

Voyager Mission Support (II)

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This is a continuation of the Deep Space Network report on Tracking and Data Acquisition for Voyager Project. This article covers the period August through October 1980.

I. Introduction

Voyager 1 has entered the Encounter Phase of the mission, while Voyager 2 continues in the Jupiter-Saturn Cruise Phase.

The Saturn Encounter activities are divided into five phases based on the field of view of the narrow-angle camera in relation to the distance to the planet. The five phases are Observatory, Far Encounter 1, Far Encounter 2, Near Encounter, and Post Encounter.

The Observatory Phase for Voyager 1 started on 22 August and ended on 24 October, when the Far Encounter 1 (FE1) Phase started. The FE1 Phase will not be concluded until 2 November 1980.

II. DSN Support

A. Near Encounter Test

Starting in July 1980, the Voyager Mission Support Team, including all support facilities, entered the Test and Training period for the forthcoming encounter. The DSN performed the required System Performance Tests and Operational Verification Tests necessary to check out and train operations personnel on the new capabilities as implemented. This training activity was culminated in a Near Encounter Test on

19 and 20 August 1980. The Near Encounter Test (NET) was a simulation of the 18-hour period of closest approach to Saturn. During this test both the 34-meter and 64-meter Deep Space Stations were configured and supported the Project in the same manner as will be required during the actual closest approach. During the NET, a Radio Science Operational Readiness Test (ORT No. 3) was conducted with DSSs 43 and 63. The period tested corresponds to the one required to support the closest approach and occultation periods.

Although the support provided by the Deep Space Network was considered satisfactory overall, several areas were identified for which additional attention was required. It was found that generally the arrayed antenna performance was lower than expected. Problems causing this were suspected to be telemetry string degradation due to improper calibration, an adverse weather condition affecting the X-band data, and variations between the 34-meter and 64-meter station symbol signal-to-noise ratio. Special emphasis was given to improving the array performance for the movie phase that was included in the observatory phase.

Problems were likewise encountered during the ORT. Occultation Data Assembly halts were experienced, which caused some data loss. Appropriate calibration on the antenna at DSS 63 could not be performed one time due to an

unusually heavy rain storm during the pretrack preparation period. Although the Spectral Signal Indicator (SSI) and Precision Signal Power Monitor (PPM) data was generally good, problems were experienced in displaying the data at JPL. These were determined to be procedural problems that had not been fully resolved due to the newness of the capability. Additional voice communication had been provided for the radio science support, which was improperly configured and caused operational confusion.

Other minor operational problems, procedural and equipment failures, were experienced that provided insight into the events that could occur during the actual closest approach. The test results provided a valuable insight into the areas for improvement and correction. Actions have been taken to accomplish this and internal testing has validated the measures taken.

B. Observatory Phase

1. Movie phase – solar conjunction. A time-lapse movie was compiled from photographs taken during the period 12 to 14 September 1980. Photographs were taken every 4.8 minutes during four Saturn rotations. The photographs were transmitted in either real-time or during a playback period over DSS 14. To obtain the telemetry data, including the photographs, the 34-meter and 64-meter networks were arrayed for the entire support period. Overlapping this movie activity was the solar conjunction (3 September to 6 October) when the Sun was between the Earth and spacecraft. The smaller the angle of the Sun-Earth Probe, the more hampering of the radio communications was experienced. However, the condition did allow study of the Sun as the radio signals passed through the corona. Although the "noise" was evident in some photographs, the movie requirement was met. Likewise, the solar conjunction data were collected from both Voyager 1 and Voyager 2 to complete the experiment data requirements.

2. Target maneuver. On 5 to 6 August, a target maneuver with Voyager 1 was supported by Deep Space Network. The purpose of the maneuver was to calibrate the imaging and IRIS instruments. The spacecraft was maneuvered to point at the science calibration target and several wide- and narrow-angle images plus IRIS data are recorded. Following the calibration, the scan platform was slewed to a safe position for a turn unwind. The spacecraft then maneuvered to reacquire the Sun and Canopus, and the scan platform slewed back to a neutral position. Recorded calibration data were played back on 7 August. DSSs 12, 61, and 43 were involved in the transmission of the command load, the enable file, tracking until loss of data due to the spacecraft turning, reacquisition of the

spacecraft signal on return to Earth point, and receipt of the transmission of the calibration data. The sequence went as scheduled, and the spacecraft was reacquired by DSS 43 as anticipated.

3. Trajectory correction maneuver (TCM-A8). On 10 to 11 October, Voyager 1 was again maneuvered; however, this time it was to accomplish a trajectory correction. The correction was to change the trajectory so that the spacecraft would pass the Saturnian satellite Titan at a closest point of 4600 km from the surface. Without the maneuver, there was the possibility of a collision with Titan.

The spacecraft was rolled 90 degrees and then turned through a yaw of 136.7 degrees to place it in the proper position for the change in velocity vector. A "burn" of 806 seconds was used to produce a velocity delta change of 1.778 meters/second. The spacecraft was then maneuvered back to Earth point.

During this maneuver, the downlink was lost from the spacecraft after the positioning for the "burn," and reacquired after the spacecraft reposition on Earth point. This maneuver was supported by DSSs 61, 12, and 14.

To provide a small margin of additional capability, the stations were instructed to put in the bypass to the antenna film-height detector. Under normal operation when the antenna film-height detector senses a low film height, the antenna is automatically stopped. By placing the bypass "IN," the operator will manually stop the antenna if the film-height detector alarm is activated for 15 seconds or more. By using this option, false alarms are reduced, and unnecessary data loss avoided during critical data periods.

