MSC-00126 -
SUPPLEMENT 7

K

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

APOLLO 10 MISSION REPORT
SUPPLEMENT 7

DRA

DESCENT PROPULSION SYSTEM
FINAL FLIGHT EVALUATION

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
DECEMBER 1969

MSC-00126
Supplement 7

APOLLO 10 MISSION REPORT

SUPPLEMENT 7

DESCENT PROPULSION SYSTEM
FINAL FLIGHT EVALUATION

PREPARED BY

TRW Systems

APPROVED BY

James A. McDivitt
Manager, Apollo Spacecraft Program

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
December 1969

11176-H314-R0-00

PROJECT TECHNICAL REPORT

APOLLO 10

LM-4

DESCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

NAS 9-8166

8 August 1969

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

Prepared by
R. K. M. Seto
Propulsion Technology Section
Power Systems Department

NASA/MSC

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

Concurred by: ____________________
W. R. Hammock, Jr., Manager
Descent Propulsion Subsystem

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

TRW SYSTEMS

Approved by: P. H. Janak for
R. J. Smith, Manager
Task E-19E

Approved by: P. H. Janak
P. H. Janak, Head
Propulsion Technology Section

Approved by: D W Vernon
D. W. Vernon, Manager
Power Systems Department

TRW
SYSTEMS GROUP

CONTENTS

TABLES

ILLUSTRATIONS

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 10 Mission. This report documents additional analysis of the DPS. This report has been prepared as Supplement 7 to the Apollo 10 Mission Report (MSC-00126).

2. SUMMARY

The performance of the LM-4 Descent Propulsion System during the Apollo 10 Mission was evaluated and found to be satisfactory.

Because system data during the DOI maneuver was not recorded and due to the short length of burns, no detailed performance study using the Apollo Propulsion Analysis Program was possible. However, the preflight model was used with flight data to approximate the performance at representative times during the Phasing Burn. For minimum throttle operation, (13.1% of full thrust) thrust, specific impulse, and mixture ratio were calculated to be 1371 lbf, 296.5 seconds and 1.605, respectively. For FTP, the values were 9841 lbf, 304.0 seconds and 1.599. These values can be considered as representative only.

Instrumentation biases were determined on the regulator outlet pressure measurement (GQ3018P), the oxidizer interface pressure measurement and the chamber pressure measurement with values of +4.0, +7.5 and -0.8 to -1.6 psi, respectively.

The supercritical helium tank experienced an average pressure rise rate of 5.84 psi/hr during the coast period between launch and first DPS engine firing. This value was less than anticipated from ground tests.

Although the fuel quantity gages (Fu 1 and Fu 2) never read off scale (greater than the maximum 95 percent indication) as expected prior to the Phasing Burn, they did respond with propellant consumption and were within both the expected accuracy of 3.5% and the specification limits of 1.3% at the end of the burn. The oxidizer gages (Ox 1 and Ox 2) operated as expected prior to the phasing burn. However, at the end of the burn, it appeared that the Ox 2 gage was reading 1.6% higher than expected. This reading was still within the expected accuracy of 2.7%.

The engine start and shutdown transients compared very well with predicted values. The shutdown transient time, however, was 0.09 seconds greater than the specification limit of 0.25 seconds. The throttle response from 13.1% to FTP was acceptable.

3. INTRODUCTION

The Apollo 10 Mission was the tenth in a series of flights using specification Apollo hardware. It was the third flight test and the second manned flight of the Lunar Module (LM). The mission was the fourth manned flight of Block II Command and Service Module (CSM) and the third manned flight using a Saturn V launch vehicle.

The overall mission objective[1] was to duplicate, as closely as possible, a G type mission with the exception of lunar landing and liftoff. This included the performance of the Descent Orbit Insertion maneuver by the Descent Propulsion System (DPS) and the rendezvous maneuvers by the Ascent Propulsion System (APS). Also included as objectives were to verify LM operation in a lunar environment, verify mission support of all spacecrafts during all mission phases at lunar distances and to obtain more information about the lunar potential.

The space vehicle was launched from the Kennedy Space Center (KSC) at 12:49:00 P.M. (EST) on May 18, 1969. Following a normal launch phase, the S-IVB stage inserted the spacecraft into an orbit of 102.6 by 99.6 nautical miles. Two and a half hours after launch the S-IVB performed the trans-lunar injection maneuver. The CSM docked with the LM and the docked spacecrafts were ejected from the S-IVB approximately four hours after launch. During the next 76 hours, four SPS burns were performed. Undocking of the LM from the CSM in lunar orbit occurred 98.5 hours after launch. At approximately 100 hours, the first DPS maneuver, the Descent Orbit Insertion (DOI) burn was performed. The burn duration was 27.4 seconds and included operation at the minimum throttle setting and throttling to the 40% of full thrust

[1]Reference 2.

level. This burn put the LM into a lunar orbit of 61.2 by 8.4 nautical miles. At approximately 101 hours after launch, the DPS performed a Phasing Maneuver burn, 39.9 seconds in duration. The spacecraft was now in a lunar orbit of 190.1 by 11.0 nautical miles. The burn included operation at the minimum throttle setting and a short duration segment at the Fixed Throttle Position (FTP). The Phasing Maneuver ended the DPS mission duty cycle. The descent stage was separated from the ascent stage about two hours later. The APS performed two firings, the latter being to propellant depletion and the SPS performed one more burn during the subsequent portion of the mission.

The actual ignition and shutdown times for the two DPS firings are shown in Table 1.

The Apollo 10 Mission utilized LM-4 which was equipped with DPS engine S/N 1039. The engine and feed system characteristics are presented in Table 2.

Each DPS burn was prededed with a two jet +X LM Reaction Control System (RCS) ullage maneuver to settle propellants.

There was one Apollo 10 Mission Detailed Test Objective (DTO) specifically related to the DPS.

P13.14 LM Supercritical Helium.

The functional test objective of this DTO was:

1) Obtain data on DPS supercritical helium pressure profile during standby and during DPS DOI and phasing burns.

The detailed requirements of this objective are described in Reference 3.

4. PERFORMANCE ANALYSIS

Due to the insufficient duration of the two DPS maneuvers performed during the Apollo 10 Mission, a meaningful detailed analysis using the Apollo Performance Analysis Program could not be made. Analysis was further hampered by the loss of the DOI burn data. The burn was performed behind the moon and the CSM failed to record the LM data.

Upon activating the ambient helium start bottle in preparation for the DOI burn, DPS pressures appeared nominal with the exception of the oxidizer interface pressure measurement (GQ 4111P) and the redundant helium regulator outlet pressure measurement (GQ 3018P). At this time, the regulator outlet pressure (GQ 3025P) and fuel interface pressure (GQ 3611P) were approximately equal at 251 and 250 psia, respectively. The oxidizer interface pressure was 241 psia while the redundant regulator outlet pressure (GQ 3018P) was 247 psia. The chamber pressure measurement had a bias of from 0.8 to 1.6 psia prior to the burn. Simulation of an FTP time slice during the phasing burn indicated that the oxidizer pressure transducer must have incurred a downward shift. Had the interface pressure been as measured, the mean chamber pressure (with bias included) would have been more than one psi lower than observed. It was concluded that at FTP, the interface pressures were essentially equal and that there was a bias of approximately 7.5 psi on the oxidizer interface transducer. Similar reasoning,and the fact that the regulator outlet pressure as measured by GQ 3025P matched the predicted value during the burn, indicated that the regulator outlet pressure measurement (GQ 3018P) was biased low by approximately 4 psi. Table 3 presents the flight measurements for the Descent Propulsion System.

