## FLIGHT DATA USED IN STEADY STATE ANALYSIS

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

TABLE 4

## SERVICE PROPULSION SYSTEM PERFORMANCE SUMMARY FIFTH SPS BURN

| INSTRUMENTED PARAMETERS              | PREDICTED<br>FLIGHT <sup>(1)</sup> | MEASURED | ACTUAL<br>FLIGHT <sup>(2)</sup> |
|--------------------------------------|------------------------------------|----------|---------------------------------|
| OXIDIZER TANK PRESSURE<br>PSIA       | 177.3                              | 177.6    | 177.9                           |
| FUEL TANK PRESSURE<br>PSIA           | 175.4                              | 175.0    | 175.7                           |
| OXIDIZER INTERFACE PRESSURE<br>PSIA  | 165.7                              | 163.0    | 166.2                           |
| FUEL INTERFACE PRESSURE<br>PSIA      | 173.2                              | 170.6    | 173.5                           |
| ENGINE CHAMBER PRESSURE<br>PSIA      | 103.0                              | 102.5    | 102.6                           |
| CALCULATED PERFORMANCE<br>PARAMETERS |                                    |          |                                 |
| OXIDIZER FLOW RATE-lbm/sec           | 41.53                              | ---      | 41.56                           |
| FUEL FLOW RATE-lbm/sec               | 25.84                              | ---      | 25.90                           |
| PROPELLANT MIXTURE RATIO             | 1.607                              | ---      | 1.605                           |
| VACUUM SPECIFIC-IMPULSE              | 312.5                              | ---      | 314.0                           |
| VACUUM THRUST-lbf                    | 21058                              | ---      | 21180                           |

(1) "Apollo 7 (Mission C/AS-205/CSM 101) Service Propulsion Final Preflight Performance Analysis," EP Memorandum to distribution, 16 September 1968.

(2) Calculated by Propulsion Analysis Program

TABLE 5

## SUMMARY OF STEADY STATE PRESSURE DATA

| MEASUREMENT            | BURN  |       |       |       |       |       |  |
|------------------------|-------|-------|-------|-------|-------|-------|--|
|                        | 1     | 2     | 3     | 5     | 7     | 8     |  |
| SP0003P Oxid Tank      | 175.7 | 176.0 | 176.0 | 177.0 | 175.0 | 175.0 |  |
| SP0931P Oxid Interface | 161.2 | 162.8 | 163.5 | 163.0 | 163.5 | 161.3 |  |
| SP0006P Fuel Tank      | 173.0 | 173.0 | 172   | 174.2 | 172   | 171.5 |  |
| SP0930P Fuel Interface | 168.5 | 169.5 | 168.5 | 170.1 | 167.5 | 167.5 |  |
| SP0661P Chamber        | 99.0  | 101.0 | 100.0 | 102.5 | 101.0 | 100.0 |  |

NOTE: All values in psia.

TABLE 6

## SERVICE PROPULSION SYSTEM FLIGHT SUMMARY OF TRANSIENT DATA

|                                                                                                                | Apollo 7<br>first<br>firing | Apollo 7<br>second<br>firing | Apollo 7<br>third<br>firing | Apollo 7<br>fifth<br>firing | Apollo 7<br>seventh<br>firing | Apollo 7<br>eighth<br>firing | Apollo 7<br>SPS engine<br>acceptance<br>test | Apollo 6<br>first<br>firing | Specifica-<br>tion<br>values |
|----------------------------------------------------------------------------------------------------------------|-----------------------------|------------------------------|-----------------------------|-----------------------------|-------------------------------|------------------------------|----------------------------------------------|-----------------------------|------------------------------|
| Start transient total<br>vacuum impulse from<br>ignition to 90 per-<br>cent of steady-state<br>thrust, lbf-sec | 112                         | 121                          | 233                         | 340                         | 1,114                         | 407                          | 226                                          | 225                         | 100-600                      |
| Time from ignition to<br>90 percent of steady-<br>state thrust, sec                                            | 0.5                         | 0.461                        | 0.414                       | 0.449                       | 0.509                         | 0.504                        | 0.532                                        | 0.420                       | 0.450-0.650                  |
| Time from ignition to<br>steady-state thrust,<br>sec                                                           | 1.35                        | 1.34                         | 1.52                        | 1.48                        | 1.66                          | 1.88                         | -                                            | -                           | -                            |
| Chamber pressure over-<br>shoot during start,<br>percent (based on<br>nominal 100 psia<br>chamber pressure)    | 43                          | 50*                          | 20                          | 24                          | 20                            | 19                           | -                                            | 56                          | -                            |
| Chamber pressure over-<br>shoot duration, sec                                                                  | 0.9                         | 1.0                          | 1.2                         | 1.2                         | 1.2                           | 1.0                          | -                                            | 1.2                         | -                            |
| Shutdown transient<br>total vacuum impulse<br>from cutoff to<br>0 percent steady-<br>state thrust, lbf-sec     | 11,619                      | 11,983                       | 12,676                      | 12,692                      | 12,076                        | 12,411                       | 9,650                                        | 11,905                      | 8,000-13,000                 |
| Time from cutoff to<br>10 percent of steady-<br>state thrust, sec                                              | 0.946                       | 0.944                        | 0.984                       | 0.973                       | 0.994                         | 0.970                        | 0.938                                        | 0.920                       | 0.750-1.100                  |

