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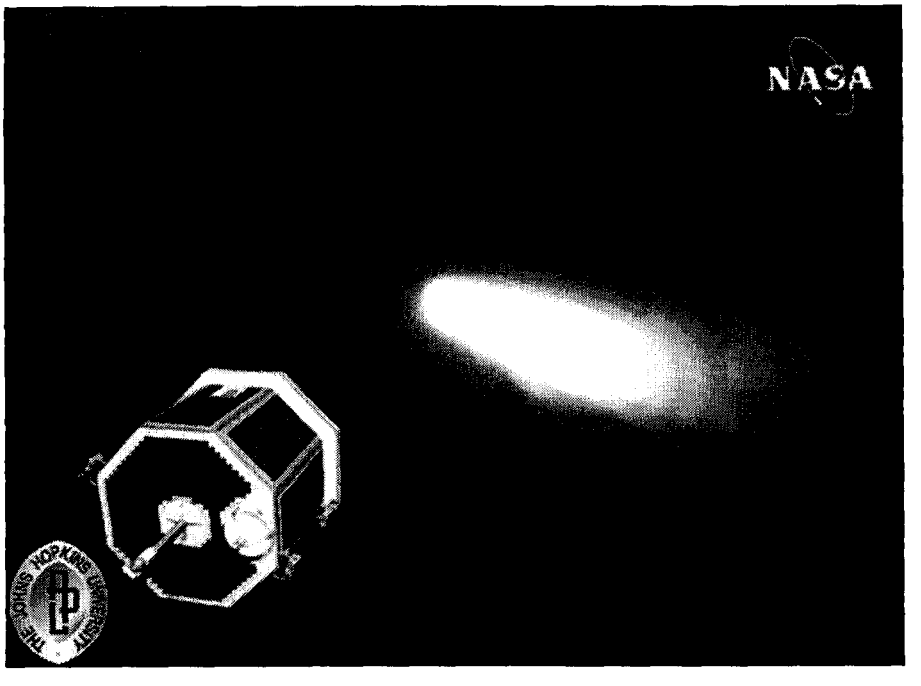


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# NAVIGATING CONTOUR USING THE NONCOHERENT TRANSCEIVER TECHNIQUE

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The successful navigation of the Comet Nucleus Tour spacecraft was performed at the Jet Propulsion Laboratory and was conducted with the use of the new noncoherent transceiver technique developed by the Applied Physics Laboratory. Descriptions of the mission and trajectory are provided, followed by a summary of the challenges to navigation. After launch, about six weeks of noncoherent tracking data were acquired while the spacecraft was in the initial phasing orbits about Earth. Unfortunately, radio contact with CONTOUR could not be re-established after the solid rocket motor (SRM) burn that sent the spacecraft onto its interplanetary trajectory. Following the SRM, ground based optical measurements indicated the spacecraft had broken into pieces and was presumed lost. Discussions include the conditioning performed on the 2-way noncoherent Doppler data, the orbit determination process, and the post SRM trajectory reconstruction.

## INTRODUCTION

The COmet Nucleus TOUR (CONTOUR) spacecraft was launched on July 3, 2002 from Cape Canaveral, Florida. CONTOUR was the sixth mission flown in the National Aeronautics and Space Administration's Discovery Program, a program which aims at achieving highly focused planetary science investigations at a low cost. The objective of the $159 million CONTOUR mission was to conduct scientific flyby studies of comets, Encke and Schwassmann-Wachmann 3, with the option of changing targets in-flight or visiting an additional target after the Schwassmann-Wachmann 3 encounter. The flyby science goals included the following: image nucleus parts at a resolution of 4 m/pixel to reveal details of morphology and comet processes; determine nucleus size, shape, rotation state, albedo/color heterogeneity and activity; map composition of nucleus surface and coma; obtain detailed compositional measurements of gas and dust in the near-nucleus environment; and assess the level of outgassing[1].

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This paper will describe the mission's planned trajectory, the navigation challenges, as well as the navigation tasks that were involved in achieving mission objectives. The challenges include transceiver issues, orbit determination of the spacecraft during the six-week Earth orbit phase, maneuver design, and post solid rocket motor burn reconstruction.

## TRAJECTORY AND SPACECRAFT DESCRIPTIONS

CONTOUR was designed, built, operated and managed by the Johns Hopkins University's Applied Physics Laboratory (APL), while the California Institute of Technology's Jet Propulsion Laboratory (JPL) navigated it. The Deep Space Network (DSN), managed by JPL, provided spacecraft telecommunication support. APL successfully launched its spacecraft aboard a three-stage Boeing Delta II rocket on July 3, 2002 at 02:41 A.M. (EDT). It was placed into a highly eccentric Earth parking orbit with an apogee radius of 115,000 km, a perigee radius of 6,620 km (240 km in altitude), an inclination of 30.5 degrees, and an orbital period of 1.73 days (see Fig. 1 & 2). CONTOUR remained in this parking orbit from launch until August 15, 2002, when it fired its STAR-30 solid rocket motor (SRM) to leave Earth orbit and enter a heliocentric orbit. CONTOUR was to remain in this heliocentric orbit through an Earth flyby in the summer of 2003, followed by a flyby of its first target (comet Encke) in November of the same year. However, due to an anomaly during the SRM burn, radio contact with the spacecraft could not be re-established and it was presumed lost.