Descent Orbit Insertion Burn

Although the data for the DOI Burn was not recorded, indications are that the DPS performed satisfactorily. Prior to and after the maneuver, the system pressures appeared nominal. Astronaut reports of the burn indicated normal operation. The burn was initiated at the minimum throttle setting of 13.1% of full thrust. After approximately 14 seconds, the engine was to be manually throttled to the 40% level for the remainder of the maneuver. The length of the burn was reported to have been approximately 27.4 seconds with a measured velocity change of 70.66 ft/sec. The actual velocity gain target was 71.25 ft/sec. The preflight performance predicted burn time was 28.0 seconds with a simulated velocity change of 71.6 ft/sec. There are three primary reasons for the difference between predicted and actual burn times: 1) differences in velocity gain, 2) simulated minimum throttle setting, and 3) simulation of the throttling transient from 13.1% to 40%. The preflight assumed minimum throttle position was 11.3% while the actual inflight setting was 13.1%. In simulating the burn, a step change between throttle settings was assumed while the actual maneuver requires approximately one second. If these differences are accounted for, it appears that the predicted and the actual burn time would differ by less than 0.1 seconds. Other uncertainties about the burn include actual start transient, time of throttling to 40%, actual throttle position after throttling (since the maneuver was performed manually) and spacecraft weight errors. In view of the above, it was concluded that the performance was nominal.

The attainment of the target velocity gain is extremely critical to the descent trajectory. The small residual (difference between target and actual) of 0.6 ft/sec was easily nulled by use of the LM-RCS.

Phasing Burn

The Phasing Burn was performed satisfactorily. The burn was initiated at the minimum throttle setting. After 26 seconds the engine was automatically throttled to the Fixed Throttle Position (FTP) for the remainder of the maneuver. System pressures appeared nominal during and after the burn. The actual burn time was 39.94 seconds with velocity gain of 175.8 ft/sec, while the predicted burn time was 40.3 seconds for a velocity gain of 174.5 ft/sec. The actual target velocity gain was 176.9 ft/sec. As with the DOI Burn, the difference in predicted and actual velocity gain and time can be essentially accounted for by the difference in the simulated and inflight throttling transients, minimum throttle setting and start transient.

Table 4 presents the inflight measured data at typical points during each of the two throttle positions experienced in the Phasing Burn. The preflight predicted values, obtained from Reference 5, are also presented for comparison. The inflight measured data compares well with preflight predicted data. Deviations at the minimum throttle setting (FS-1 + 10 seconds) are due to the difference between flight and predicted throttle setting. Although detailed performance analysis could not be made, the flight data was used in the prediction model to give an indication of approximate inflight performance. The results are also presented in Table 4. Figure 1 through 9 present DPS inflight measured supercritical helium tank pressure, regulator outlet pressure, interface pressures, chamber pressure and gaging system readings during the Phasing Burn.

5. PRESSURIZATION SYSTEM EVALUATION

The performance of the pressurization system was considered satisfactory.

The ambient start bottle was loaded with approximately 1.1 lbm of helium at a pressure of 1619 psia at approximately 72.5°F. At launch, the pressure was approximately 1612 psia. Five days prior to launch, the oxidizer and fuel tank pressures were increased from their load pressures to 186.2 and 193.3 psia, respectively. At launch, the propellant tank pressures had decreased to approximately 168 and 188 psia, respectively. Approximately 30 hours prior to launch, the supercritical helium (SHe) tank fill procedures were completed with approximately 48 lbm of helium loaded at a pressure of about 95 psia. At launch, the pressure had risen to approximately 316 psia. The SHe tank pressure increase during this period was approximately 7.65 psi/hr due to normal heat leak into the system from the surrounding environment. During the 119 hour countdown demonstration test, the pressure rise rate was 7.31 psi/hr.

At 97.5 hours after launch, prior to pre-burn propellant tanks pressurization, the ambient helium bottle pressure was 1577 psia, the SHe tank pressure was 885 psia, the oxidizer tank pressure was 97 psia and the fuel tank pressure was 152 psia. The pressure decay in the propellant tanks was attributed to helium going into solution (Reference 6). The decay in the ambient start bottle pressure was greater than expected when only temperature effects are considered. In the case of Apollo 9/LM-3, the start bottle pressure showed little decay during the four days prior to launch. Indications were that the temperature in the bay where the start bottle is located, prior to the first DPS burn, were similar to LM-3 (which showed little start bottle pressure decay from launch to burn). It is, therefore, possible that there was a small helium leak which could have been caused by launch vibrations. An accurate analysis could not be made due to pressure measurement inaccuracies and the lack of system temperature measurement. Upon activation of the ambient start bottle, the pressures increased to 248.5[1] and 249 psia in the oxidizer and fuel tank, respectively. Thus, although there may have been a helium leak, the ambient start bottle performed as expected and caused no anomalies in propellant pressurization. The average SHe tank pressure rise, from launch was approximately 5.84 psi/hr. This flight pressure rise rate was somewhat less than anticipated based on ground tests. Similar reductions of inflight pressure rise rate was experienced on LM-3. Because of a known helium leak observed in the SHe system after the first DPS burn, however, it was not clear whether the reduced rise rate was due to zero-g coast conditions or the existence of the leak prior to the first burn. Based on the similar pressure rise rate experienced during the Apollo 10 Mission, it appears that the LM-3 pressure rise was normal and that the leak occurred after system activation prior to the first burn. In view of the above, the flight pressure rise rate to to be used for system predictions is being revised.

From the available flight data, it appears that the SHe system operated normally during both DPS burns.

[1] Includes apparent 7.5 psi bias as discussed in Performance Analysis section.

6. PQGS EVALUATION AND PROPELLANT LOADING

Propellant Quantity Gaging System

At engine ignition for the second DPS burn, the oxidizer propellant gages (Ox 1 and Ox 2) were reading off scale, as expected (greater than the maximum 95 percent indication). The fuel tank probes (Fu 1 and Fu 2) had readings of 94.2 and 94.5 percent, respectively. Based on the best estimate of consumed propellant during the DOI maneuver, the fuel tank measurements should also have been reading off scale at ignition. This deviation was also noted prior to launch. After ignition, the fuel quantities remained relatively constant for approximately 31 and 27 seconds for Fu 1 and Fu 2, respectively, at which time propellant consumption was indicated. The oxidizer gages began to show consumption at approximately 35 and 37 seconds for Ox 1 and Ox 2, respectively. At the end of the burn, the propellant gages were reading 92.4, 92.0, 93.8 and 94.5 percent for Fu 1, Fu 2, Ox 1 and Ox 2, respectively. Table 5 presents a comparison of the measured data and the best estimate of the actual values at the end of the Phasing Burn. Although the Ox 2 gage is outside the specification limits of 1.3%, it should be noted that the lack of data from the DOI burn somewhat compromises the calculated values. Although initially giving erroneous output, the fuel gages appeared to be functioning within specification limits at engine shutdown. All values were within the expected accuracties of 2.7% and 3.5% for oxidizer and fuel (Reference 7). These accuracies were developed from recent tests conducted at the White Sands Test Facility (WSTF).

The failure of the fuel gages to reach a maximum reading when greater than that amount of propellant was in the tanks has been attributed to either chemical reaction with alodine or aluminum impurities with the fuel, or contamination of the fuel sensors due to the referee propellant (used instead of live propellants in probe manufacture and calibration) or alodine surface treatment (Reference 8). A chemical reaction between the fuel and impurities, which are not clearly understood, could cause in insulating barrier to be set up such that the conductance within the sensing portion of the gaging system probe is reduced, thus causing a reduction in the full scale reading. This barrier could be in the form of bubbles forming on the inner electrode when the sensor is submersed in stagnant fuel. A small quantity of residual from the referee propellant or from the alodine surface treatment of the gage (prior to installation) could combine with the propellant and form a conductive component in the fuel that settles in the reference region at the bottom of the gaging probe causing the signal to be low at gage activation. For either of these to happen, the propellant would have to be in a stagnant condition. It was thus concluded in Reference 8 that under zero gravity conditions, these problems should not occur, particularly due to RCS and SPS activities which would tend to keep the propellant reasonably active inside the tanks. In the case of this flight, it is possible that the propellant movement prior to engine burn was not great enough to remove contamination from the reference region.

Propellant Loading[1]

Prior to propellant loading a density determination was made for the oxidizer and fuel. The analysis yielded an oxidizer density of 90.22 lbm/ft^3 and a fuel density of 56.44 lbm/ft^3 at a pressure of 240 psia and a temper-

[1]Reference 9

ature of 70° F. The oxidizer and fuel were loaded to their planned overfill quantities of 11400.4 lbm and 7136.7 lbm, respectively. Off-loading was planned such that the target loads of 11209.4 lbm of oxidizer and 7054.8 lbm of fuel would be obtained. During this procedure, however, 45.3 lbm more fuel was off-loaded than planned. The actual propellant loads at launch were 11209.2 lbm of oxidizer and 7009.5 lbm of fuel.