\*Measurement Maximum Scale

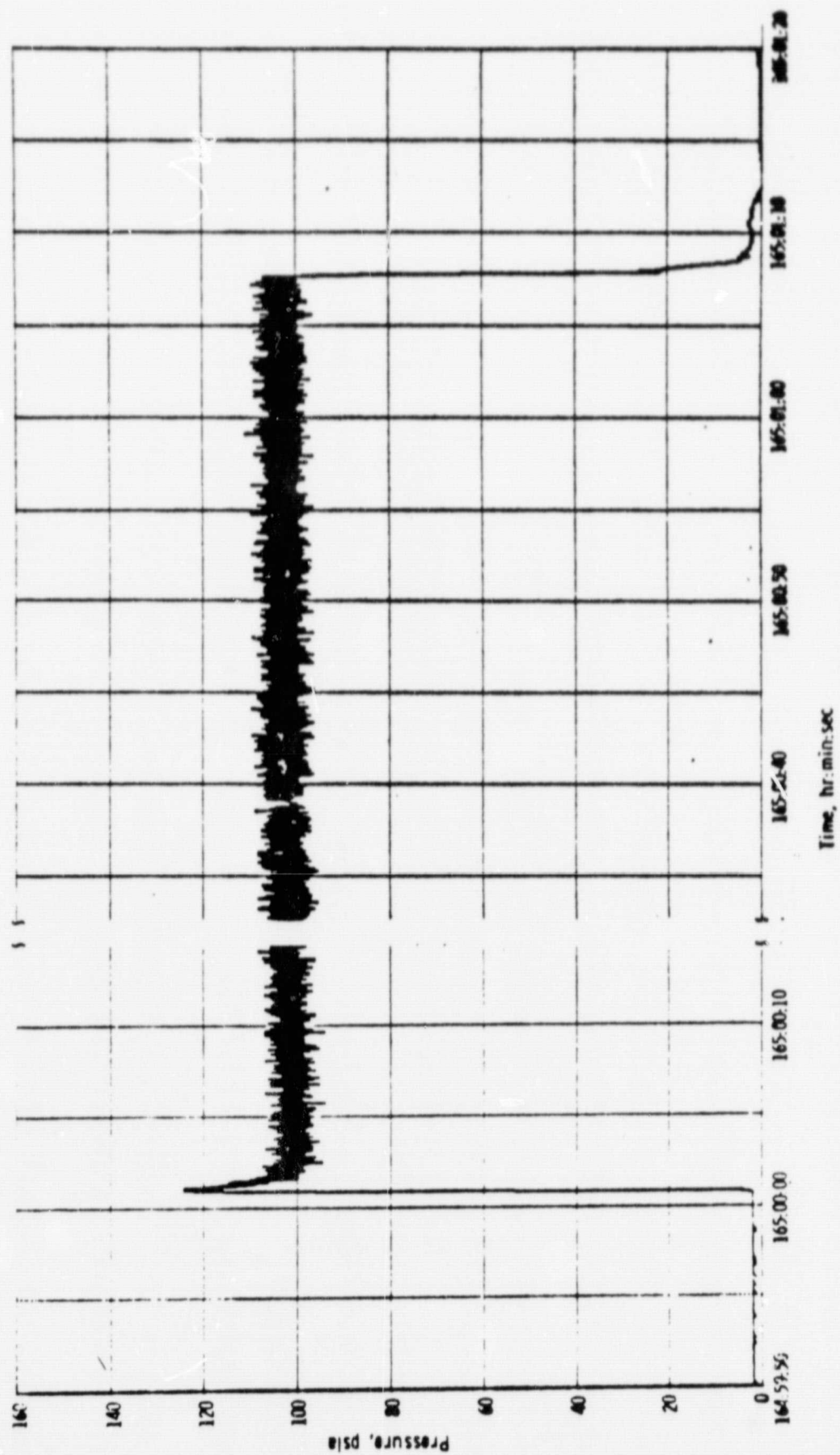
TABLE 7  
SERVICE PROPULSION SYSTEM FLIGHT SUMMARY OF MINIMUM IMPULSE DATA

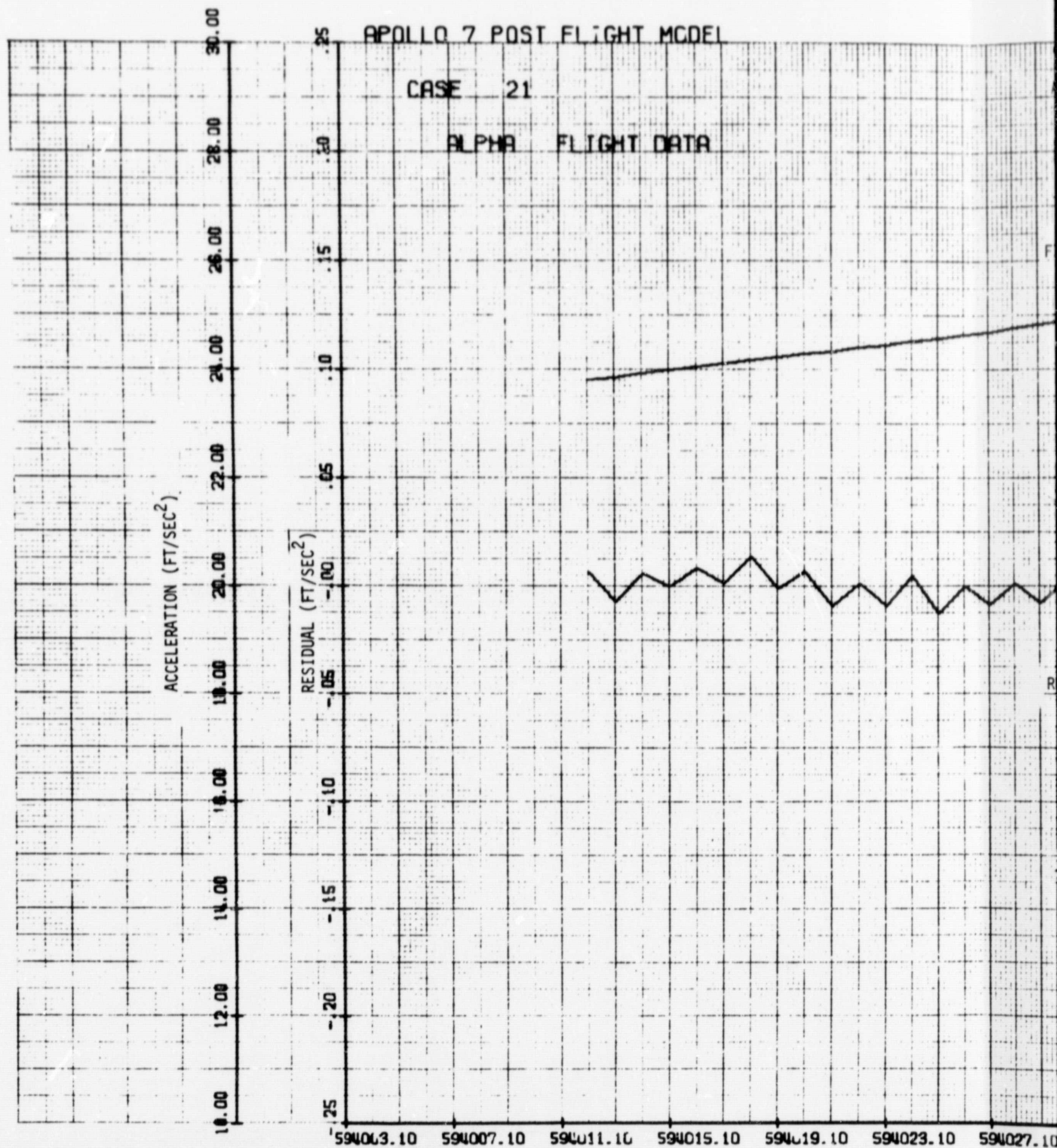
|                                                        | Apollo 7<br>fourth<br>firing | Apollo 7<br>sixth<br>firing |
|--------------------------------------------------------|------------------------------|-----------------------------|
| Time from ignition<br>signal to cutoff<br>signal (sec) | 0.5                          | 0.5                         |
| Time duration of<br>thrusting (sec)                    | 2.67                         | 2.82                        |
| Total impulse of*<br>firing, lbf-sec                   | 13,080                       | 11,907                      |
| Total impulse of**<br>firing, lbf-sec                  | 10,243                       | 10,188                      |

\*Determined from chamber pressure measurement.

\*\*Determined from accelerometer measurement.