4. Array readiness. After the implementation of the Real-Time Combiner (RTC) at the 64-meter stations, and during the testing of the arraying operation, it was found that there was a variation in the actual gain achieved by a station during a series of passes. A good deal of the variation was due to the calibration of the telemetry string. Precise and consistent calibration was required to produce the desired gain over a series of passes. To assist the stations in this effort, a period of RTC and telemetry string maintenance was scheduled each week for the station to accomplish the required work.

DSSs 12 and 14 started operational support of Voyager 1 with the array configuration on 23 August 1980. The level of proficiency and improved results were evident after a few days of operation. To insure DSSs 42 and 43 and DSSs 61 and 63 crew proficiency, a series of proficiency test passes were

provided during the normal scheduled support of Voyager 1 by these stations. The gain on the arrayed data has been within the desired level for the encounter period. DSSs 42 and 43 and DSSs 61 and 63 began active support of Voyager 1 on 24 October at the start of the Far Encounter Phase.

5. Radio Science Subsystem (RSS) Operational Readiness Test-4 (ORT-4). The fourth RSS ORT-4 was conducted on 2 to 3 October 1980. The ORT was to demonstrate (1) Occultation Data Assembly (ODA) medium band width right circular polarization/left circular polarization (RCP/LCP) S- and X-band and narrow-bandwidth S- and X-band data acquisition, (2) acquisition of radio science wideband width backup data acquisition using the Digital Recorder Assembly (DRA), (3) capability of measuring and recording system temperatures in real-time, and (4) demonstrate SSI performance and utility for occultation data acquisition.

Although the ORT was conditionally successful, there were problems with the SSI equipment and the wideband backup equipment. There were misunderstandings on the calibration procedures for input and/or output levels of equipment.

To resolve these problems and to increase the proficiency in the operation and support of the RSS effort, a series of 14 OVTs were scheduled for DSSs 43 and 63. The results obtained from the OVTs in increased efficiency and effectiveness became more evident with each succeeding test. Procedures for calibration and operation of system and equipment were clarified and standardized as a product of the tests. By the end of October, the RSS problems were mainly resolved and the system operation greatly improved. ORT-5 is now scheduled for 5 to 6 November 1980 for the final validation of the system and operations for support of the closest approach and occultation experiments with Saturn and Titan.

Improvements in Galileo–Mars Navigation Using the Viking Lander

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When the geocentric angular separation between two spacecraft is but a few degrees, navigational advantages may be achieved by navigating one spacecraft with respect to the other. In this dual-spacecraft navigation technique, radio metric data from the two spacecraft are not treated independently, but differenced to cancel common observable modeling errors.

In the circumstance of the Galileo Spacecraft flyby of Mars, the Mars Viking Lander 1 might provide a radio beacon that could be used to navigate Galileo past Mars. The Viking Lander has been operating on the Martian surface since July 20, 1976, and is expected to continue through 1990. It is intended that the navigational delivery accuracy capability of Galileo at Mars [25 km (1σ)] is going to be met with interferometric angular measurements (VLBI) and range and range-rate measurements. Like VLBI, however, dual-spacecraft differenced range has little sensitivity to transmission media modeling errors, and to tracking-station location errors. Similarly, differenced range provides angular information about the separation between the Mars Viking Lander 1 and the Galileo Spacecraft. In covariance studies, dual-spacecraft differenced range coupled with conventional range and doppler is shown to estimate the Galileo-Mars flyby distance to better than 10 km (1σ), which is favorably comparable to the projected Δ VLBI performance. For the Galileo-Mars flyby, dual-spacecraft differenced range promises to be an excellent backup to VLBI if the Mars Viking Lander remains operational.

I. Introduction

The Galileo mission (Ref. 1) is a highly ambitious scientific project and this will be the first time an artificial satellite will be placed in an orbit around an outer planet. The major objectives of this mission are to maximize the number of flybys of the Galilean satellites and maximize the scientific return about the Jovian neighborhood. This and the constraints due to the shuttle-IUS (Interim Upper Stage) launch capability requires

an interplanetary trajectory that includes a Mars flyby, which provides a needed gravity assist. Subsequent to this flyby, a propulsive maneuver is executed that places the spacecraft on a Jupiter-bound trajectory. To minimize the ΔV required for this maneuver, it is desired to pass as close to Mars as possible, consistent with planetary quarantine and spacecraft safety constraints. These constraints have placed the requirement on the navigation system that the delivery accuracy at Mars should be better than 20 to 25 km (1σ).

The conventional radio metric data, two-way doppler and range measurements from the Deep Space Network stations, can provide a heliocentric position accuracy for the Galileo spacecraft to an accuracy of about 35 km (1σ). However, this does not include the uncertainty in the Mars ephemeris. When the projected Mars ephemeris uncertainty of 40 km (1σ) is considered, the spacecraft position uncertainty at the Mars encounter point exceeds 50 km (1σ). Thus, the Galileo Project is planning to augment conventional radio metric two-way doppler and range with VLBI to achieve the required accuracy.

The VLBI method for navigational application has been extensively studied (Refs. 2 and 3); however, the concept is yet to be successfully demonstrated (Ref. 4). The potential application of the wideband VLBI system is discussed by Brown and Hildebrand (Ref. 5). This VLBI technique will be utilized in a differential mode with VLBI data referenced to an extragalactic radio source (EGRS) to difference out various common error sources. This imposes an additional requirement that the Mars ephemeris should be known with reference to an extragalactic radio source frame with an accuracy of better than 20 to 25 km (1σ). There exists an on-going activity, using Viking orbiter data, that is expected to provide the needed accuracy (Newhall, 1980, personal communication).

The Galileo project is also considering the utilization of the onboard optical system to achieve the desired accuracy. The use of optical data at Mars encounter has been successfully demonstrated during the Viking mission. It is not currently decided whether the optical system will be available for approach navigation at the Mars encounter phase.