7. ENGINE TRANSIENT ANALYSIS

The mission duty cycle of the DPS during Apollo 10 included two starts at the minimum throttle setting, one shutdown at approximately 40% throttle and one shutdown at FTP. During the DOI Burn the engine was manually throttled to 40% throttle and during the Phasing Burn the engine was automatically throttled to FTP.

Due to data loss during the DOI Burn, only the transients for the Phasing Burn were analyzed. The transients for this burn were considered satisfactory since they compared well with predicted values. It should be noted, however, that the shutdown transient time was greater than the specification limit by approximately 0.09 seconds.

Phasing Burn Start and Shutdown Transients

In determining the time of engine fire switch signals (FS-1 and FS-2), the technique as developed in Reference 10 was used. This method, developed from White Sands Test Facility (WSTF) test data, assumes that approximately 0.030 seconds after the engine start command (FS-1), an oscillation in the fuel interface pressure occurs. Similarly, 0.092 seconds after the engine shutdown signal (FS-2) another oscillation in the fuel interface pressure occurs. Thus, start and shutdown oscillations of the fuel interface pressure were noted and the appropriate time lead applied.

The ignition delay from FS-1 to first rise in chamber pressure was approximately 0.85 seconds. It has been shown from past flights that the first start of a duty cycle is generally longer than subsequent starts by a factor of approximately two. This difference appears to be because of a difference in engine priming conditions, since prior to the first start, certain engine ducts are dry. Since this was the second start of the duty cycle, the delay time appeared reasonable and compared favorably with similar starts experienced during Apollo 5 and Apollo 9 flights.

The start transient from FS-1 to 90% of the steady-state throttle setting (13.1% of full thrust) required 2.13 seconds with a start impulse of 728 lbf-sec. The transient time was well within the specification limit of 4.0 seconds for a minimum throttle start. The measured impulse compared favorably with the predicted (Reference 5) nominal value of 862 lbf-seconds (although the nominal predicted time was approximately one second greater than measured) as well as similar starts performed during Apollo 5. The measured value was somewhat low when compared with DPS starts on Apollo 9. One possible reason this deviation may be the coast time between burns. Although there is insufficient flight data to fully correlate the effects, it appears that the magnitude of the start impulse may be proportional to the coast time between burns. This is due to residual propellants freezing in the injector at engine shutdown before they can reach the combustion chamber. An appreciable amount of time is required for these propellants to sublime away. The result can be partially primed injector at engine restart. The coast time between the burns performed on Apollo 10 was approximately 72 minutes which is less than all coast periods with the exception of the coast between DPS 2 and DPS 3 on Apollo 5 (0.5 min). The magnitude of the start impulse for the Phasing Burn falls between that of the Apollo 5 DPS 3 start and the other starts from Apollo 5 and Apollo 9.

The shutdown transient required 0.34 seconds from FS-2 to 10% of the steady-state throttle setting (FTP) with an impulse of 2041 lbf-sec. Both the time and impulse for the transient are greater than observed during Apollo 5, where similar shutdowns were conducted, but compares favorably with the nominal predicted values of 0.32 seconds and 2017 lbf-sec. The transient time was, however, greater than the specification limit of 0.25 seconds for shutdowns performed from FTP. There is no specification limit on impulse. The impulse from FS-2 to zero thrust as determined by consideration of spacecraft weight and vehicle velocity gain was 2948 lbf-sec. This agrees well with the predicted value of 3089 lbf-secs but is somewhat greater than the impulse experienced on Apollo 5 shutdowns. Table 6 presents a summary of the transients.

Throttle Response

During the Phasing Burn, the engine was automatically throttled from the minimum throttle position to FTP. The time from first movement of the engine actuator, to five psi less than steady-state chamber pressure at FTP was 0.94 seconds. This was within the specification limit of 1.0 seconds. This value is 0.6 seconds greater than a similar throttle change performed during Apollo 5 but was similar to like throttling performed during Apollo 9 (40% to FTP in 0.82 seconds).

8. REFERENCES

1. TRW IOC 69.4354.2-54, "DPS Input to Apollo 10 Mission Report," from R. L. Barrows to D. W. Vernon, dated 16 June 1969.

2. NASA/MSC Report MSC-00126, "Apollo 10 Mission Report," dated Aug., 1969.

3. SPD9-R-037, "Mission Requirements, SA-505/CSM-106/LM-4, F Type Mission, Lunar Orbit," dated 31 January 1969.

4. TRW Letter 69.7254.3-61, "Support to the Postflight Analysis of the GN&C Systems on Apollo 10," from D. L. Rue to J. F. Hanaway, dated 1 July 1969.

5. TRW IOC 69.4354.2-12, "Apollo Mission F/LM-4/DPS Preflight Performance Report," from R. K. M. Seto, D. F. Rosow, S. C. Wood and J. O. Ware to D. W. Vernon, dated 21 February 1969.

6. TRW IOC 69.4354.1-70, "LM-4 DPS Fuel Ullage Pressure Prior to First Burn," from R. L. Barrows to P. H. Janak, 6 June 1969.

7. GAEC Report LED-271-98, "PQGS Accuracy Study," dated 14 May 1969.

8. Trans Sonic, Inc. Report No. 4-04003C, "Final Report Analysis of Stagnation (Time Lag) Anomaly in Propellant Quantity Gaging Section," dated April, 1969.

9. SNA-8-D-027 (III), Rev. 1, "CSM/LM Spacecraft Operational Data Book," Vol. III, Mass Properties, dated November 1968.

10. MSC Memorandum EP22-41-69, "Transient Analysis of Apollo 9 LMDE," from EP2/Systems Analysis Section to EP2/Chief, Primary Propulsion Branch, dated 5 May 1969.

TABLE 1

DPS MISSION DUTY CYCLE (4)

BURN	FS-1 (HR:MIN:SEC)	FS-1 (SECONDS)	FS-2 (HR:MIN:SEC)	FS-2 (SECONDS)	BURN DURATION (SEC)	VELOCITY CHANGE (FT/SEC)
DPS 1 (1)	99:46:00.89	359160.89	99:46:28.3	359188.3	27.4 (2)	70.66
DPS 2 (3)	100:58:25.89	363505.89	100:59:05.83	363545.83	39.94	175.82

(1) Reference 4

(2) Astronaut Reports.

(3) Transient Analysis

(4) Referenced to Ground Elapsed Time (GET)

TABLE 2

LM-4 DPS ENGINE AND FEED SYSTEM

PHYSICAL CHARACTERISTICS

ENGINE	
Engine Number	1039
Chamber Throat Area, In^2	53.740[1]
Nozzle Exit Area, In^2	2569.7[4]
Nozzle Expansion Ratio	47.6[4]
Oxidizer Interface To Chamber Resistance at FTP $\frac{lbm\text{-}sec^2}{lbf\text{-}ft^5}$	3904.6[3]
Fuel Interface To Chamber Resistance At FTP $\frac{lbm\text{-}sec^2}{lbf\text{-}ft^5}$	6207.9
FEED SYSTEM	
Oxidizer Propellant Tanks, Total Ambient Volume, Ft^3	126.0
Fuel Propellant Tanks, Total Ambient Volume, Ft^3	126.0[4]
Oxidizer Tank To Interface Resistance, $\frac{lbm\text{-}sec^2}{lbf\text{-}ft^5}$	496.11[2]
Fuel Tank To Interface Resistance, $\frac{lbm\text{-}sec^2}{lbf\text{-}ft^5}$	757.68[2]

[1] TRW No. 01827-6125-T000, TRW LM Descent Engine Serial No. 1039 Acceptance Test Performance Report Paragraph 6.9, 8 December 1967.

[2] GAEC Cold Flow Tests

[3] TRW No. 4721.3.68-188, LM-4, Engine Serial No. 1039 Descent Engine Characteristic Equations, July 1968.