CONTOUR was designed in the shape of an octagon cylinder, approximately 3 m in diameter and 2 m in height, and had a 1 m mast located at the center of its aft end (see Fig. 3). Its total weight was approximately 970 kg; this includes a dry weight of 397 kg, 70 kg of Hydrazine fuel, and 503 kg for the STAR-30 SRM[1]. (The SRM was built into the spacecraft; it was not a stage that could be jettisoned.) The spacecraft's sides and aft end were covered with solar array panels. A seven-layered dust shield covered the craft's front end; its purpose was to protect CONTOUR and the science instruments from dust particles during the comet flybys. CONTOUR had four science instruments on board: the CONTOUR Remote Imager/Spectrograph (CRISP), the CONTOUR Forward Imager (CFI), the CONTOUR Dust Analyzer (CIDA), and the Neutral Gas Ion Mass Spectrometer (NGIMS). CONTOUR also carried four antennas, three on the aft end and one on the forward end. The Aft Low Gain and Pancake antennas were located on the mast, which was on the spacecraft spin axis, while the High Gain antenna was located 50 cm from the spin axis on the body of the craft's aft end; the Forward Low Gain antenna was located 40 cm from the spin axis on the front end. Each antenna used the noncoherent transceiver system for communications; the system used one of APL's long line of ultra-stable oscillators[2]. Figure 4 shows CONTOUR's oscillator specifications.

## NAVIGATION CHALLENGES

The greatest technical challenge to navigation was posed by the adoption of the noncoherent navigation technique by the project. Since this was the first operational use of the technique, it had to be validated before launch, and new software and procedures had to be incorporated into the orbit determination process for post-launch operations. Navigation usage of both Doppler and ranging data were affected by this technique.

CONTOUR was a low-cost interplanetary missions flown by NASA. The small navigation team had a full-time equivalent staffing of 5 people from six months before launch until 3 months after launch. Two of these people were dedicated to the noncoherent transceiver technique, leaving 3 to deal with the traditional navigation activities of preparation for launch and operations after launch. This consequently delayed or even prevented some pre-launch analysis; e.g., there was no covariance analysis done for the orbit phase, which is a departure from the normal navigation practice. It also resulted in late operational validation of the non-coherent navigation technique on a previously launched spacecraft (TIMED[3]). It was also a factor in the late development and incomplete delivery of navigation interfaces from APL, the last delivery being made only a few days before launch, leaving little time for testing the small forces and attitude predicts interfaces.

## NONCOHERENT TRANSCEIVER

CONTOUR was the first interplanetary spacecraft to use a transceiver instead of a transponder. This resulted in a number of departures in the task of obtaining Doppler and range data for use in orbit determination. To use transceiver-based tracking data, new procedures and software were established and exercised prior to launch[4,5].

The primary data type was 2-way X-band noncoherent Doppler received by the Deep Space Network (DSN) almost continuously during the Earth phasing orbits. Because noncoherent Doppler is usually much too inaccurate for navigation because of spacecraft oscillator frequency drift, this data required correction in order to be usable in the standard JPL orbit determination process. The correction process was developed by APL, and used both non-coherent Doppler and spacecraft telemetry data to obtain nearly coherent 2-way Doppler data; therefore, the successful navigation of the CONTOUR spacecraft relied not only on the radiometric data, but on the telemetry data, as well. (This later dependence was a first in deep space navigation.) The corrected Doppler data were treated as equivalent to coherent, 2-way Doppler data, the standard data type used in navigation. The details of this correction are described in detail in a paper written by coauthor Dr. J. R. Jensen[6].

The use of a transceiver onboard the spacecraft meant that the downlink frequency, driven by the onboard oscillator, was independent of the uplink frequency. This independence introduced errors in the 2-way Doppler data, which were caused by bias and drift of the spacecraft frequency reference. In order to support radiometric accuracy requirements, CONTOUR carried additional hardware that, in essence, measured the difference between

the received uplink frequency and the transmitted downlink frequency and placed these measurements into the telemetry. On the ground, the correction software then used this information to convert the Doppler observables to those that would have been obtained had a transponder been used[2]. Figure 5 shows both the corrected and uncorrected 2-way Doppler observables for a pass that occurred toward the end of July 18, 2002 UTC (Day Of Year 199); the X represents the uncorrected observable and the O represents the corrected observable. Although it is difficult to see in this pass, the corrected data has a near-constant offset of 3 Hz from the uncorrected data after DOY 199.8602. Prior to this, it can be seen that the corrected data has a sharp positive slope. This is caused by the uplink frequency sweep the station performs when attempting to acquire the spacecraft. The uncorrected observable is not affected during the sweep because the downlink frequency observed is independent of the uplink signal. This typical figure validates what is expected from the noncoherent transceiver system. Approximately 96% of all 2-way noncoherent Doppler data was corrected during flight.

The ranging system used by the DSN was originally designed to use coherent Doppler data to "rate aid" the correlation of the received ranging modulation code with the local code. Without rate aiding, the received and local waveforms drift with respect to each other and the correlation fails. In order to achieve correlation using CONTOUR's noncoherent transceiver system, it was necessary to continually ramp the uplink frequency from the DSN so that the signal arriving at the spacecraft was pre-compensated for the Doppler shift on the uplink leg. Thus, the spacecraft oscillator's downlink frequency approximated the frequency that would result from a transponder-equipped spacecraft receiving the same ramped signal and turning it around to the downlink. This technique required precise control of the uplink frequency. It introduced a vulnerability to small frequency errors in the uplink signal or the spacecraft oscillator, which resulted in failed correlations whenever the received and local waveforms drifted more than 1/4 wavelength relative to each other.