Although the use of interplanetary beacons in general deep-space navigation has not been adequately investigated, the possible use of the Viking Lander as a beacon in Galileo navigation has been examined in covariance studies. Galileo-Viking, dual DSS range promises to improve conventional radio metric range and doppler orbit determination by a factor of 4. Its performance is favorably comparable to that of VLBI.

II. Radio Metric Measurements

Since the early 1960s, interplanetary navigation has been accomplished with such conventional radio metric measurements as the coherent two-way doppler data and two-way range data (Ref. 7). Measurement accuracies and model accuracies have been improved significantly over the years; however, the new anticipated navigation functions require alternate radio metric techniques to achieve the projected accuracy requirements.

A brief examination of the information content of these radio metric measurements is presented here. By processing

one pass of coherent doppler data, the primary orbit parameters (the geocentric range rate, the right ascension, and declination of the spacecraft) can be determined. The accuracy with which these parameters can be determined is given by the following equations:

$$\sigma_{\dot{r}} = \sigma_{\dot{\rho}}$$

$$\sigma_{\alpha} = \frac{1}{\omega r_s \cos \delta} \pi \sigma_{\dot{\rho}}$$

$$\sigma_{\delta} = \frac{1}{\omega r_s \sin \delta} \frac{\pi}{2} \sigma_{\dot{\rho}}$$

where r , α , and δ are the geocentric range rate, the right ascension, and declination of the spacecraft respectively, $\dot{\rho}$ is the range rate measurement, ω is the spin rate of the earth, and r_s is the distance off the spin axis of the DSS.

Single-station range data taken over a pass provides the same information as that using doppler; however, the geocentric range information is better known. These, of course, are radial measurements. These measurements are corrupted primarily by transmission media modeling errors, station location errors, instrumentation errors, and unmodeled spacecraft accelerations. It is possible to minimize some of these errors by combining radio metric doppler and range from two stations and from two spacecraft into first and second differences. Errors that are common cancel. For first differenced range and doppler, spacecraft unmodeled accelerations, solar plasma corruptions, and spacecraft oscillator instability cancel, while ionospheric and tropospheric corruptions and station location and clock errors do not cancel. For dual-spacecraft-dual-station, differenced range and range-rate that is doubly differenced data, even the ionospheric, tropospheric, and station errors mostly cancel to yield relatively error-free observables. For doubly differenced range where one spacecraft is tied to the planet, even the planet ephemeris errors cancel to a large degree. The information that doubly differenced range does process is the relative right ascension and declination between two spacecraft.

$$\Delta^2 \rho = \Delta Z [\cos \delta \Delta \delta]$$

$$- \Delta L [\sin (\alpha - \theta) \cos \delta \Delta \alpha + \cos (\alpha - \theta) \sin \delta \Delta \delta]$$

with ΔZ and ΔL being the east-west and north-south projections of the baseline on the plane-of-sky, and $\Delta \alpha$ and $\Delta \delta$ being the separation between the spacecraft and the beacon in right ascension and declination. θ is the local sidereal time.

III. Doubly Differenced Range Measurements

The use of differenced radio metric data from two spacecraft has been previously studied (Refs. 6 and 7). Dual-spacecraft-dual-station doubly differenced range simply carries the process one step further as suggested by Chao (Ref. 7).

Figure 1 shows the various components required to form this measurement. Stations A and B shown in Fig. 1 are separated by intercontinental distance. The two spacecrafts are shown as the Viking Lander (1) and the Galileo Spacecraft (2). Station A transmits a signal to the Lander and one round-trip light time later the range measurement is acquired. Range measurements from the Lander are acquired for about 10 to 15 minutes. Station B then transmits a signal to the Lander and ranging data are acquired a round-trip light time later. Then this procedure is repeated using both stations with Galileo. The doubly differenced range measurement can be formed from the following equation:

$$\Delta^2 \rho = \{ \rho_{1A}(t_1) - \rho_{1B}(t_2) \} - \{ \rho_{2A}(t_3) - \rho_{2B}(t_4) \}$$

where ρ_{ij} ($i = 1, 2; j = A, B$) are the two-way range measurements, and t_k ($k = 1, 2, 3, 4$) are the corresponding station acquisition times.

As discussed earlier, each two-way range component of this measurement is susceptible to various error sources with most of the errors thought common to multiple links. Thus, the differencing process is expected to achieve cancellation of most of these errors. A theoretical error budget has been formed to account for those errors that do not completely cancel in the formation of dual spacecraft differenced range (Table 1). The assumptions made in generating this error budget are that the Viking Lander/Galileo separation angle is about 5 degrees, the data are taken near Mars opposition and at about 0.7-AU distance, and two stations observe the Lander for approximately 15 minutes each and then observe the Galileo spacecraft for the same amount of time. Also, the measurements are assumed to be taken at about 25-deg elevation angle. This may not be totally realistic because in general the observations from at least one of the two stations has a low elevation (10 to 15 deg) when the spacecraft is visible from two widely separated stations. However, 90 percent of the error budget stems from the system noise term and not the media errors. A root-sum-square (rss) error of about 2.2 m is obtained for a doubly differenced range measurement, and this is assumed to be random because the error is mostly due to the thermal white noise.

Since the achievable navigation accuracy using the technique described in this article strongly depends on the assumed

measurement error, it is important to validate the measurement accuracy using existing spacecraft. Thus, near-simultaneous ranging experiments have been already conducted using the Viking Orbiter 1 and Lander 1, and are being planned using the Voyager 1 and 2 spacecraft.