[4] Approximate Values

TABLE 3

DESCENT PROPULSION SYSTEM FLIGHT DATA

MEASUREMENT NUMBER	DESCRIPTION	RANGE	SAMPLE RATE SAMPLE/SEC
GQ3435P	Pressure, Supercritical Helium Tank	0-2000 psia	1
GQ3015P	Pressure, Ambient Helium Bottle	0-1750 psia	1
GQ3018P	Pressure, Helium Regulator Outlet Manifold	0-300 psia	1
GQ3025P	Pressure, Helium Regulator Outlet Manifold	0-300 psia	1
GQ3611P	Pressure, Engine Fuel Interface	0-300 psia	200
GQ4111P	Pressure, Engine Oxidizer Interface	0-300 psia	200
GQ6510P	Pressure, Engine Thrust Chamber	0-200 psia	200
GQ3603Q	Quantity, Fuel Tank No. 1	0-95 percent	1
GQ3604Q	Quantity, Fuel Tank No. 2	0-95 percent	1
GQ4103Q	Quantity, Oxidizer Tank No. 1	0-95 percent	1
GQ4104Q	Quantity, Oxidizer Tank No. 2	0-95 percent	1
GQ4455X	Low Point Sensor, Propellant Tanks Liquid Level	Off-On	1
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
GQ6806H	Position, Variable Injector Actuator	0-100 percent	50
GH1311V	Volts, Manual Thrust Command	0-14.6 VDC	1
GH1331V	Volts, Auto Thrust Command	0-12 VDC	10
GG0001X	PGNS Downlink Data	40 Bits	1/2

TABLE 4

DESCENT PROPULSION SYSTEM STEADY STATE PERFORMANCE
PHASING BURN

PARAMETER	FS-1 + 10 Seconds			FS-2 + 35 Seconds		
Instrumented	Preflight Predicted	Measured	Postflight Predicted	Preflight Predicted	Measured	Postflight Predicted
Throttle Position, %	11.3	13.1	13.1	FTP	FTP	FTP
Regulator Outlet Pressure, psia	247	247	247	247	247	247
Oxidizer Interface Pressure, psia	243.7	243[1]	243.0	224.6	225[1]	225.0
Fuel Interface Pressure, psia	243.6	243	243.0	224.9	225	225.0
Engine Chamber Pressure, psia	12.7	14	14.8	105.9	106	105.9
Oxidizer Bulk Temperature, °F	70.0	69	69.0	70.0	69	69.0
Fuel Bulk Temp., °F	70.0	70	70.0	70.0	70	70.0
Derived						
Oxidizer Flowrate, lbm/sec	2.44		2.85	19.87		19.92
Fuel Flowrate, lbm/sec	1.53		1.77	12.51		12.46
Propellant Mixture Ratio	1.601		1.605	1.589		1.599
Vacuum Specific Impulse, sec	296.79		296.48	303.64		303.98
Vacuum Thrust, lbf	1181		1371	9831		9841
Throat Erosion, %	0.993		0.993	0.994		0.994

[1]These values corrected in accordance with text.

TABLE 5

DPS GAGING SYSTEM PERFORMANCE

END OF PHASING BURN

PARAMETER	Time, hr:min:sec
	100:59:06
Oxidizer Tank 1	
Measured Quantity, percent	93.8
Calculated quantity, percent	92.9
Difference, percent	+0.9
Oxidizer Tank 2	
Measured quantity, percent	94.5
Calculated quantity, percent	92.9
Difference, percent	+1.6
Fuel Tank 1	
Measured Quantity, percent	92.4
Calculated quantity, percent	92.7
Difference, percent	-0.3
Fuel Tank 2	
Measured quantity, percent	92.0
Calculated quantity, percent	92.7
Difference, percent	-0.7

TABLE 6

DPS START AND SHUTDOWN IMPULSE SUMMARY

	Apollo 10 LM-4/DPS-2	Apollo 5 LM-1/DPS-2	Apollo 5 LM-1/DPS-3	Apollo 9 LM-3/DPS-1	Apollo 9 LM-3/DPS-2	Apollo 9 LM-3/DPS-3	SPECIFICATION LIMITS
STARTS							
Steady-State Throttle Position, Percent	13.1	12.4	12.4	12.7	12.7	12.7	
Total Vacuum Start Impulse (FS-1 to 90% steady state), lbf-sec	728	894	574	805	1029	950	
Start Time (FS-1 to 90% steady state), sec.	2.13	2.66	2.13	2.5[1]	2.1	2.3[1]	4.0
Coast Time From Prior Burn, Minutes	72	131	0.5	From Launch	2640	111	
SHUTDOWNS							
Steady-State Throttle Position, percent	FTP	FTP	FTP	40	40	12.7	
Total Vacuum Shut-down Impulse (FS-2 to 10% Steady State), lbf-sec	2041	1727	1713	---[2]	1730	748	
Shutdown Time (FS-1 to 90% steady-state), sec	0.34	0.26	0.30	1.1[1]	1.1	1.8[1]	0.25[4]
Repeatability, lbf-sec		1734 ±7	1734 ±7				±100[4]
Total Vacuum Shutdown Impulse (FS-2 to Zero Thrust) from Velocity Gain Data, lbf-sec	2948	2493	---[3]	1777	---[2]	1040	

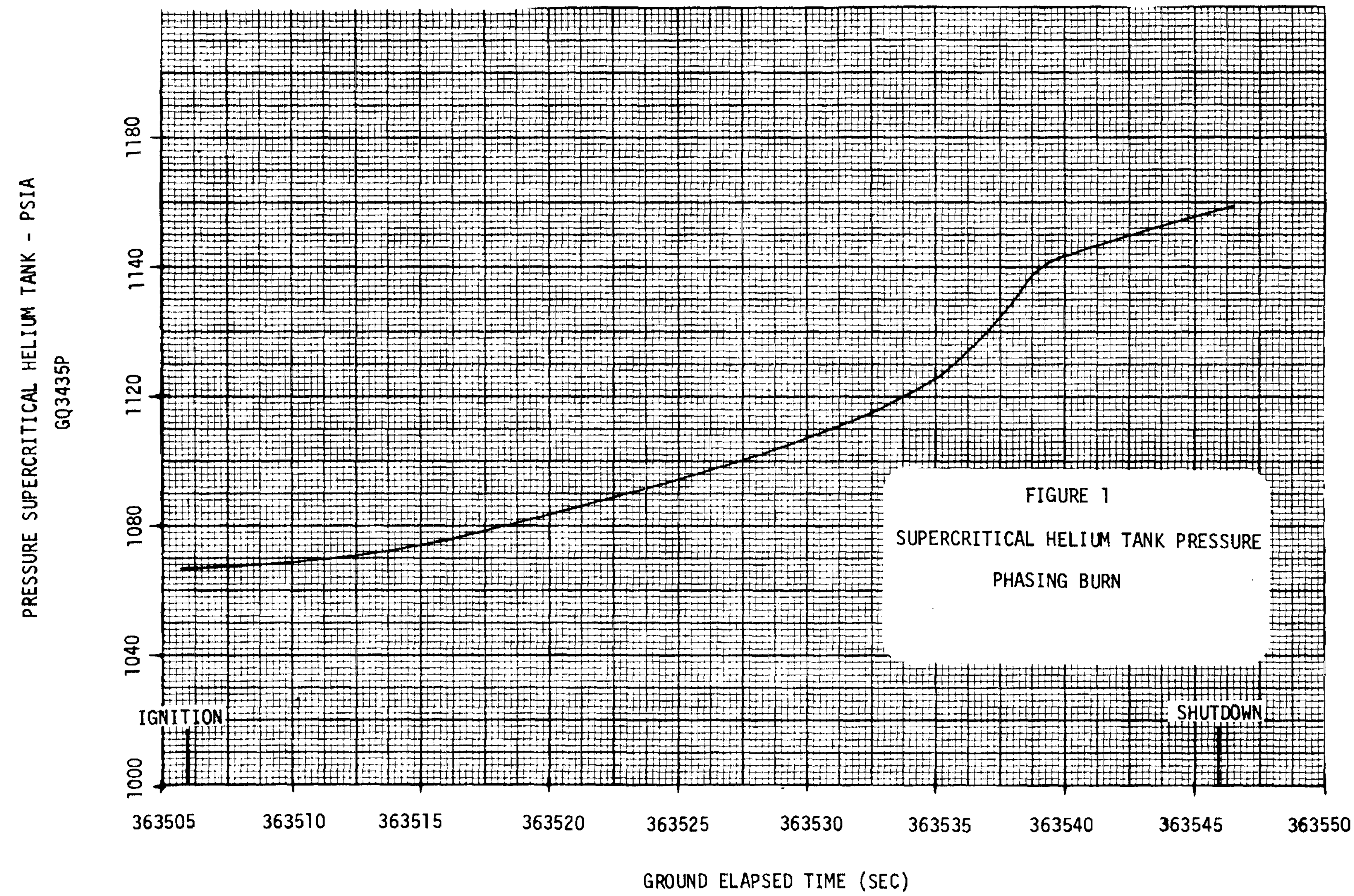
[1]Reference 10.