FIGURE 1  
CHAMBER PRESSURE DURING FIFTH SERVICE PROPULSION MANEUVER

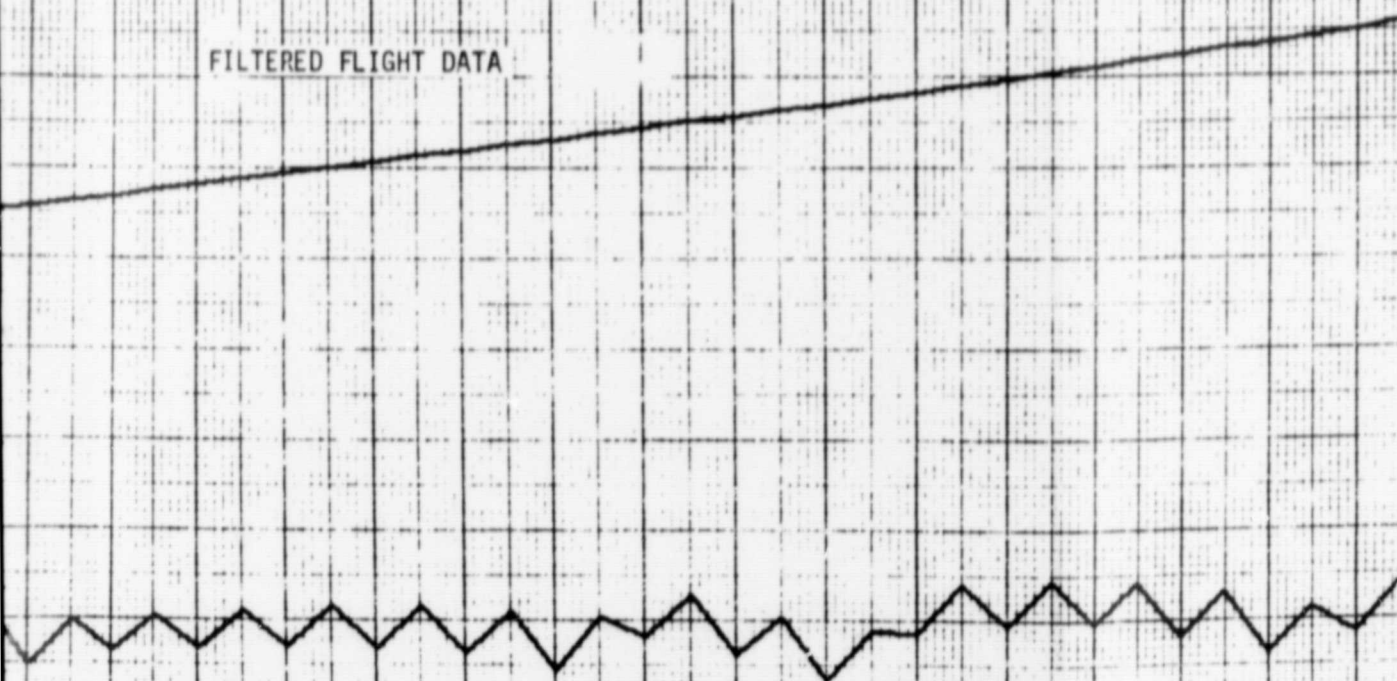




RECORDING CHARTS GRAPHIC CONTROLS CORPORATION BUFFALO NEW YORK

FIGURE 2  
ACCELERATION MATCH

INTERCEPT =  $-.00038$   
SLOPE =  $.000013$   
SUM  $Y^2 = .00245$   
PLOT NUMBER 1



RESIDUAL



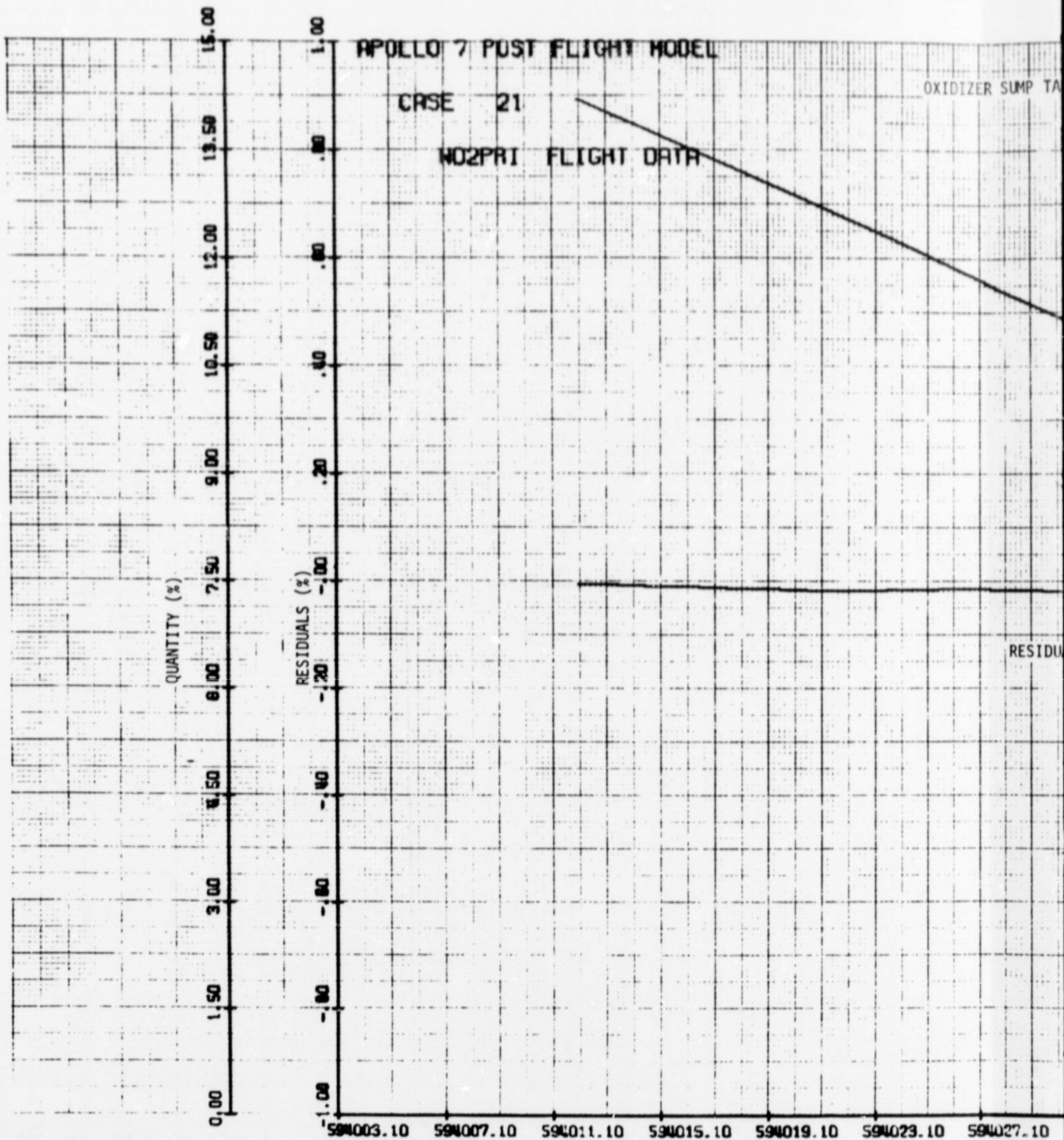
Page 31

10 594027.10 594031.10 594035.10 594039.10 594043.10 594047.10 594051.10 594055.10 594059.10 594063.10  
TIME (SECONDS)

PRINTED IN U.S.A.