Before the Orbiter became inoperative in July 1980, there were two opportunities to acquire doubly differenced range: 5 June 1980 (this attempt failed), and 28 June 1980. The 28 June experiment was successful. The difference between the Lander and Orbiter relative range residuals was 3.4 m. Details of this data validation process are contained in the Appendix. To date, this is the only empirical assessment of doubly differenced range rms error. Voyager may provide additional opportunities in the near future.

IV. Galileo-Mars Flyby Navigation

The Galileo mission presents a number of navigational challenges (Ref. 1); one of the more stringent of these relates to the Mars flyby phase of the mission.

Galileo Project plans call for the Galileo Spacecraft to flyby Mars ~ 200 km ($\sigma_d = 25$ km) above the planet's surface (Ref. 1). To achieve this accurate flyby, two new technological advances must be accomplished: one, the Mars ephemeris must be improved to better than 25 km (1σ); and two, a wide-band Very Long Base Interferometry (VLBI) technology must be developed that will permit the Galileo Spacecraft and Mars trajectories to be defined relative to a quasar inertial reference frame. Both efforts are underway and offer a means to reduce the Galileo-Mars relative trajectory errors, and also to obtain observables free from the preponderance of the Deep Space Station (DSS) location effects and transmission media effects.

What is shown here is that doubly differenced range from Viking Lander 1 and the Galileo Spacecraft can, like VLBI, achieve the Project Mars flyby requirement, providing the Lander survives. However, doubly differenced range is (1) essentially independent of planetary ephemeris uncertainty, and (2) is operationally simpler to use than VLBI.

There are 43 opportunities to obtain Viking range during the Galileo Mars approach. Viking has been programmed to transpond range on those 43 occasions (Table 2). This limit in opportunities exists because

- (1) Galileo cruise time from Earth to Mars is ~ 93 days (March 1984 to June 1984).
- (2) Lander thermal and power constraints permit only two 13-minute contiguous ranging segments per day.

- (3) The Lander is not always in view during overlapping portions of tracking station view periods of Mars.

These joint Viking-Galileo ranging opportunities are shown on Fig. 2 and listed in Table 2. Since the Martian day ($\sim 24^h 37^m 23^s$) and Earth day are of comparable length, for a given hour angle of the Earth the relative geometry between DSS baselines and the Lander changes very slowly. Each baseline, in turn, can view the Lander for nearly eight days continuously.

The DSS identification numbers 14, 43, and 63 represent the Deep Space Network Stations at Goldstone, California; Canberra, Australia; and Madrid, Spain, respectively.

Each baseline's performance is not only time dependent, but is also governed by the alignment of the baseline with respect to the Galileo-Mars direction at encounter (Fig. 3). In essence, first differenced range from a spacecraft provides information as to the direction of that spacecraft with respect to the baseline but only in the direction of the baseline. Orthogonal to the baseline, there is no information. When measurements from two spacecraft are differenced to obtain $\Delta^2\rho$, $\Delta^2\rho$ defines the component of the earth-centered angular separation between the two spacecraft in the baseline direction. Figure 3 shows the baseline orientations relative to the Mars-Galileo direction at encounter. The DSS 43 - DSS 63 baseline, which is approximately 4-deg offset, yields the strongest information concerning the flyby distance, while the DSS 63 - DSS 14 (~ 12 -deg offset) and the DSS 14 - DSS 43 (~ 60 -deg offset) baselines provide progressively less information.

V. Covariance Analysis

The covariance analysis performed in this paper assumed a maximum likelihood estimator with gaussian errors on the observations. The assumed observations include two-way coherent doppler data from the Galileo Spacecraft using the three Deep Space Network stations continuously, one Doppler measurement every hour, one range measurement from the Goldstone station every day and the available doubly differenced range measurements as shown in Fig. 2. Since the dynamical state parameters are nonlinear functions of the measurements, the observation equations are linearized and the results obtained are based on a linear estimator. When a standard maximum likelihood estimator is constructed, the computed statistics based on data noise errors do not reflect the effect of model errors in the solution. Thus, the statistics must be adjusted to account for their effects.

The measurement equation can be written in this form:

$$z = Ax + Cp + e$$

where z is the vector of measurements, x the vector of estimated parameters, p the vector of model parameters whose effects on the estimated parameters are to be investigated, and e the vector of measurement errors. A weighted least squares estimator of (\hat{x}) can be obtained by

$$(\hat{x}) = [A^T P^{-1} A]^{-1} A^T P^{-1} z$$

with the assumption that p is a random vector of zero mean with covariance P_c , $E(e) = 0$, $\text{cov}(e) = P$ and $E(p e^T) = 0$ and the covariance of (\hat{x}) is given by

$$P_x^c = \text{Cov}(\hat{x}) = P_x + P_x A^T P^{-1} C P_c C^T P^{-1} A P_x$$

where $P_x = (A^T P^{-1} A)^{-1}$ is the noise covariance matrix. The matrix P_x^c is known as the "consider" covariance matrix and the matrices A and C are the partial derivatives of the measurement with respect to the estimated and the consider parameters. Both station locations and Mars ephemeris parameters are treated as "considered" parameters. Only the Galileo trajectory is being estimated. The a priori uncertainties of the parameters are given in Table 3.

In this covariance analysis the doppler data accuracy is assumed to be 1 mm/s with a 60-s averaging time, the range data is weighted with 1-km accuracy. The doubly differenced range measurements are assumed to be accurate to 2 m. Figure 4 presents the results of the covariance analysis. The flyby distance uncertainty is below the 10-km level about 25 days before the encounter.

If $\Delta^2\rho$ is not employed, conventional range and doppler estimates of the Galileo-Mars flyby distance are dominated by the Mars ephemeris uncertainty of 40 km (1σ). If conventional and VLBI data are employed with an improved Mars ephemeris, 25 km (1σ), the uncertainty in the estimates is still dominated by the ephemeris uncertainty and is always greater than 25 km (1σ).