[2]Data unavailable.

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

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

APOLLO 10 SC 106/LM4 - DPS



GQ3435P

FIGURE 1

SUPERCRITICAL HELIUM TANK PRESSURE

PHASING BURN

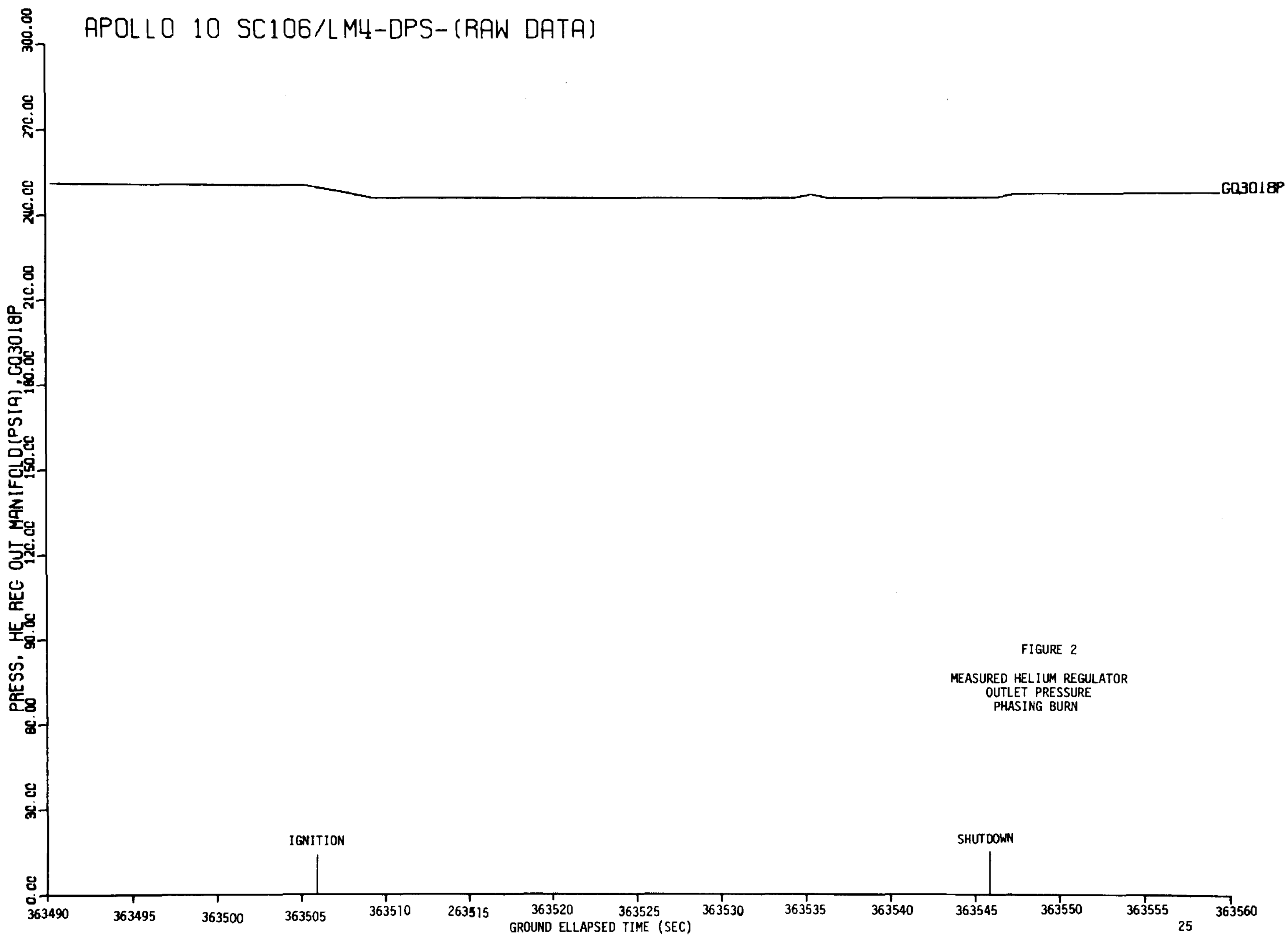