NO. 02

FOLDOUT FRAME 2



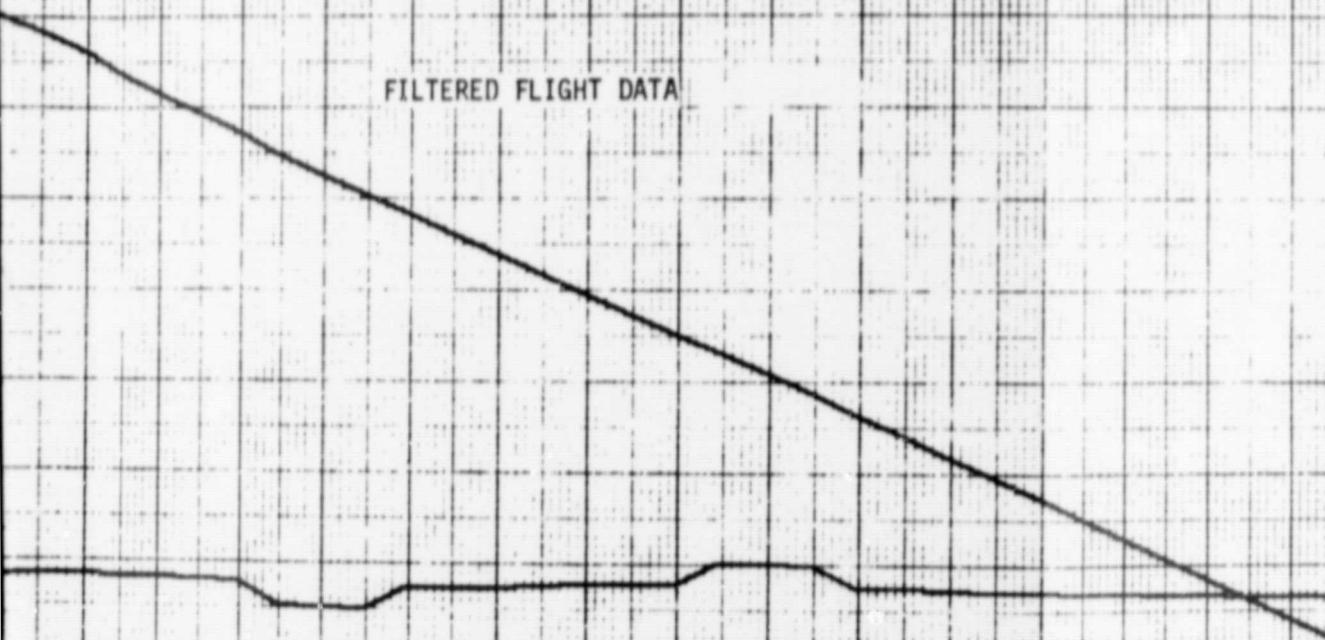
RECORDING CHARTS

FOLDOUT FRAME /

FIGURE 3

XIDIZER SUMP TANK PRIMARY GAGING QUANTITY MATCH

INTERCEPT= .02164  
 SLOPE= -.000434  
 SUM YR<sup>2</sup>= .00906  
 PLOT NUMBER 3



RESIDUAL

Page 32

594027.10 594031.10 594035.10 594039.10 594043.10 594047.10 594051.10 594055.10 594059.10 594063.10

TIME (SECONDS)