In order to minimize the potential for range correlation errors due to frequency prediction errors, a nonstandard set of ranging tones were used during the first half of the mission. (See Table 1.) This choice of ranging components had the unexpected effect of causing errors in the sequential ranging assembly (SRA) and the loss of 50% of the range measurements. (See Fig 6, this plot shows the ranging anomalies associated with miscorrelations. The "good" (useable) residuals are near 0, while the anomalous residuals due to miscorrelations are spread over several million meters in distinct layers.) This problem was eliminated later when a standard set of components were used. The SRA correlation errors were due to the choice of ranging tones employed, rather than directly due to the use of a transceiver to support range measurements. The source of these correlation errors is not yet understood. Following the change of range components, 97% of the range measurements were good, except during one interval following the spin-up from 20 to 60 rpm where a change in the spacecraft oscillator that was not included in the uplink predictions caused range correlation errors.

## ORBIT DETERMINATION MODELING

Orbit Determination (OD) for CONTOUR was performed using corrected 2-way X-band noncoherent Doppler and 2-way noncoherent ranging data collected by the DSN's 34 m network. Doppler data were collected and corrected at 1-second intervals, then compressed to 60 second intervals (except on launch day, when 1-second data were used). Ranging data were collected at sample rates of 29 and 37 seconds, depending on the specific ranging system parameter values used. When using the compressed data during flight, the 2-way Doppler data were weighted at a sigma from 2 mm/sec to 10 mm/sec and the range data were weighted at a sigma from 5 m to 100 m. Invalid data were removed from both data sets, and both sets were calibrated for media effects (troposphere and ionosphere). Additionally, the Doppler data were calibrated for spin-polarization biases, and the ranging data were calibrated for the timing delay at the spacecraft. The orientation of the spacecraft's - Z-axis (the spin pole) relative to each DSN station was modeled but the spin signature was not removed from the Doppler tracking data. (CONTOUR spin rate varied from 20 to 60 rpm)

The dynamic models used in determining the CONTOUR orbits were the Newtonian point-mass model, the relativity model, the Earth and Lunar oblateness and solid tide models, the impulsive maneuver model for unintentional delta-V events, the finite maneuver model for intentional delta-V events, the atmospheric drag model, and the solar radiation pressure model. The Newtonian point-mass model computed the gravitational acceleration of the spacecraft due to the nine planets, the Sun and the Moon by treating the bodies as point-masses. The relativity model computed the relativistic perturbative acceleration caused by the Sun. The oblateness and solid tide models computed the acceleration of the spacecraft due to Earth and Moon's oblateness and the tidal acceleration effect of the Earth and Lunar solid tides caused by the Sun, respectively. The impulsive maneuver model was used to model unintentional delta-Vs (e.g., incidentally caused by turns), and the finite maneuver model was used to model intentional delta-Vs, i.e., the Orbit Change Maneuvers (OCMs) and the Solid Rocket Motor (SRM) maneuver. The atmospheric drag model computed the deceleration of CONTOUR caused by Earth's atmosphere. The solar radiation pressure models computed the acceleration of the spacecraft due to solar radiation.

The Earth gravity model used in the OD was the JGM-3 gravity model[7], truncated to the 50$^{th}$ degree and order. The Earth atmospheric model used for drag was the Jacchia-Roberts atmosphere density model[8]. This model was used in place of the DTM model[9,10], which was used for initial studies, because the former model was built into certain software APL used. Figure 7 shows a comparison between the Jacchia-Roberts model and the DTM model during the first perigee pass of CONTOUR; it can be seen that both models show reasonable agreement to each other. The figure also shows the large drag pressure exerted on the spacecraft over the brief period CONTOUR passed through perigee. The CONTOUR spacecraft shape model for drag was basic and was composed of three components: (1) a 1-sided flat plate for the aft side (the side that had the high

gain antenna), (2) an open-ended cylinder to represent the sides of the spacecraft, and (3) a 1-sided plate for the forward side (the side with the SRM nozzle).

Other models that were used in the OD process include the solid Earth tide correction model, the continental plate motion and ocean loading models, the precession and nutation models, and the relativistic light time correction model. The precession and nutation models used the JPL Earth Orientation Parameters[11]. The Earth-fixed coordinate system used in the OD was consistent with the International Earth Rotation Service (IERS) terrestrial reference frame labeled ITRF93[12]. The locations of the DSN stations were consistent with the ITRF93 reference frame[13]. The ephemerides of the Sun, the Moon, and the planets were defined by the JPL DE405 planetary ephemeris[14,15]. The inertial coordinate system used for orbit integration was Earth centered, Earth mean equator and vernal equinox system at J2000. The *a priori* sigma values for each component of the spacecraft's position were 100 km and 0.1 m/sec for the spacecraft's velocity components. See Table 2 for a listing of *a priori* sigma values for other parameters estimated in the filter, as well as a listing of the consider parameters.

The software used to process the tracking data was JPL's Orbit Determination Program (ODP)[16]. The ODP solves for spacecraft position, velocity, and other requested parameters using a square root information (SRIF) weighted least squares filter[17,18]. The Doppler data corrections and the orbit fits were performed on a 700 MHz Intel Pentium II workstation running RedHat Linux 6.1.