FIGURE 2

MEASURED HELIUM REGULATOR
OUTLET PRESSURE
PHASING BURN

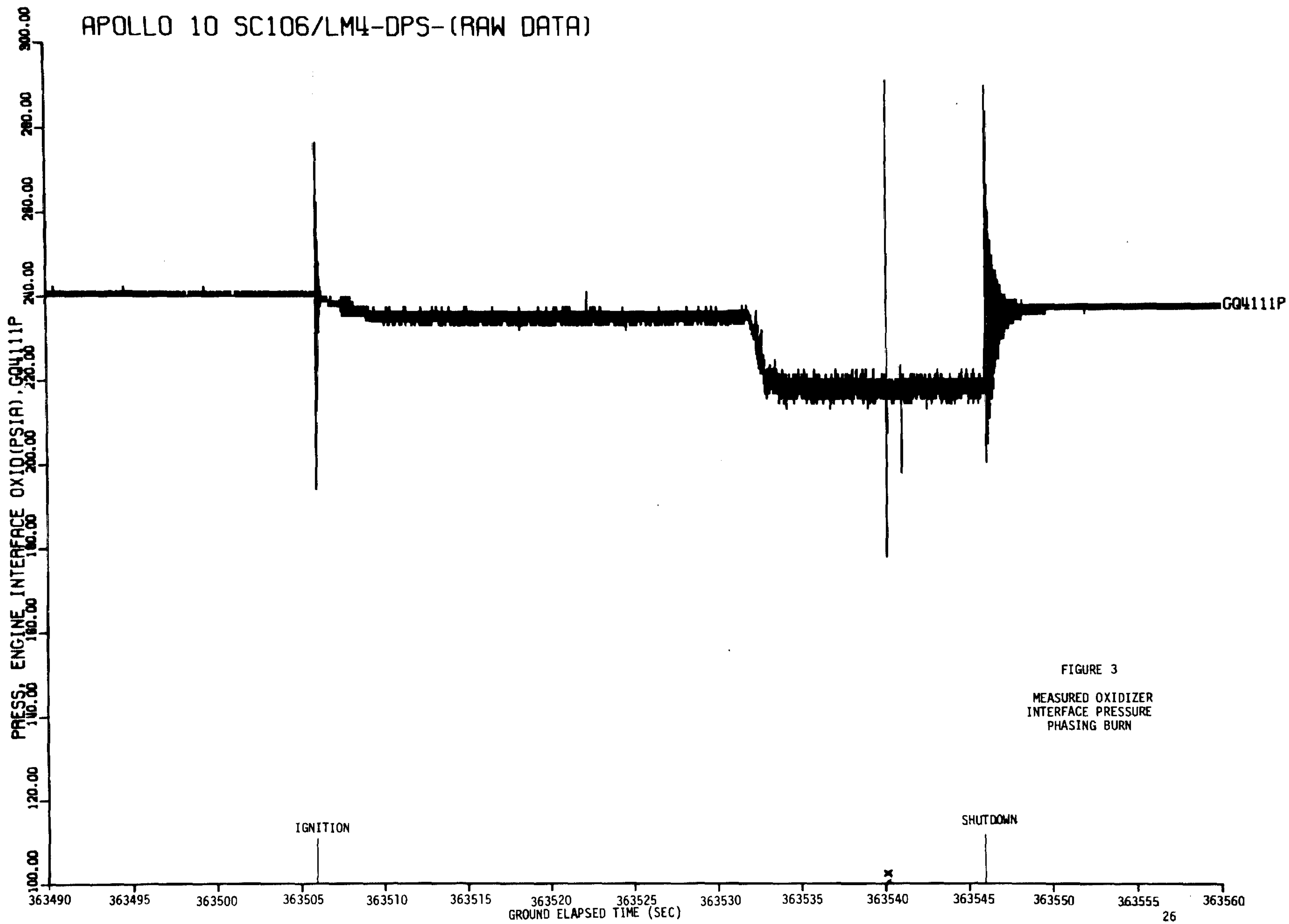


FIGURE 3

MEASURED OXIDIZER
INTERFACE PRESSURE
PHASING BURN

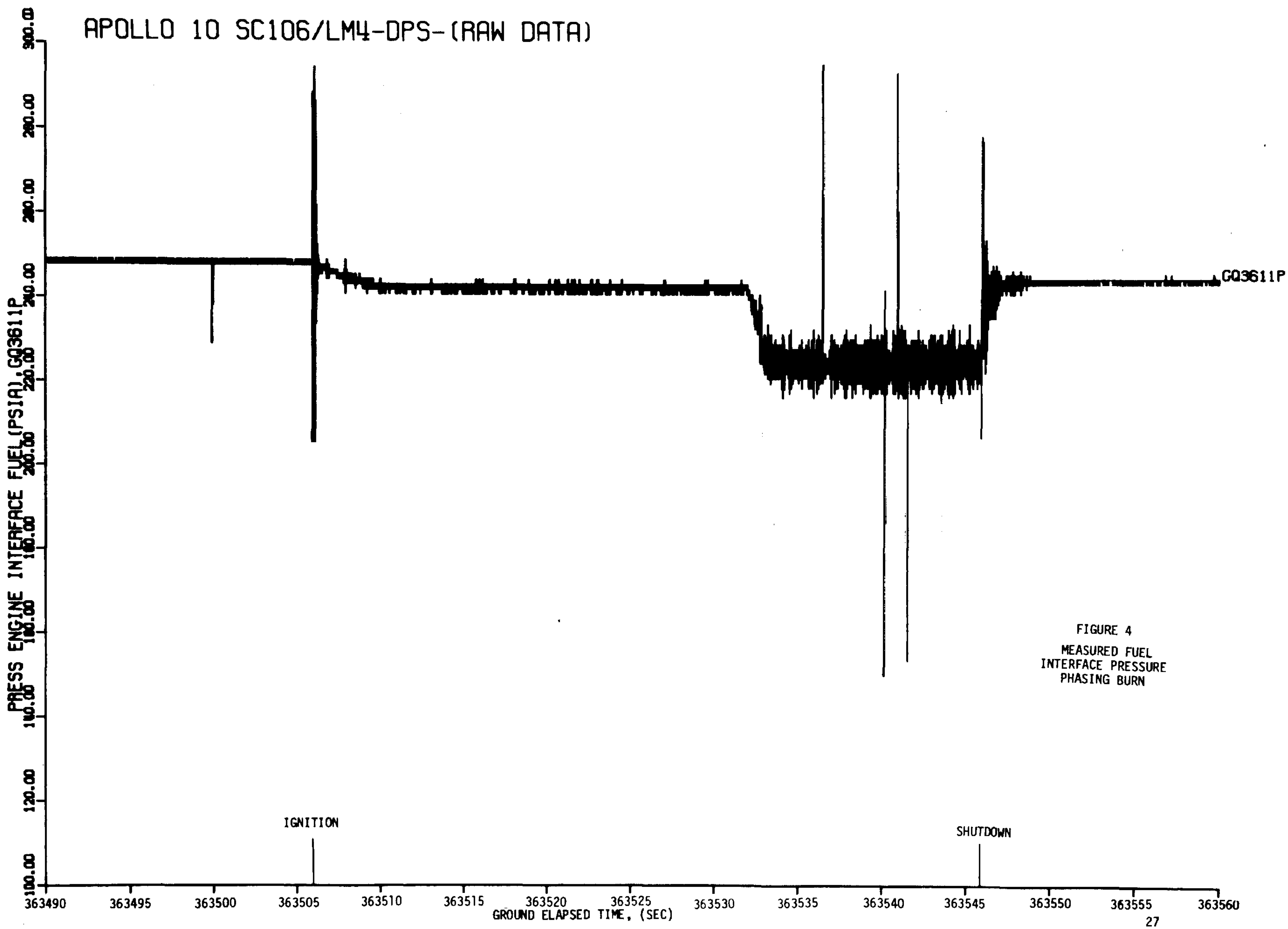


FIGURE 4
MEASURED FUEL
INTERFACE PRESSURE
PHASING BURN