RECOR

FOLDOUT FRAME 2

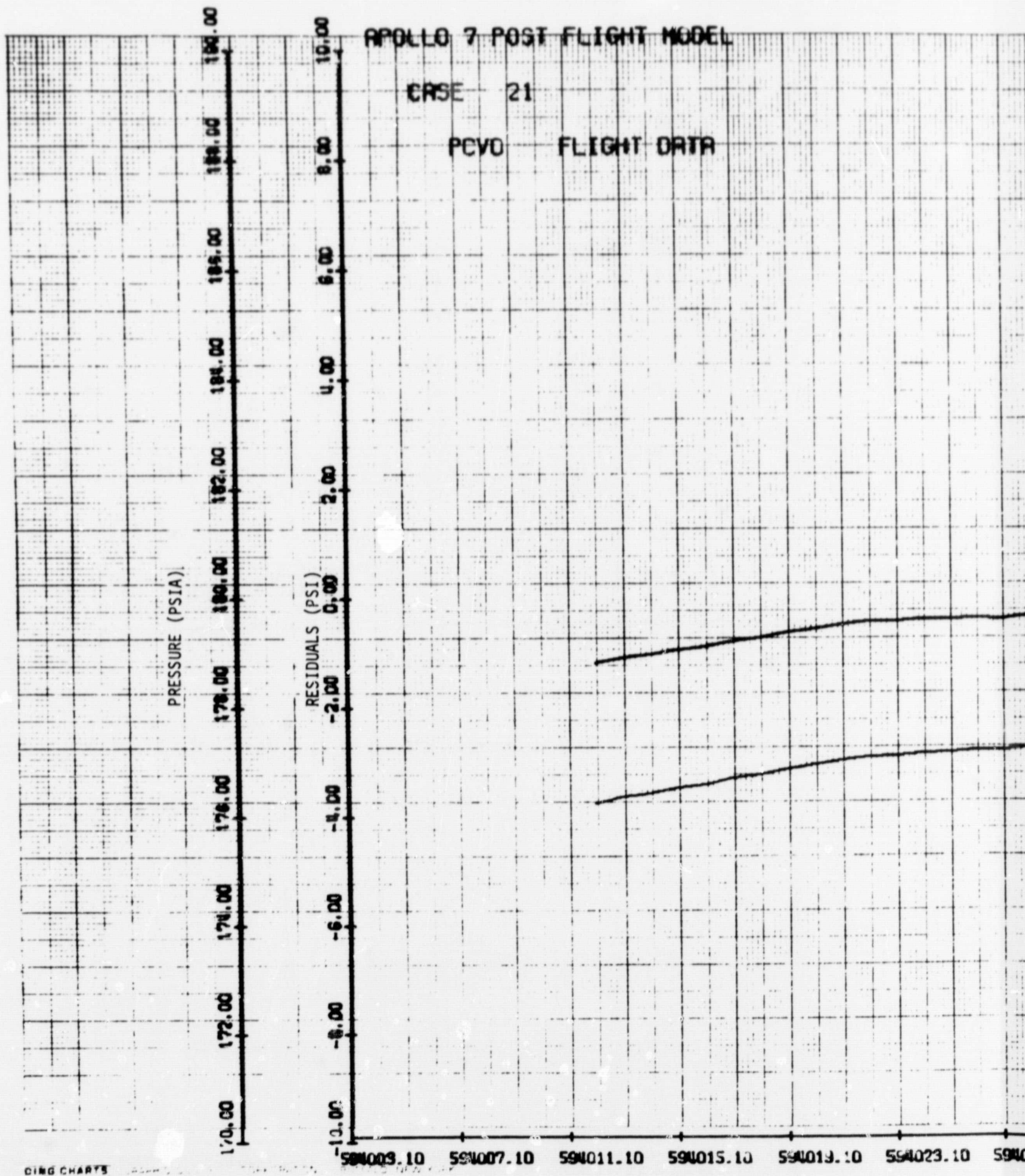


FIGURE 4  
OXIDIZER TANK PRESSURE MATCH

INTERCEPT=  $-.94009$   
SLOPE=  $.012009$   
SUM  $YR^2$ =  $16.66794$   
PLOT NUMBER 4

RESIDUAL

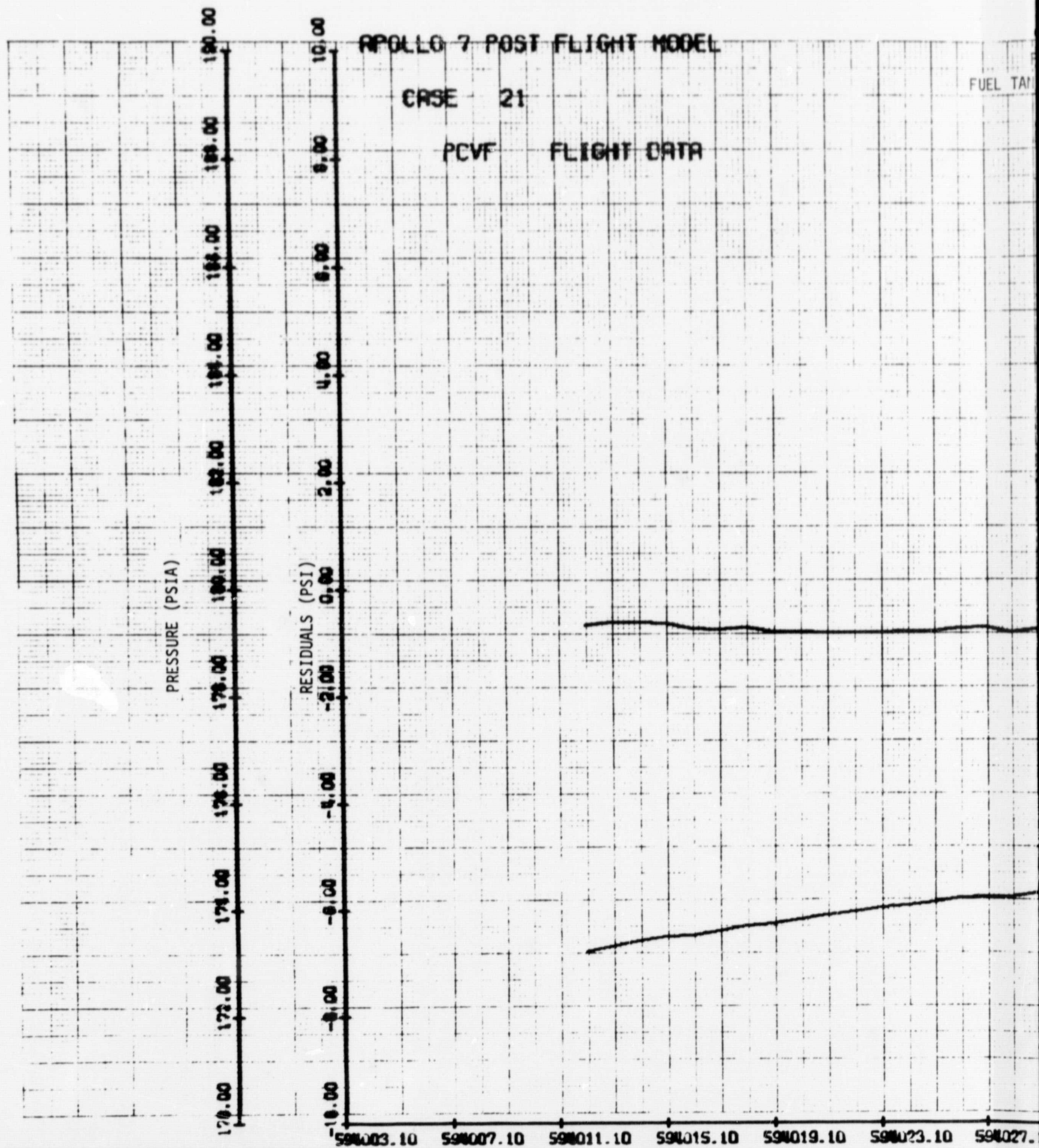
FILTERED FLIGHT DATA

Page 33

591027.10 591031.10 591035.10 591039.10 591043.10 591047.10 591051.10 591055.10 591059.10 591063.10  
TIME<sup>02</sup> (SECONDS)

RECORDING CHARTS GI

FOLDOUT FRAME 2



FOLDOUT FRAME

FIGURE 5  
FUEL TANK PRESSURE MATCH

INTERCEPT=  $-.69785$   
SLOPE=  $-.001751$   
SUM  $YR^2$ =  $25.65856$   
PLOT NUMBER 5

RESIDUAL

FILTERED FLIGHT DATA

591027.10 591031.10 591035.10 591039.10 591043.10 591047.10 591051.10 591055.10 591059.10 591063.10  
TIME (SECONDS)  
RECORDING CHART 8 GRAPHIC CONTROLS