## ORBIT DETERMINATION RESULTS

The concern for launch was that the new noncoherent transceiver technique would either not work at all, or that there would be significant but subtle undetected biases in the corrected data, or that the correction process would encounter unplanned delays during the first critical pass over Goldstone, and consequently there would be no useful coherent Doppler data available for early orbit determination. Early OD deliveries were to begin at Launch + 2 hours (one hour into the initial Goldstone pass) in order to support acquisition at the next DSN track at Canberra beginning at L + 7 hours. Failure to deliver an accurate orbit could have meant a loss of the Canberra pass, and perhaps serious difficulties acquiring the spacecraft after that if there were launch anomalies. Another OD delivery was needed no later than L + 11 hours to support a decision to do a periapsis raise maneuver at the first apoapsis (L + 22 hours) to prevent losing the spacecraft in the event of a serious launch error.

Since coherent Doppler is the workhorse data type of JPL navigation during launch phase, it was necessary to put more emphasis on other data types in case of its failure. The other data types would either replace the corrected Doppler data if necessary, or serve as independent validation of it. Ranging would ordinarily be the next data type of choice, but the same feature that made it necessary to correct the Doppler data (i.e., noncoherence) unfortunately also put the ranging data at risk because of the trajectory-

induced uncertainty in the frequency that the DSN needed to transmit in order to insure that the noncoherent ranging technique (without the conventional rate aiding) would work. Figure 8 shows the a priori errors in the knowledge of the spacecraft downlink frequency. In order to achieve valid ranging, the DSN would have to transmit at a frequency which, after being Doppler shifted on the uplink and turned around at the spacecraft by the same ratio that a transponder would have used, would be within a given error of the actual downlink frequency. In the case of the DSN ranging setup used (and optimized) for launch, the maximum allowable frequency error would be about 1300 Hz. Comparing this to the limits in Figure 8 indicates that there was a reasonable (but not excellent) chance that ranging would be usable beginning about two hours after launch, but that the conditions would then worsen later in the pass unless the actual downlink frequency was determined and the DSN changed its uplink frequency accordingly.

Only one DSN data type would be totally unaffected by noncoherence during launch, and that was the tracking angle data of the acquisition antennas, one of them at Goldstone and one at Canberra. These data are used only at launch, and are not as powerful as coherent Doppler and ranging, so there was an accuracy issue as well as a reliability issue associated with their use.

Fortunately, on launch day the Doppler correction technique worked without suffering a long processing delay. Ranging also appeared mostly successful, as did the angle data. Experiments were tried in near real-time with all three data types, individually and in combinations, and the solutions were consistent with each other to within the errors expected of each type, so that the worry of a bias being introduced into the Doppler by the correction process was alleviated.

The next major concern for launch was over the signature modulated onto the Doppler data by the spin of the spacecraft in combination with the offset of the forward low gain antenna from the spin axis. The forward low-gain antenna was in use from the initial Goldstone acquisition into the beginning of the first Canberra pass (from about L + 1 to L + 8.6 hours). After this initial period of forward low-gain antenna use, most of the rest of the orbital phase (except for short periods near periapsis) was flown on either the pancake antenna or the aft low-gain antenna, both of which had no nominal offset from the spin axis, and thus induced no nominal spin modulation on the Doppler.

The initial spin of the spacecraft was 49.5 RPM, a much higher rate than has been seen by JPL navigators on any other spacecraft. At about L + 2.5 hours, the spacecraft was spun down to 22.8 RPM, still faster than ever seen by JPL navigators. At L + 13 hours (by which time the spacecraft had switched to the pancake antenna), the rate was cleaned up to 20 RPM, and left close to that rate for most of the rest of the orbit phase until it was spun up again to 60 RPM in preparation for the SRM burn.

The modulation on the Doppler during the early part of the launch is seen in Figure 9. The modulation as a function of antenna offset ($\rho$), spin axis angle from the tracking station ($\alpha$), Doppler compression time ($\Delta t$), and spin rate ($\omega$ in rad/s) is

$$D = \frac{2\rho \sin\alpha}{\Delta t} \sin\left(\omega \frac{\Delta t}{2}\right) \cos\omega(t - t_0)$$

where $D$ is the 1-way Doppler modulation (in terms of range rate, not Hz). This function explains the larger envelopes at about 0.38 day and 0.42 day when the compression time was changed briefly from 1 second to 0.1 second, and why the envelope widened at 0.4 day when the spin rate actually decreased from 49.5 to 22.8 RPM (i.e., a sampling effect).

The modulation in Figure 9 is quite large, exceeding half a meter per second after 0.4 day. The best way of treating this data is not obvious. One method would be to "de-modulate" it by determining the exact parameter values in the above equation and calibrating the signature out. The danger is that if the rate is not known precisely, a large low-frequency signature could be left in the signal, the worst possible situation if the filter should allow it to masquerade as a change in orbital or other parameters. Another way would be to not model the modulation at all, but simply deweight the data to the approximate envelope of the signal, and count on the high-frequency averaging of the filter to provide an accurate mean. This was the method the navigation team used, not because of any stringent analysis, but because it was the easiest, fastest, cheapest way to approach the problem. The worry was that with longer Doppler compression times, such as 1 minute, and short spans of data (e.g., a short DSN pass), odd sampling effects and incomplete averaging might leave subtle undesirable signatures in the data. Fortunately, except for the launch, most of the orbit phase was spent on the aft and pancake low gain antennas so that there was no spin modulation to worry about. Unfortunately, there was never enough time or resources to do a thorough analysis on the best way to treat the spin-modulated data, so that this question may continue to haunt navigators of future spinning spacecraft for which antenna placement is not benign.