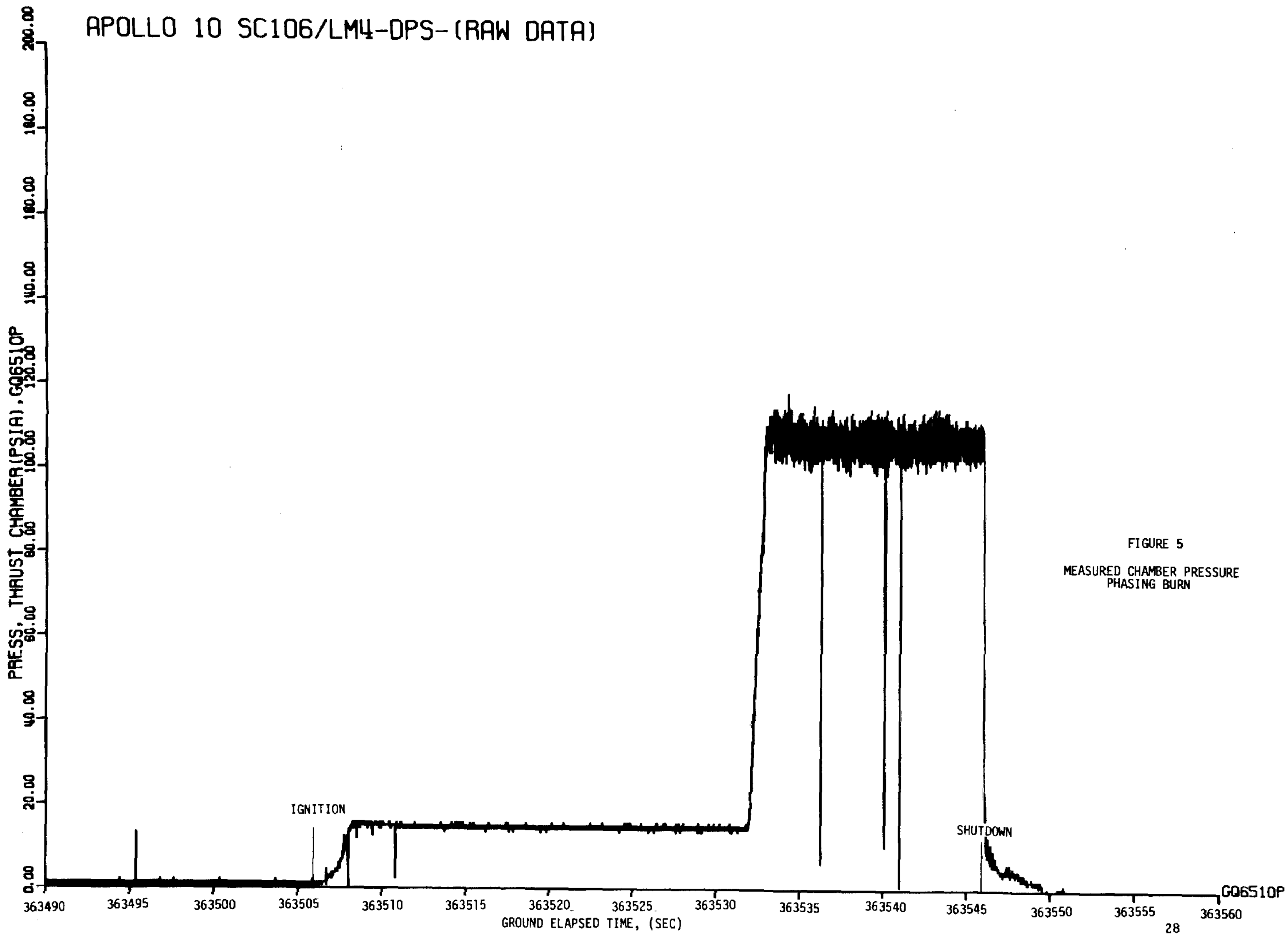


FIGURE 5

MEASURED CHAMBER PRESSURE
PHASING BURN

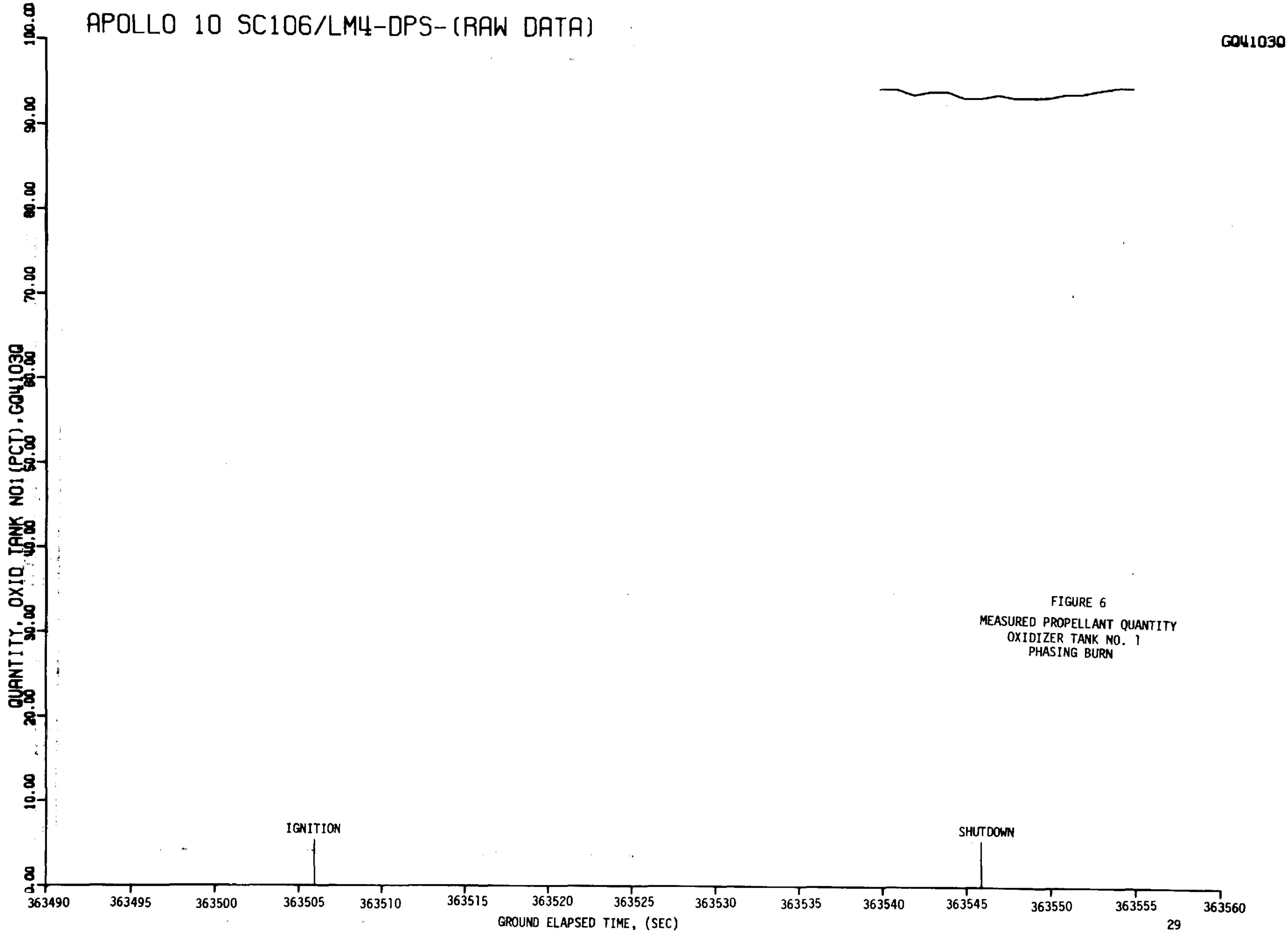


FIGURE 6
MEASURED PROPELLANT QUANTITY
OXIDIZER TANK NO. 1
PHASING BURN

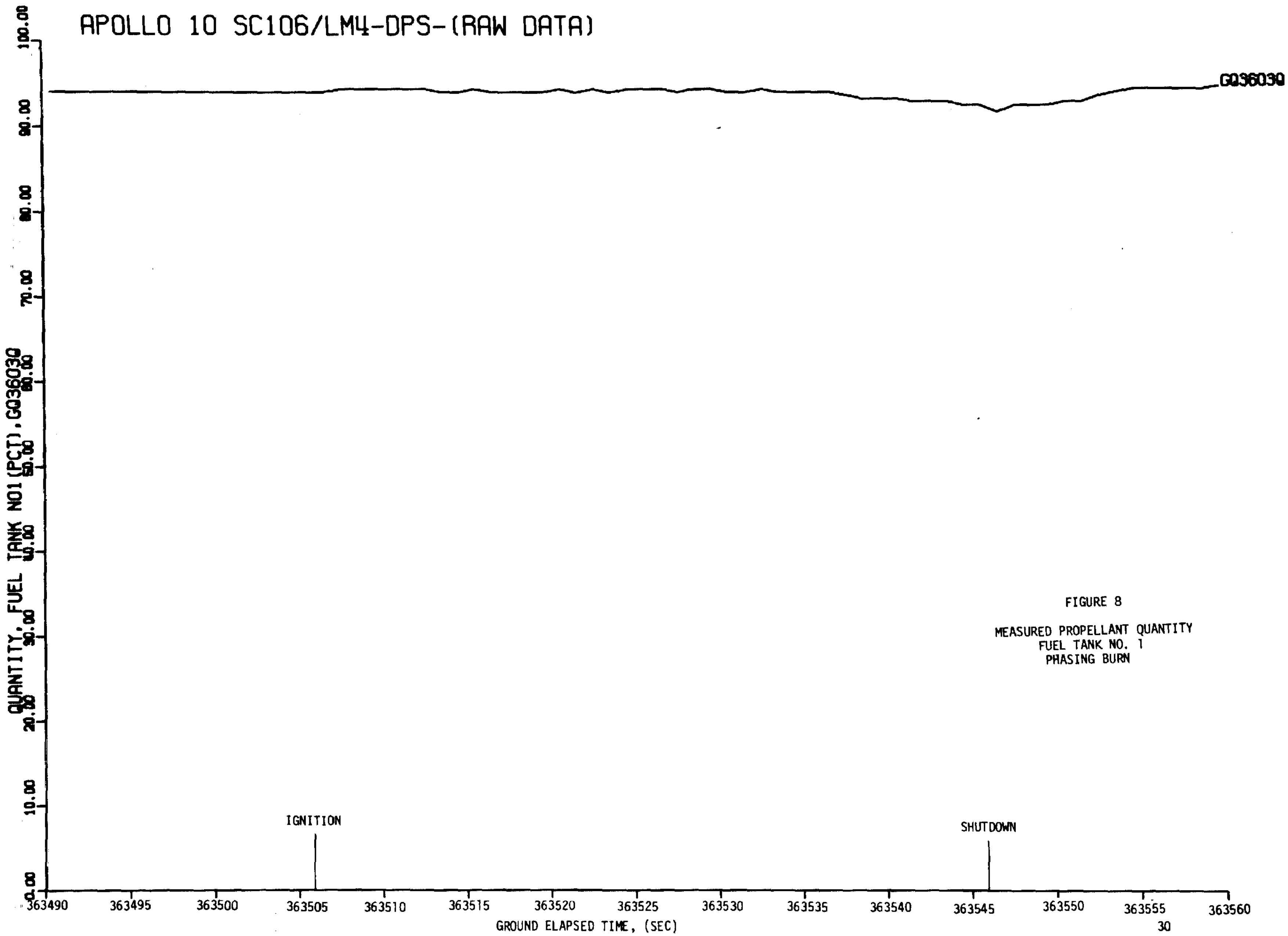


FIGURE 8

MEASURED PROPELLANT QUANTITY