FIGURE 6  
COMPARISON OF PREFLIGHT PREDICTED AND INFLIGHT PERFORMANCE

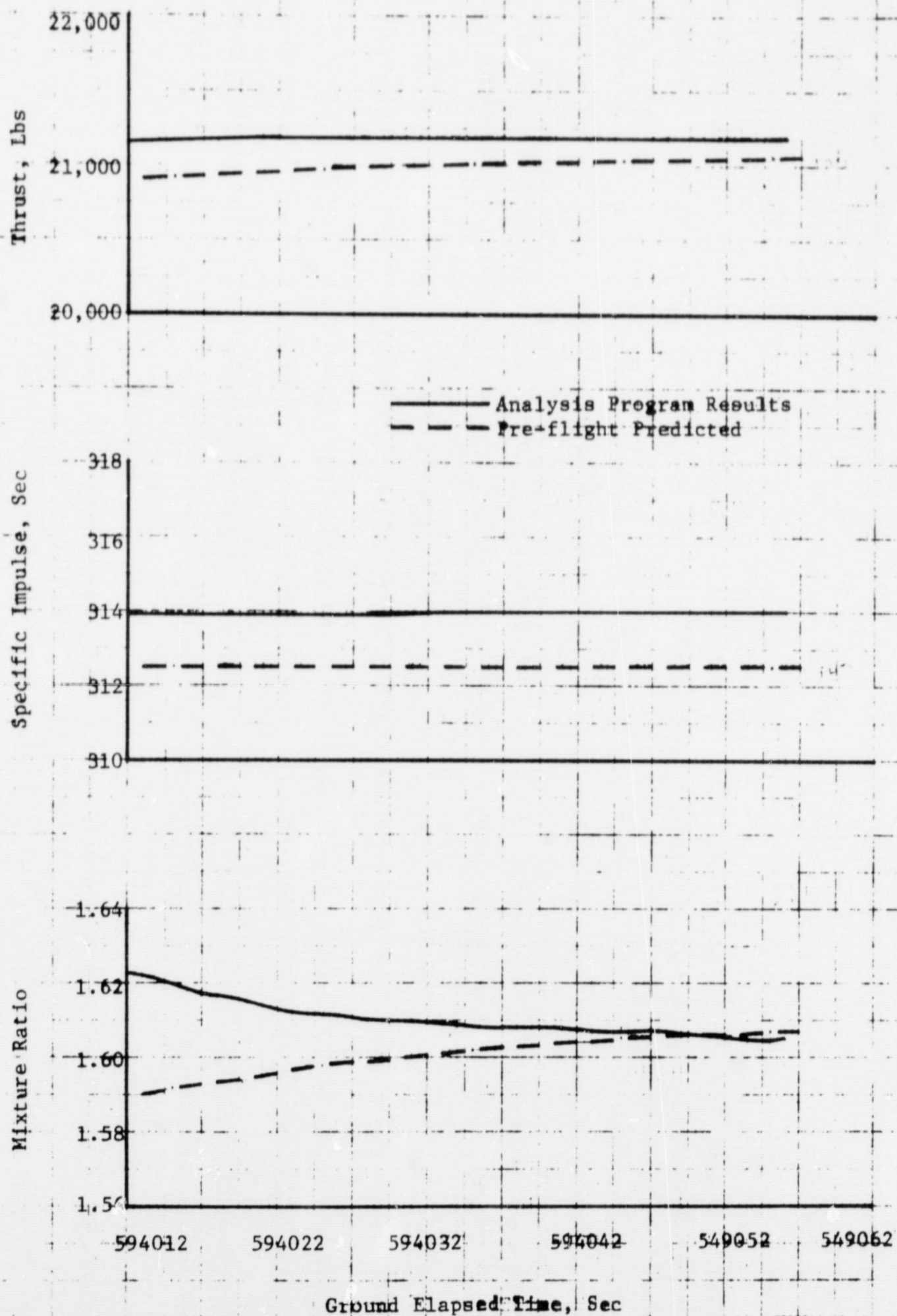
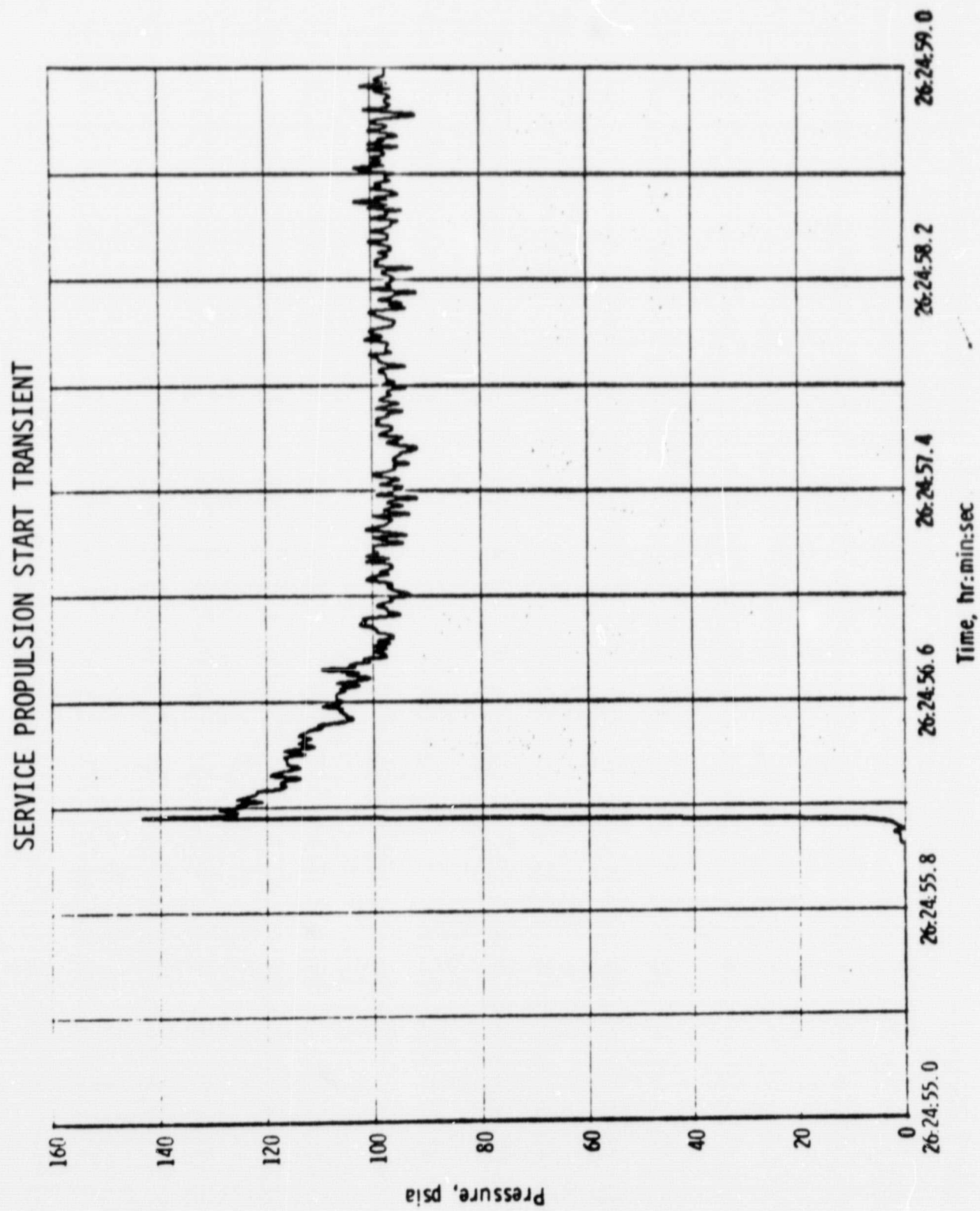
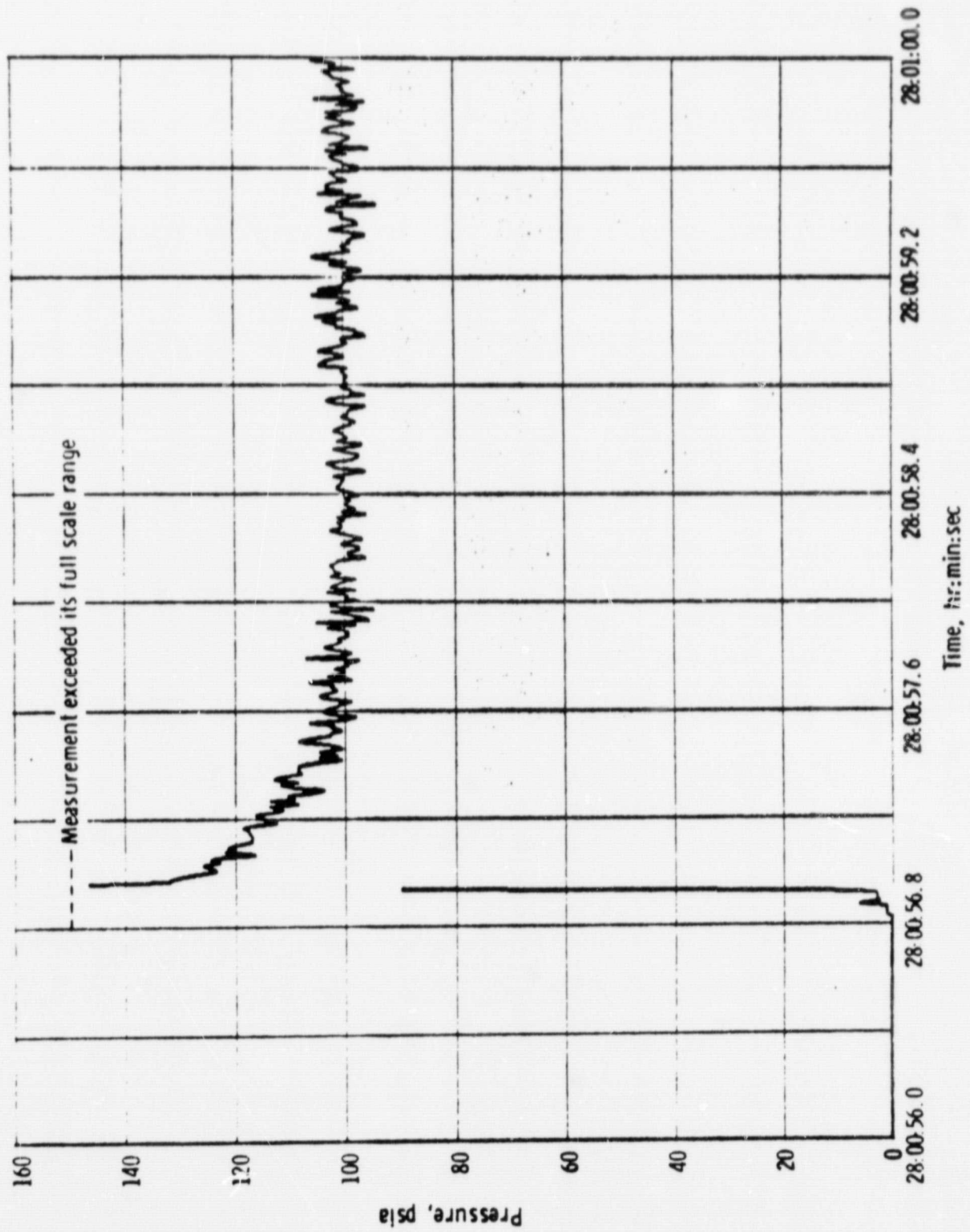