The last topic of this section concerns the modeling and estimation of spacecraft drag. The spacecraft periapsis altitude was very low, ranging between a low of 194 and a high of 249 km, and neither the Jacchia-Roberts nor the DTM atmosphere density models are particularly accurate in this range, particularly at the lower end, even with weekly updates of the solar flux inputs. The drag model we used in the navigation software used the same spacecraft components (a flat plate at each end of a cylinder) as the solar pressure model. Because of time and resource limitations, only a drag force was modeled, based on the combined cross-section of the three components, rather than a more complicated model which was available to model lift and side-slip forces. An overall drag coefficient of 2.8 was used. A constant scale factor of the area was estimated to remove bias in the model.

Despite the simplicity of the model, the estimate of the scale factor and the size of the data residuals were quite reasonable through the first two periapses. However, when the data arc was extended from launch through the third periapsis, the residuals no longer were well behaved, and the area scale factor estimate began to take on implausible values. It became obvious that estimating the scale factor as a constant throughout all periapses was the wrong thing to do. The estimation procedure was changed so that the scale factor was estimated stochastically at each periapsis, with zero correlation between periapses. This had the desired effect of improving the residuals and returning the scale factors to believable values, as well as decoupling the orbit-to-orbit error.

## MANEUVER DESIGN

There were 23 spacecraft maneuvers performed during the Earth orbit phase of the mission, which lasted 43 days. They were designed by APL to achieve proper conditions at the Earth departure maneuver. Because of the large DeltaV (1.9 km/s) and the resonant nature of the CONTOUR trajectory, the SRM had to be executed at the exact conditions determined by JPL for optimizing the interplanetary trajectory. Failure to achieve these conditions would require large clean-up maneuvers following Earth departure.

## POST SRM RECONSTRUCTION

The Solid Rocket Motor burn on August 15, 2002 was intended to terminate CONTOUR's Earth orbit phase and inject it into a heliocentric orbit, and was performed at a point where the spacecraft was below the horizon for all DSN stations. The spacecraft was last seen, in apparently good order, by DSS 65 at Madrid at the nominal end of its pass about an hour before the burn. About 40 minutes after the burn, CONTOUR would have risen for DSS 25 at Goldstone and DSSs 34 and 46 at Canberra. Radio contact could not be re-established.

On the evening of August 16, the Spacewatch 1.8 m telescope at Kitt Peak, Arizona observed two objects, denoted A and B, in the general vicinity of the nominal post-maneuver trajectory. Later, in some of the Spacewatch images of the same date, a third object, denoted C, was discovered. Thereafter additional images of all three objects were obtained by Spacewatch, the LINEAR facility in New Mexico, the University of Hawaii's 2.24 m telescope at Mauna Kea, JPL's Table Mountain 1 m telescope in California, and at the Farpoint Observatory's 0.3 m telescope in Kansas. Table 3 is a summary of these optical tracks.

These observations, plus radiometric data from 13 August until the SRM burn, were used by CONTOUR Navigation to estimate separate SRM burns and accelerations due to solar radiation pressure for each of the three objects. These results (along with their 1-sigma uncertainties) are presented in Tables 4 and 5, respectively. It was determined that one of the components, B, is apparently (with high probability) a very low mass object of between 3 and 4 kg. Component A could contain much of the 495 kg of the nominal

spacecraft or be as small as 70 kg for a 1-sigma deviation. There was too little data for C (only on August 16) to determine anything meaningful about its mass. All three objects will return to the vicinity of Earth almost exactly a year after the SRM maneuver. Earth-centered B-plane results for these encounters are given in Table 6, along with their respective 1-sigma uncertainties. Figure 10 shows the pointing uncertainties of component A for ground observers during the cruise back to Earth. Interestingly, the pointing uncertainties show a dip close to the flyby time, implying that it might be possible to find the object with a fairly narrow-field (and sensitive) telescope.

## CONCLUSIONS

The Jet Propulsion Laboratory undertook the task of navigating the CONTOUR spacecraft, which used the successful transceiver system for telecommunications. The use of this system resulted in a number of departures from standard JPL and DSN procedures for obtaining Doppler and ranging data for use in orbit determination. One of the lessons learned from the experience of CONTOUR is that although the transceiver technique was successful, the advantages of the system are in the design and development of the spacecraft, rather than to the navigation process itself. Complications to the navigation process included the requirement to apply and validate the Doppler correction, develop auxiliary file handling software and the potential for loss of data due to a reliance on telemetry. The application of the correction and validation should be eliminated as a navigation team responsibility in future missions that use this technique. If this is done, then the remaining risks due to the potential for a loss of telemetry for a critical navigation delivery must be recognized and balanced against the cost and development advantages of using a transceiver.

It does not appear necessary to model out the spin signature on the Doppler data for a spacecraft like CONTOUR, at least during the orbital phase. Adequate orbit determination was achieved by de-weighting the data to the approximate envelope of the modulation with little degradation of accuracy. It does appear necessary to stochastically model the drag at every periapsis of a CONTOUR-like orbit.