Estimates of the flyby distance based on doubly differenced range have uncertainties often smaller than the ephemeris sigmas or tracking station location standard deviations because of the like influence of these error sources on the individual range measurements that are differenced to develop a doubly differenced range measurement. The ephemeris and station uncertainties cancel. Figure 5 illustrates how the influences of ephemeris and tracking station location uncertainties are less when $\Delta^2\rho$ is included in the covariance. Here only the Galileo state is estimated. The Mars ephemeris and station location are considered in the manner previously discussed. The uncertainty of the estimate of the Mars encounter distance of Galileo due to the Mars ephemeris uncertainty ($\sigma_d |_{\text{Mars Ephemeris}}$) and due

to station uncertainty ($\sigma_d |_{\text{DSS Locations}}$) rapidly decreases as the tracking data arc of conventional data and $\Delta^2\rho$ become longer. When conventional data only is reduced, sensitivities to these consider parameters do not diminish.

VI. Summary and Conclusion

A newly proposed navigation technique utilizing two-way range data taken nearly simultaneously from two spacecraft has been analyzed and the results clearly show that the relative position of one spacecraft to the other can be determined with an accuracy depending only on the accuracy of the measurement. It is also shown that this technique can be applied during the Mars flyby phase of the Galileo mission. Covariance analyses show that the improvement in flyby navigation

accuracy is significant compared to the conventional ground-based radio metric navigation. Since the achievable navigation accuracy strongly depends on the assumed measurement accuracy, experiments were conducted to evaluate the measurement accuracy: A single experiment using Viking Lander and Orbiter was viable for this purpose and the data reductions indicate that the expected accuracy can be attained. Since the Viking Orbiter is no longer operational, future experiments will be conducted using Voyager 1 and 2 spacecraft to increase the confidence level of the measurement accuracy.

This analysis has shown that the use of beacons for interplanetary navigation, specifically for target related navigation, will be of significant value. Thus, the utilization of interplanetary beacons should be a part of the next generation navigation technology development program.

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Appendix A

Viking Lander/Orbiter Data Processing

The experiment that was conducted on June 28, 1980 employed the Deep Space Stations at Goldstone, California (DSS 14) and Madrid, Spain (DSS 63). The time sequence of data acquisitions is given in Fig. 6. The Orbiter was first tracked by DSS 63. Two-way coherent doppler and range measurements were obtained. Subsequent to this, DSS 14 was used to acquire the Lander, and range and doppler measurements were generated. After 13 minutes of data acquisition, DSS 14 handed over to DSS 63 tracking of the Lander. Both range and doppler measurements were obtained for about 13 minutes. Subsequent to this, DSS 14 was used to track the Orbiter. This completed the cycle with four independent tracking links.

From tracking the Lander for an extensive period of time, the Lander location with respect to the dynamic center of Mars has been determined very accurately (Ref. 8). The current uncertainty of the Lander location is about 300 m. The Orbiter position can be determined by processing doppler tracking data. With sufficient data, the Orbiter position can be determined with an accuracy better than 10 km. Thus, the Lander relative position of the Orbiter is known to better than 10 km, and doubly differenced range measurement residuals can be generated and the measurement accuracy can be evaluated.

The Orbiter position (state vector) is determined by processing the coherent two-way doppler data. This is accomplished by estimating state parameters (position and velocity components of the Orbiter) only. However, the low-order gravity coefficients are included in the trajectory model equations. The best fit to the orbit of the Orbiter is obtained when continuous tracking data over the whole orbit, except for a couple of hours near the periapsis, are available. When the periapsis data are available, these are often deleted out to desensitize the effects of unmodeled higher degree coefficients of the gravity field. In this experiment, although there were sufficient tracking data, the doppler data were not continuous throughout the orbit. The tracking data (doppler) were available from the Australian station (DSS 43) about one hour after periapsis, for about 2.75 hours. Then there was a gap of about 1.75 hours without any data. After this, the Spain station (DSS 63) provided the data for about 6.5 hours. Then there were no tracking data for about 7.5 hours. Once again DSS 43 provided about three hours of doppler data. Even though the tracking data were not available continuously over the orbit, previous studies (Ref. 9) have shown that the accu-

racy degradation is relatively small if the available data are distributed over the orbit.

The postfit residuals from the Orbiter data are shown in the Fig. 7. On both sides of the periapsis, the data are taken with a frequency of one measurement every minute. The rest of the data are with a frequency of one measurement every 10 minutes. The data with both sample sizes are weighted accordingly with a data weight of 15 mHz with 1-minute averaging time, in the data reduction process. The standard deviation of the postfit data residuals is less than 10 mHz indicating that the residuals do not contain any orbit related systematic signature.

As discussed previously, the doubly differenced range data are generated by explicitly differencing two differenced range data points from two different sources. In this case, one of the sources is the Viking Orbiter. Thus, it is necessary to examine the residuals of the differenced range points of the Orbiter. This is accomplished simply by generating range residuals of the data from both stations. The range residuals are obtained by passing the raw range measurements through the best fit orbit. The range residuals have to be adjusted for (1) ground station calibration, (2) transponder delay, and (3) the media effects.

A discussion of ground station calibration is given by Komarek and Otoshi (Ref. 10). The calibration for media effects is achieved by adjusting for the troposphere effects using a troposphere table based on pressure and temperature models for the stations as a function of elevation angle (Ref. 11). Ionospheric effects are calibrated either by using Faraday rotation data or multifrequency data from the spacecraft. Faraday rotation data have been used here. There is a modeling error in the Faraday rotation calibration for the ionospheric effects primarily due to the mapping of the Faraday rotation data to the line of sight of the spacecraft. However, this error is significantly less than the estimated accuracy of this new data type.