FUEL TANK NO. 1
PHASING BURN

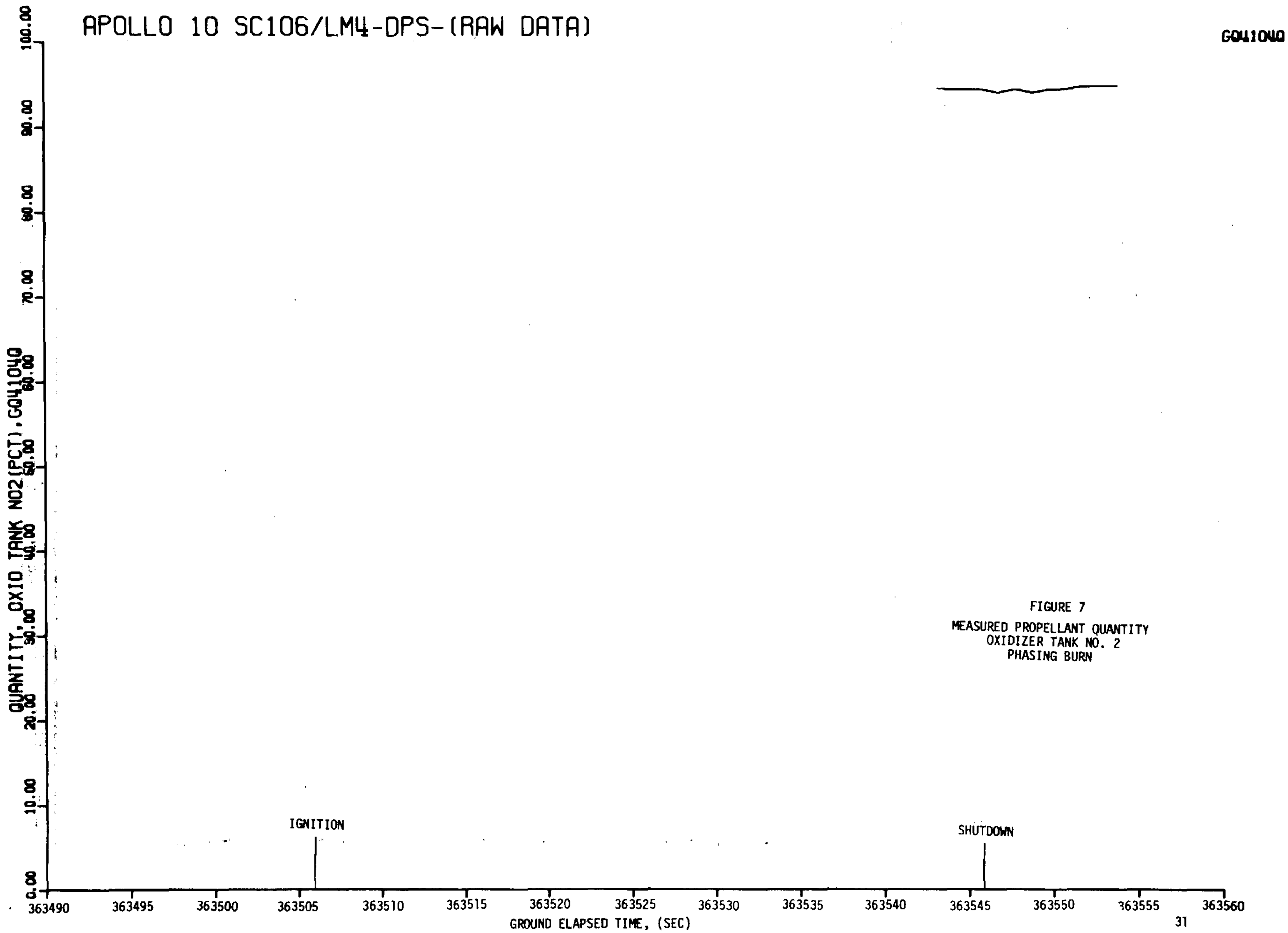


FIGURE 7
MEASURED PROPELLANT QUANTITY
OXIDIZER TANK NO. 2
PHASING BURN

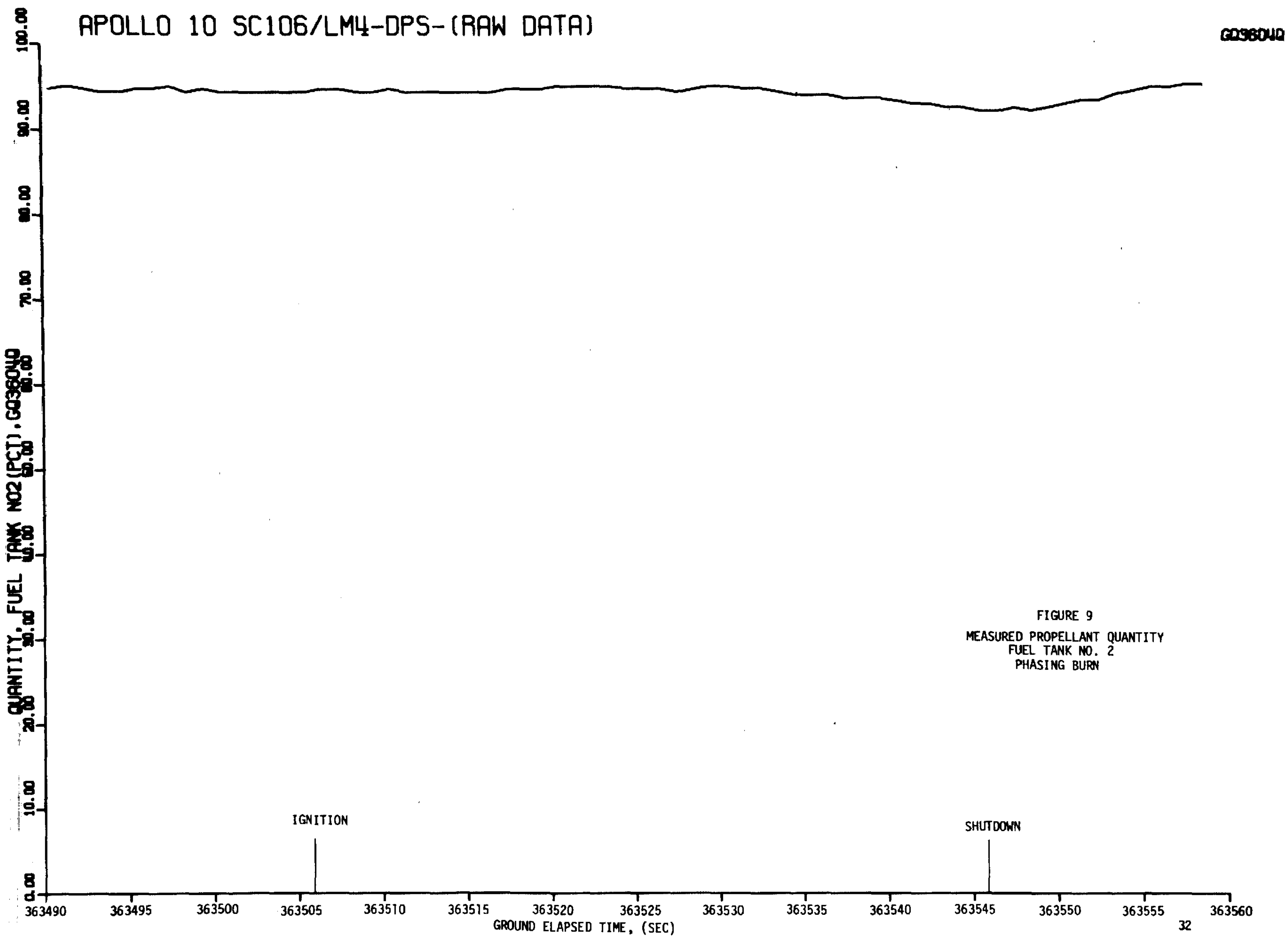


FIGURE 9
MEASURED PROPELLANT QUANTITY
FUEL TANK NO. 2
PHASING BURN

NASA — MSC