CONTOUR's launch was one of the most uncertain and challenging launches supported by JPL navigation. It was made no easier by the extraordinarily low cost of the mission, which resulted in less rigorous pre-launch analysis than usual by the navigation team due to limited time and resources. Nevertheless, CONTOUR navigation was accomplished adequately, if not excellently, during the orbit phase. However, future projects like this should be aware of, understand, and accept the increase of risk that inevitably accompanies low-cost, bare-bones operations.

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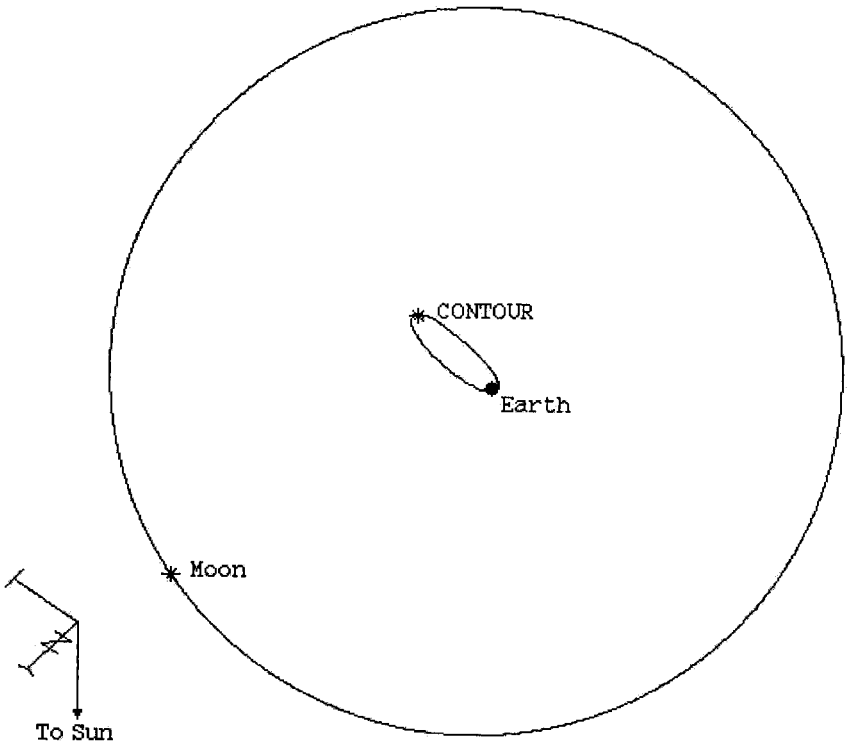



**Figure 1: CONTOUR Orbit as viewed from north ecliptic**

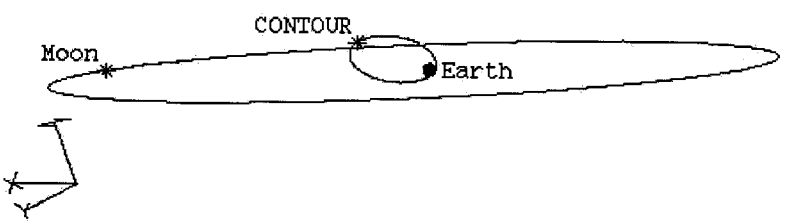



**Figure 2: CONTOUR Orbit as viewed from the Sun**

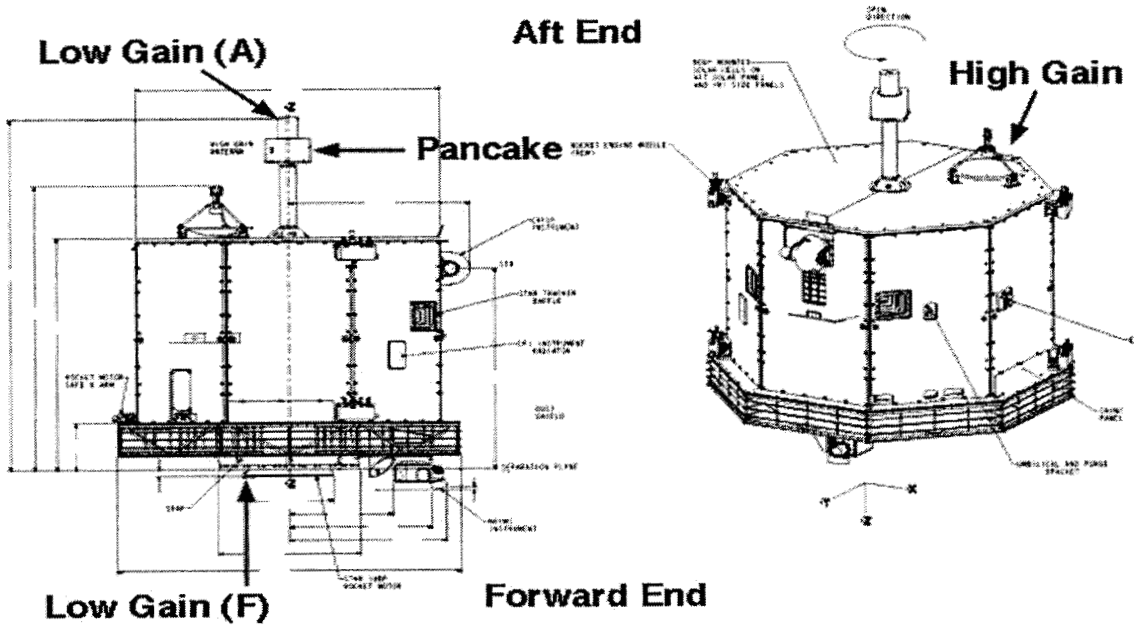
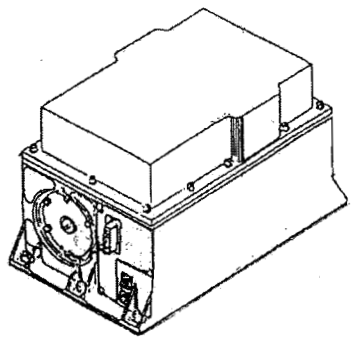