The range residuals are shown in the Fig. 8. The residuals are expressed in meters. A range bias of 13.5 m is observed between DSSs 63 and 14.

The Lander data is processed in a manner similar to the Orbiter range data, however, the Lander location is known a priori, and the parameters related to the Lander location are given in Table 3. The Lander range residuals are generated by

differencing the computed values of range points based on assumed models from the observed range points. The range residuals, after appropriate calibration is applied, are shown in Fig. 9. It is clear from this figure that there exists a bias of about 10.1 m between DSSs 63 and 14 in the same direction as observed in the case of the Orbiter data. When the doubly differenced range observable is formed, these two biases are also differenced and the resulting bias is about 3.4 m and this represents the measurement accuracy of this data type.

Both Orbiter and Lander range residuals (Figs. 8 and 9) show that the scatter within a single station measurement is considerably small (< 1 m) as predicted by theoretical error budget. This scatter is mostly due to the system noise, depending on the spanned-bandwidth of the ranging code. The range bias between two stations is often introduced by the range calibration error. Errors in both ground station calibration and media calibration can cause the 3.4-m bias. Theoretically, this

bias is expected to cancel when two spacecraft data are differenced. However, the cancellation due to media errors is a function of the spatial separation of the two spacecraft and the time separation in the data acquisition. In this experiment, the spatial separation is negligible (< 0.1 deg). However, the data acquisition time separation between the Orbiter and the Lander is over an hour. Thus, a part of the residual bias may stem from temporal changes in media. Cancellation of errors due to ground station calibration is achievable if the same instrumentation configuration is utilized in acquiring both Orbiter and Lander ranging data. In this experiment, although most of the instrumentation used in tracking the Orbiter and the Lander is the same, different receivers were employed. Thus, it is possible that the remaining residual bias in the doubly differenced data is due to the error in the calibration values and this is consistent with the currently expected calibration accuracy. More doubly differenced range data is required to understand its noise characteristics.

Table 2. Viking Lander direct link ranging opportunities for Galileo navigation, 1984

| Point | DSS Baseline ^a | Date |
|-------|---------------------------|-------------|
| 1 | 63-14 | 22 March 84 |
| 2 | 63-14 | 23 March 84 |
| 3 | 63-14 | 24 March 84 |
| 4 | 63-14 | 25 March 84 |
| 5 | 63-14 | 26 March 84 |
| 6 | 14-43 | 29 March 84 |
| 7 | 14-43 | 30 March 84 |
| 8 | 14-43 | 31 March 84 |
| 9 | 14-43 | 1 April 84 |
| 10 | 14-43 | 2 April 84 |
| 11 | 14-43 | 3 April 84 |
| 12 | 14-43 | 5 April 84 |
| 13 | 14-43 | 6 April 84 |
| 14 | 43-63 | 14 April 84 |
| 15 | 43-63 | 15 April 84 |
| 16 | 43-63 | 16 April 84 |
| 17 | 43-63 | 17 April 84 |
| 18 | 43-63 | 18 April 84 |
| 19 | 63-14 | 25 April 84 |
| 20 | 63-14 | 26 April 84 |
| 21 | 63-14 | 27 April 84 |
| 22 | 63-14 | 29 April 84 |
| 23 | 63-14 | 30 April 84 |
| 24 | 63-14 | 1 May 84 |
| 25 | 14-43 | 4 May 84 |
| 26 | 14-43 | 6 May 84 |
| 27 | 14-43 | 7 May 84 |
| 28 | 14-43 | 8 May 84 |
| 29 | 14-43 | 9 May 84 |
| 30 | 14-43 | 10 May 84 |
| 31 | 14-43 | 11 May 84 |
| 32 | 43-63 | 17 May 84 |
| 33 | 43-63 | 18 May 84 |
| 34 | 43-63 | 20 May 84 |
| 35 | 43-63 | 21 May 84 |
| 36 | 43-63 | 22 May 84 |
| 37 | 14-43 | 23 May 84 |
| 38 | 63-14 | 27 May 84 |
| 39 | 63-14 | 29 May 84 |
| 40 | 63-14 | 30 May 84 |
| 41 | 63-14 | 31 May 84 |
| 42 | 63-14 | 1 June 84 |
| 43 | 63-14 | 2 June 84 |

^aDSS 14 (Goldstone, California)
DSS 43 (Woomera, Australia)
DSS 63 (Madrid, Spain)

Table 1. Doubly differenced range data error budget

| Source | Error (1 σ), cm |
|---|-------------------------|
| Instrumentation: | |
| Station clock stability (15 min) | 4 |
| Station delay calibration | 0 |
| SNR (Thermal noise; (S-band with 2-MHz spanned band width)) | 200 |
| Waveform distortion | 88 |
| Spacecraft delay | 28 |
| Media | |
| Troposphere (25-deg elevation) | 20 |
| Ionosphere (25-deg elevation) | 6 |
| Solar wind | 15 |
| rss: ~222 cm | |

Table 3. Error analysis parameters

| Parameter | A priori σ |
|--|---|
| Galileo state | $\sigma_x = \sigma_y = \sigma_z = 10^7$ km $\sigma_{\dot{x}} = \sigma_{\dot{y}} = \sigma_{\dot{z}} = 100$ km/s |
| Mars ephemeris | $\sigma_{\text{radial}} = 10$ km, $\sigma_{\text{intrack}} = 40$ km $\sigma_{\text{out of plane}} = 70$ km |
| Station locations | $\sigma_{\text{long}} = 3$ m, $\sigma_{r_s} = 1.5$ m $\sigma_{r_z} = 15$ m |
| Viking Lander locations | $\sigma_x = 10.0$ m $\sigma_y = 40.0$ m $\sigma_z = 300.0$ m |
| Mars mass | $\sigma = 0.1 \text{ km}^3/\text{s}^2$ |
| $r_s =$ distance off the spin axis | |
| $r_z =$ distance off the earth equator plane | |