FIGURE 7



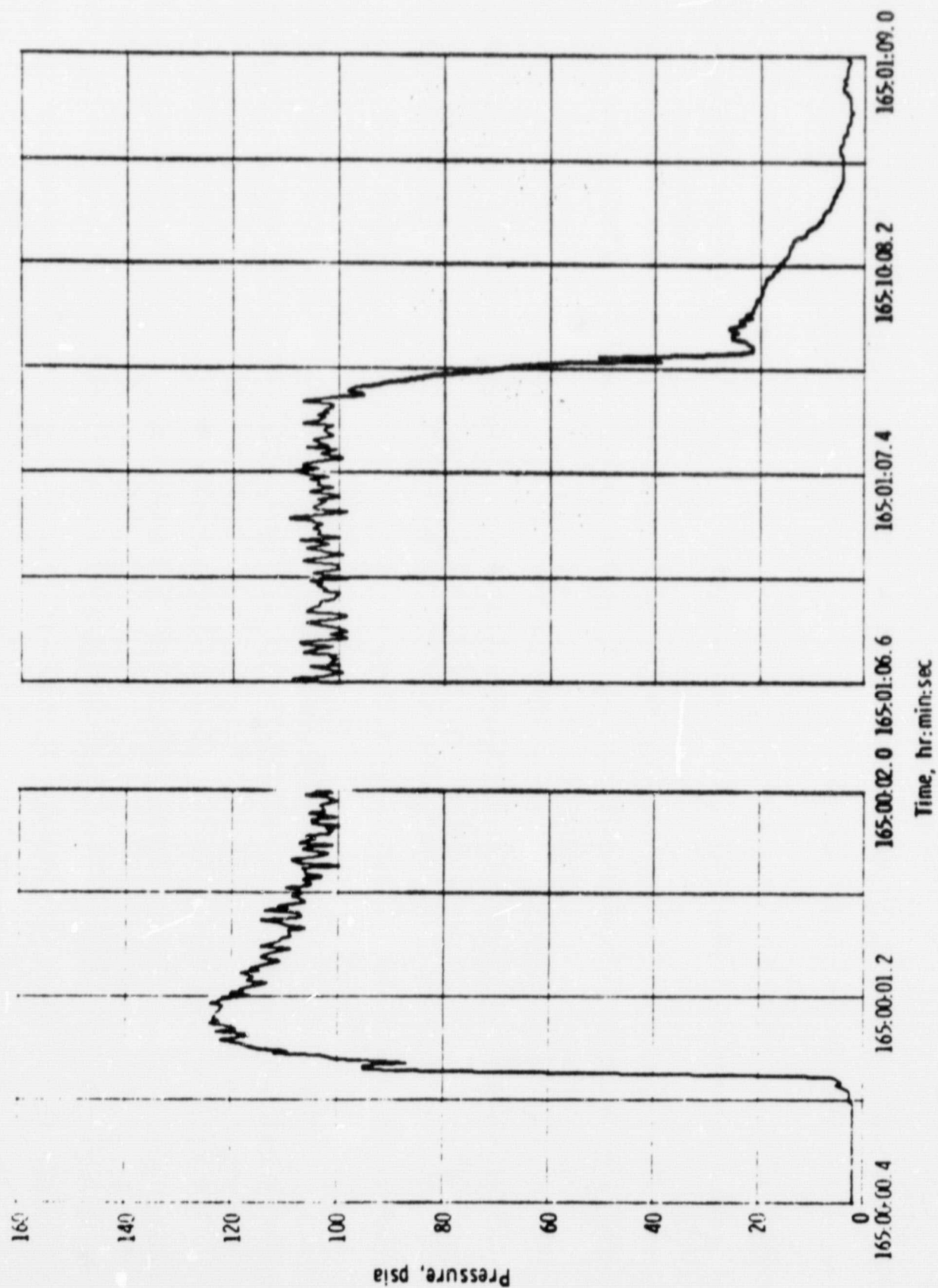
(a) First maneuver.

FIGURE 7  
CONCLUDED



(b) Second maneuver.