**Figure 3: CONTOUR Spacecraft**



| | |
|---|---|
| Output | Two 10 dBm @ 30.6 MHz |
| Drift Rate | $5 \times 10^{-10}$ per 24 hours |
| Temperature Stab. | $1 \times 10^{-11}$ per °C (-5 to +25 °C) |
| Phase Noise | -90 dBc/Hz @ 1 Hz offset |
| | -130 dBc/Hz @ 1 KHz offset |
| Allan Deviation | $1 \times 10^{-12}$ in 1, 10, or 100 sec. Periods |

**Figure 4: CONTOUR Oscillator**

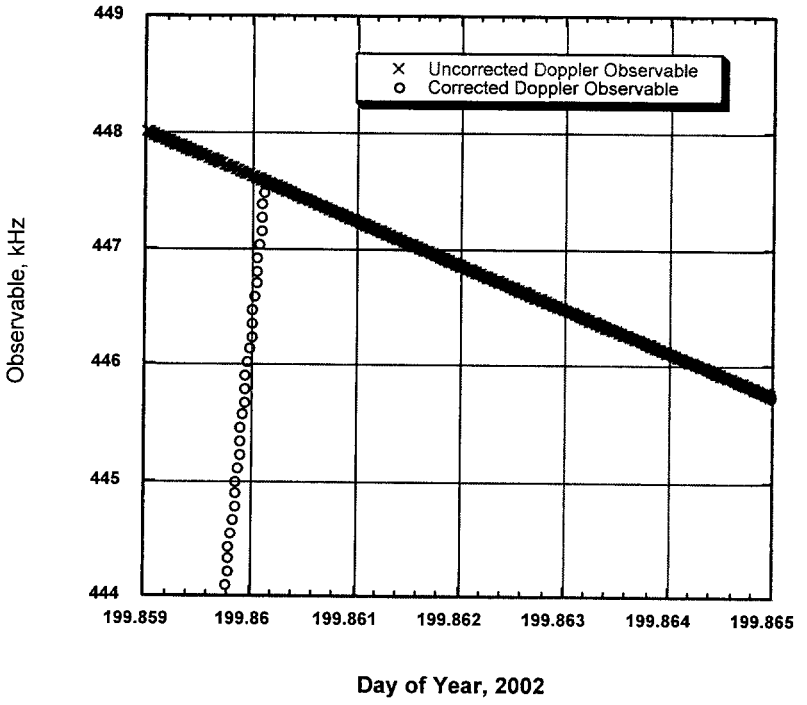