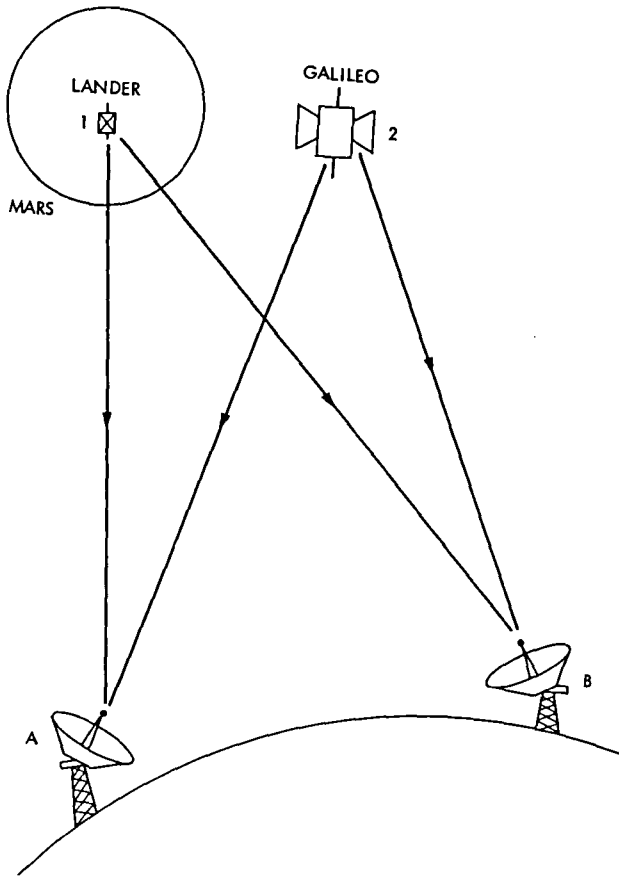


Fig. 1. Doubly differenced range data links

| | | | | | | | | |
|-------|-------|-------|-------|-------|-------|-------|-------|---------|
| 43/63 | 63/14 | 14/43 | 43/63 | 63/14 | 14/43 | 43/63 | 63/14 | |
| -80 | -70 | -60 | -50 | -40 | -30 | -20 | -10 | E(DAYS) |
| 3/14 | 3/24 | 4/3 | 4/13 | 4/23 | 5/3 | 5/13 | 5/23 | 6/2 |
| 1984 | | | | | | | | |

Fig. 2. Lander/Galileo viewing opportunities

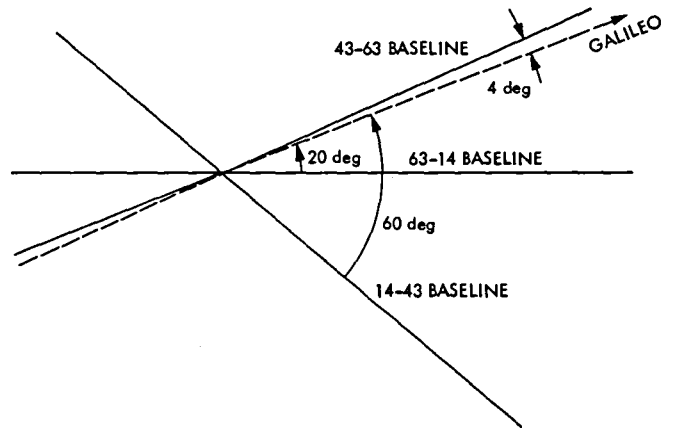


Fig. 3. Deep Space Station baselines projected in the plane normal to the Galileo approach asymptote at the Mars encounter