FIGURE 8  
START AND SHUTDOWN TRANSIENTS FOR FIFTH SERVICE PROPULSION MANEUVER



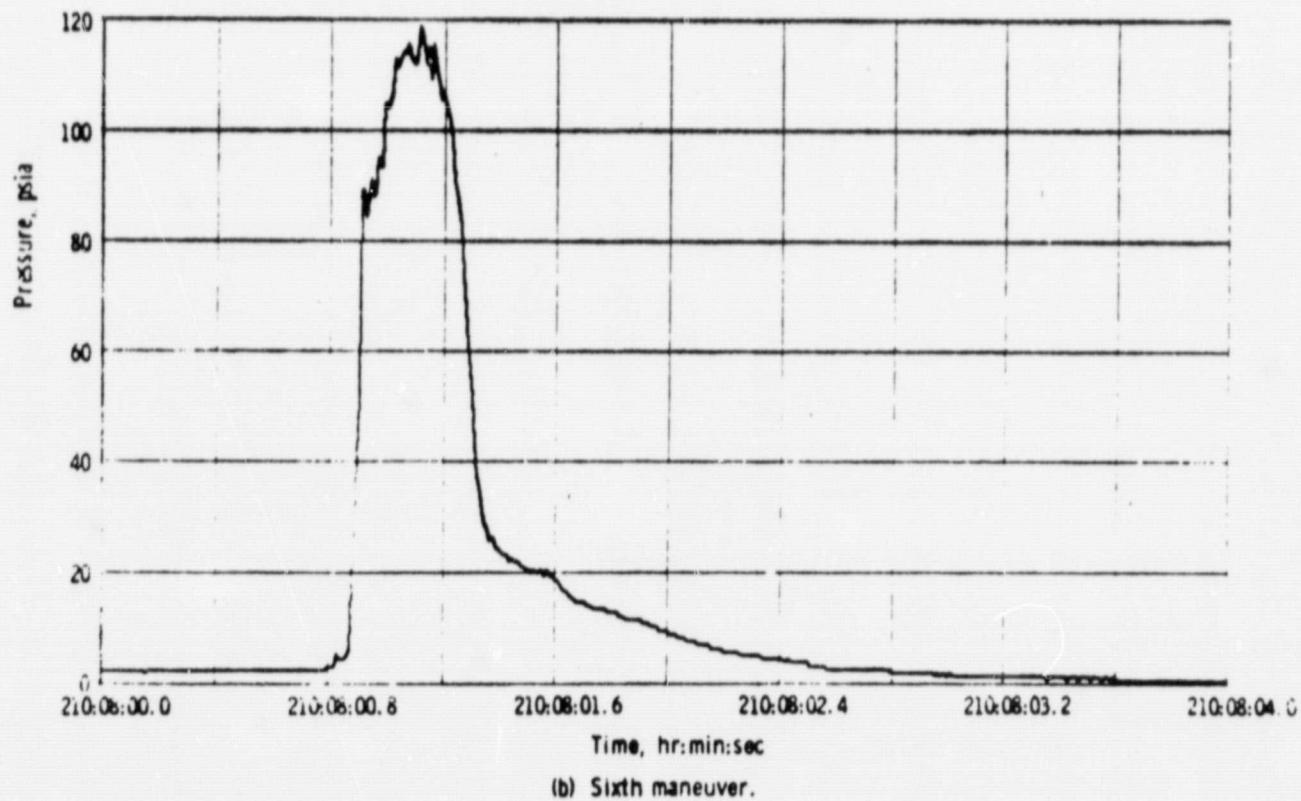
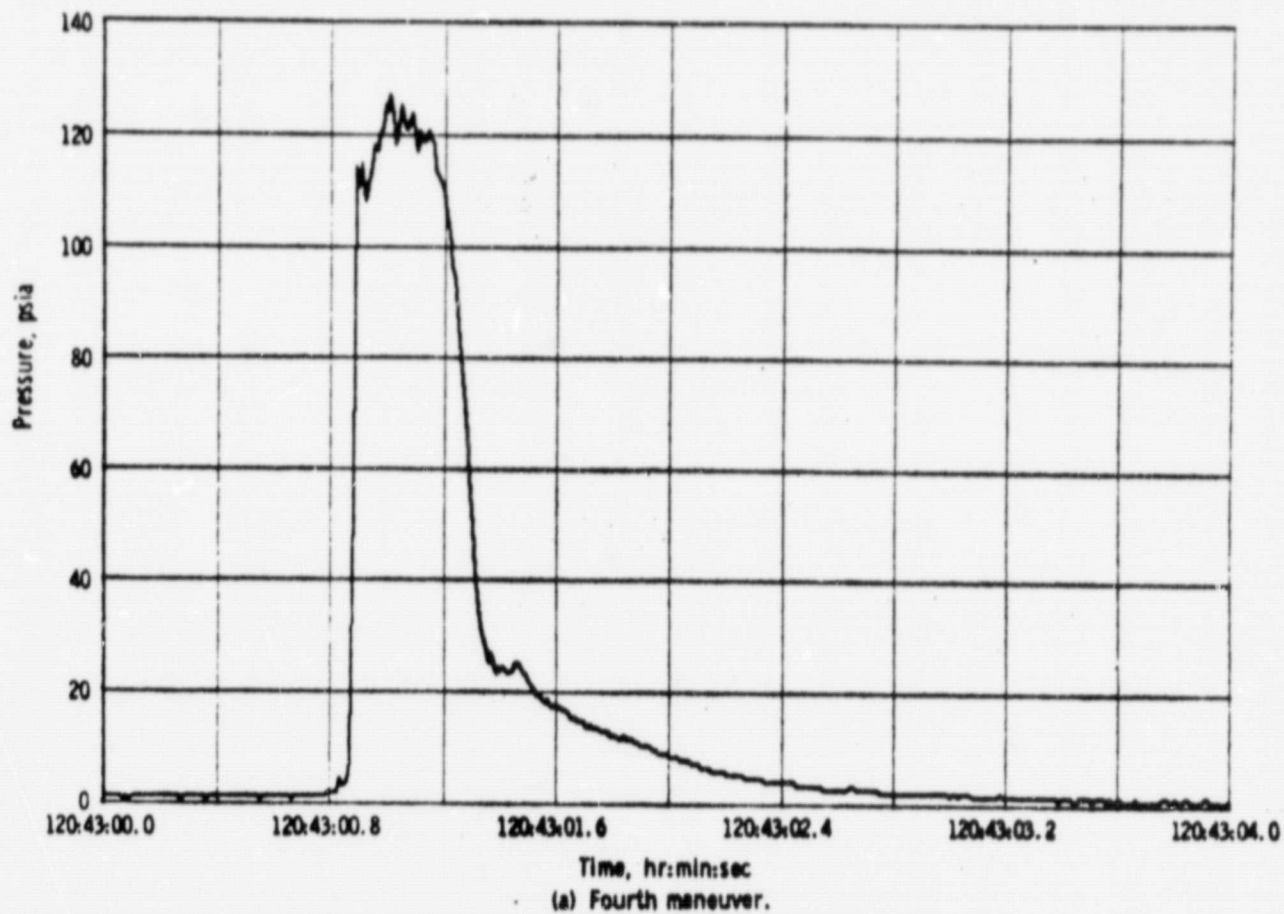
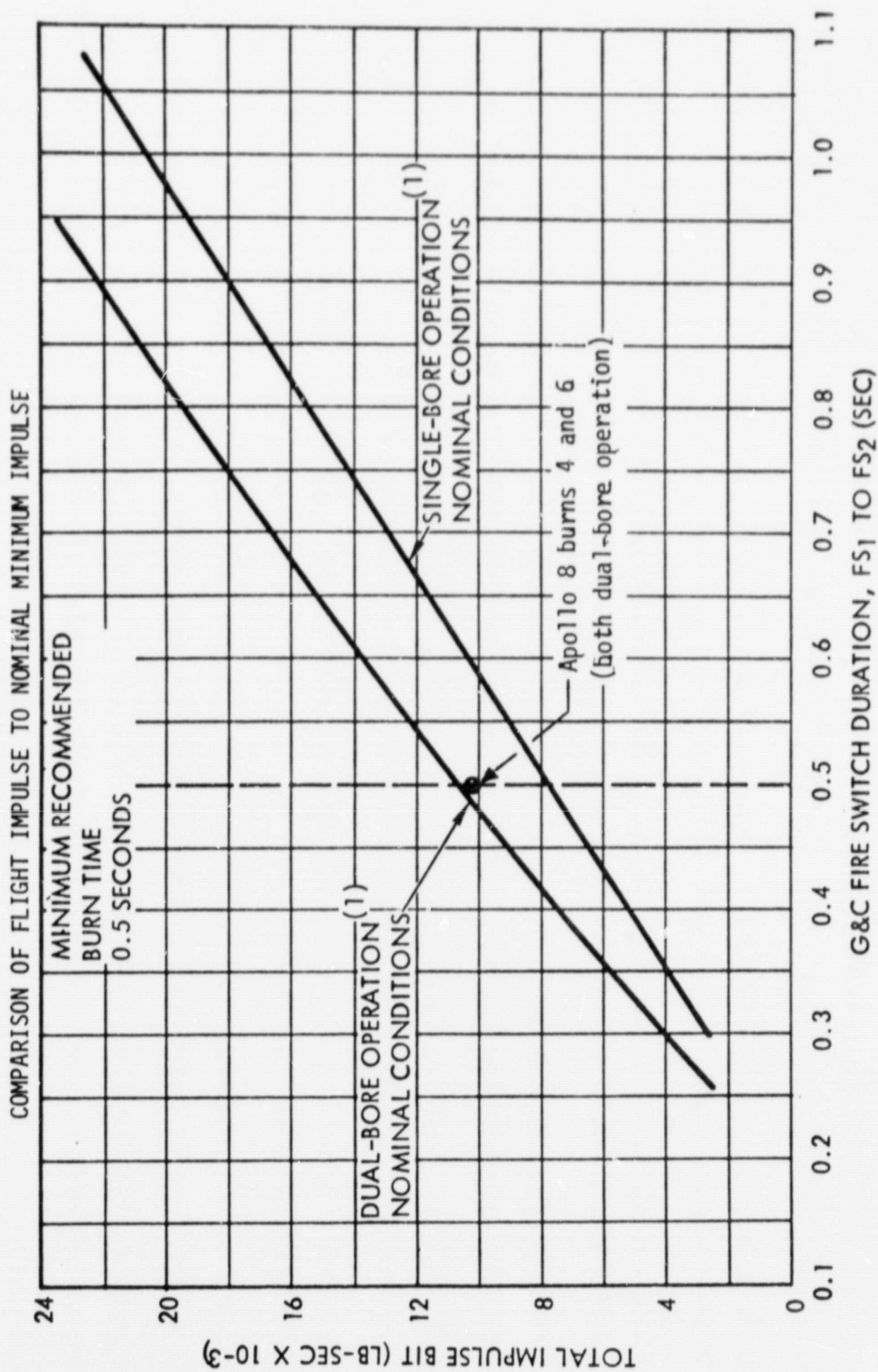


FIGURE 9  
CHAMBER PRESSURE DURING MINIMUM IMPULSE FIRINGS

FIGURE 10



(1) Nominal values from Reference 2

## REFERENCES

1. NASA Memorandum, "Apollo 7 (Mission C/AS-205/CSM 101) Service Propulsion System Final Preflight Performance Analysis," J. G. Thibodaux, Jr., dated 16 September 1968.
2. SNA-8-D-027 (I) Revision 1, "CSM/LM Spacecraft Operational Data Book, Volume I, CSM Data Book, Revision 1, Amendment 2-78," dated 4 February 1969.