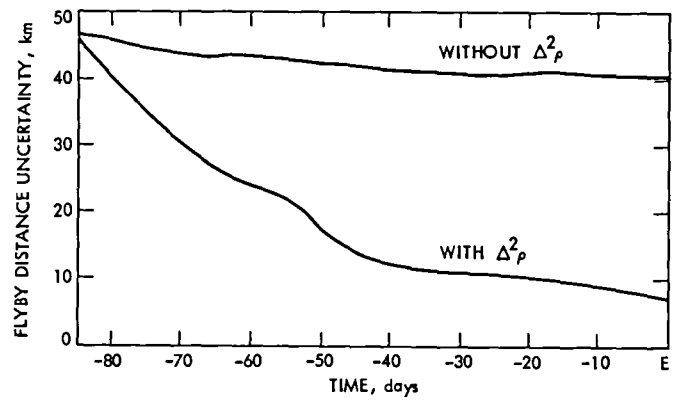


Fig. 4. Covariance analysis results

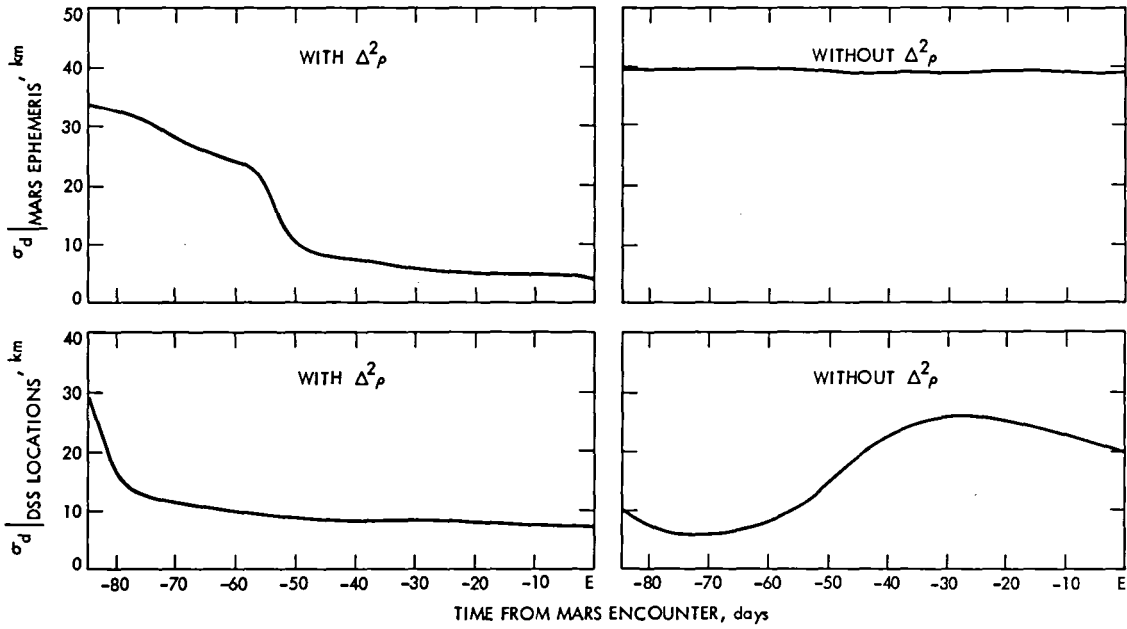


Fig. 5. Reduced sensitivity consider parameters

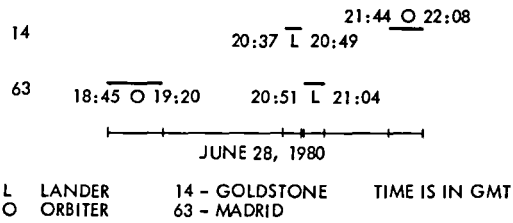


Fig. 6. Lander/Orbiter tracking sequence

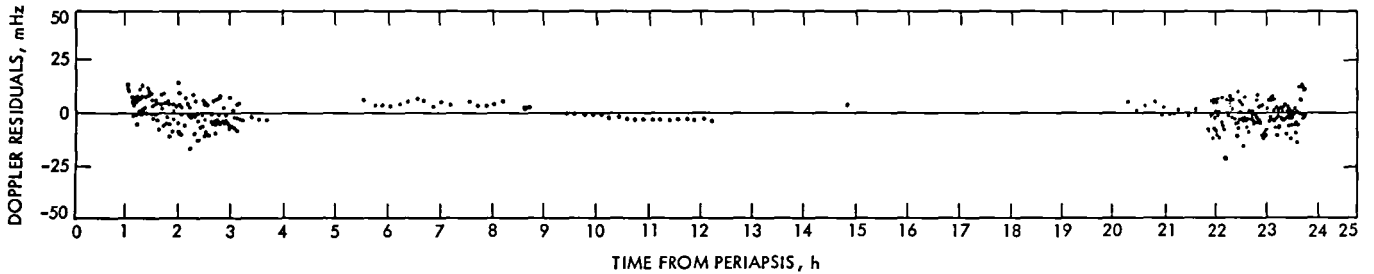


Fig. 7. Orbiter doppler data postfit residuals

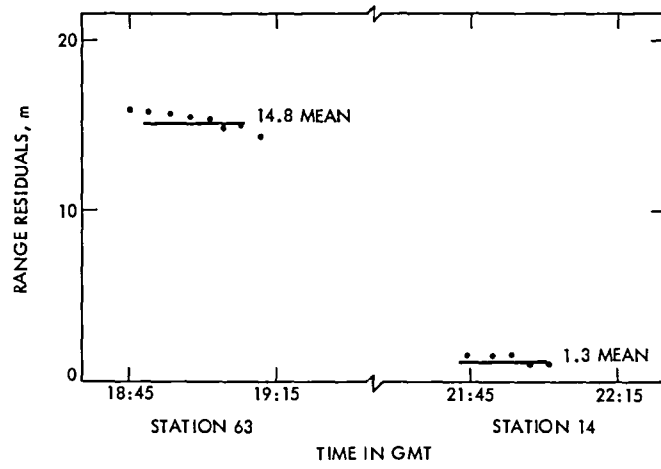


Fig. 8. Orbiter range residuals

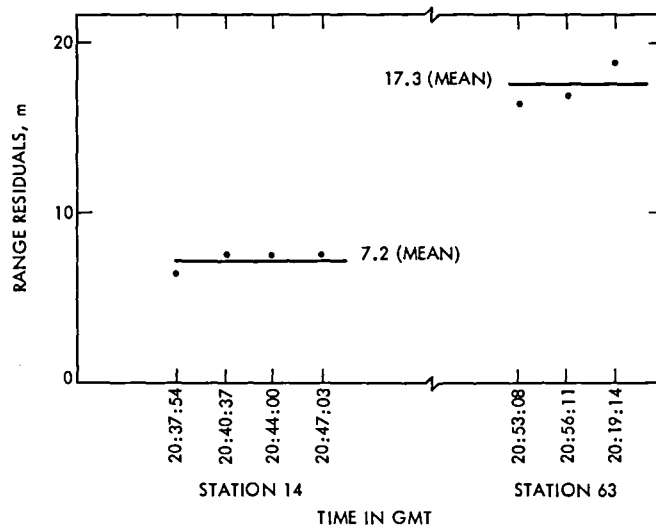


Fig. 9. Lander range residuals