**Figure 5: 2-way Doppler Observables for July 18, 2002**

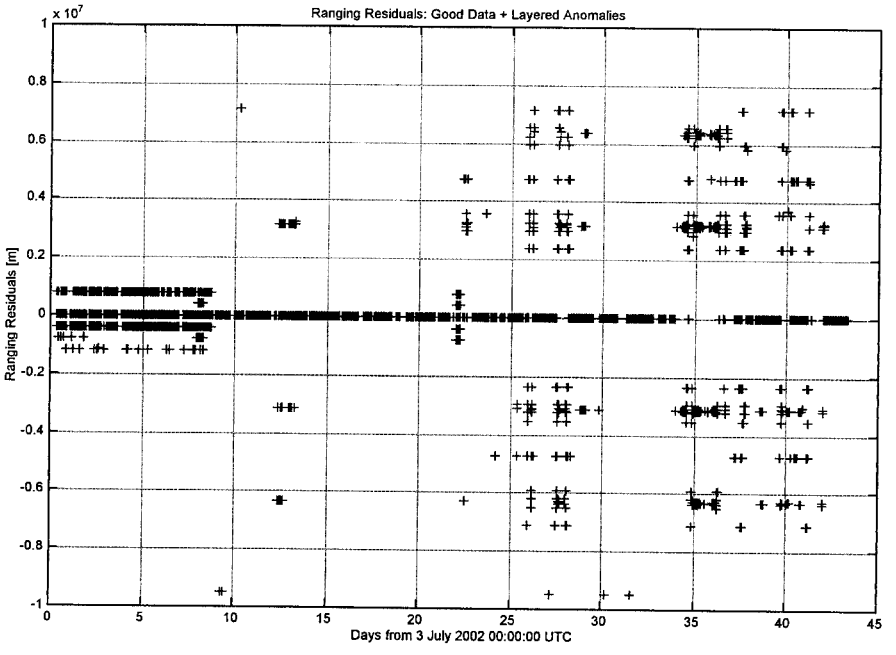



**Figure 6: Miscorrelated Range Residuals**

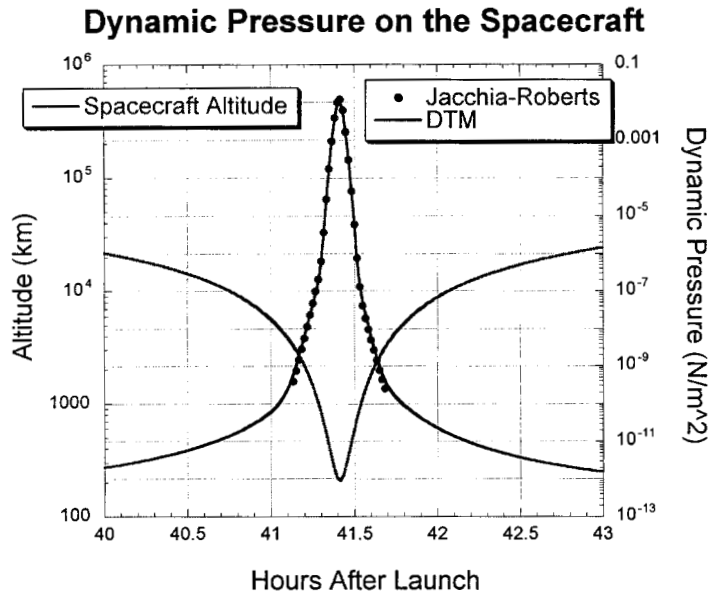



**Figure 7: Dynamic Pressure on the Spacecraft**

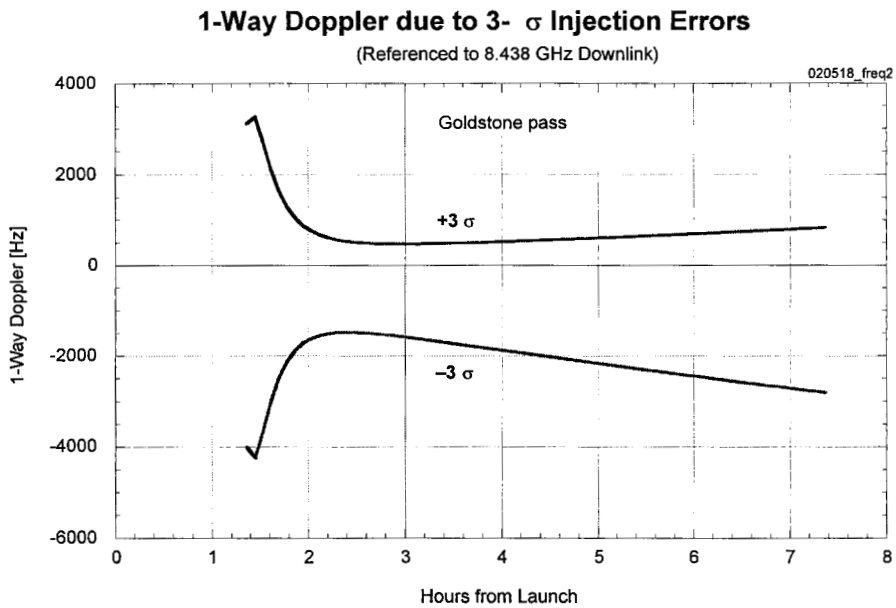



**Figure 8: Errors in Best Ranging Frequency After Launch**

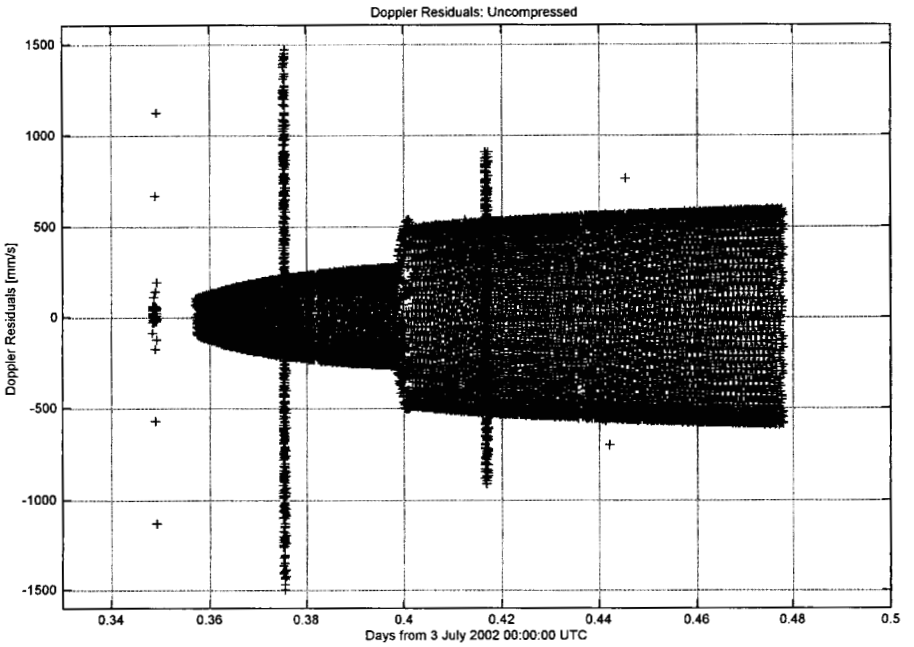



**Figure 9: Doppler Residuals**

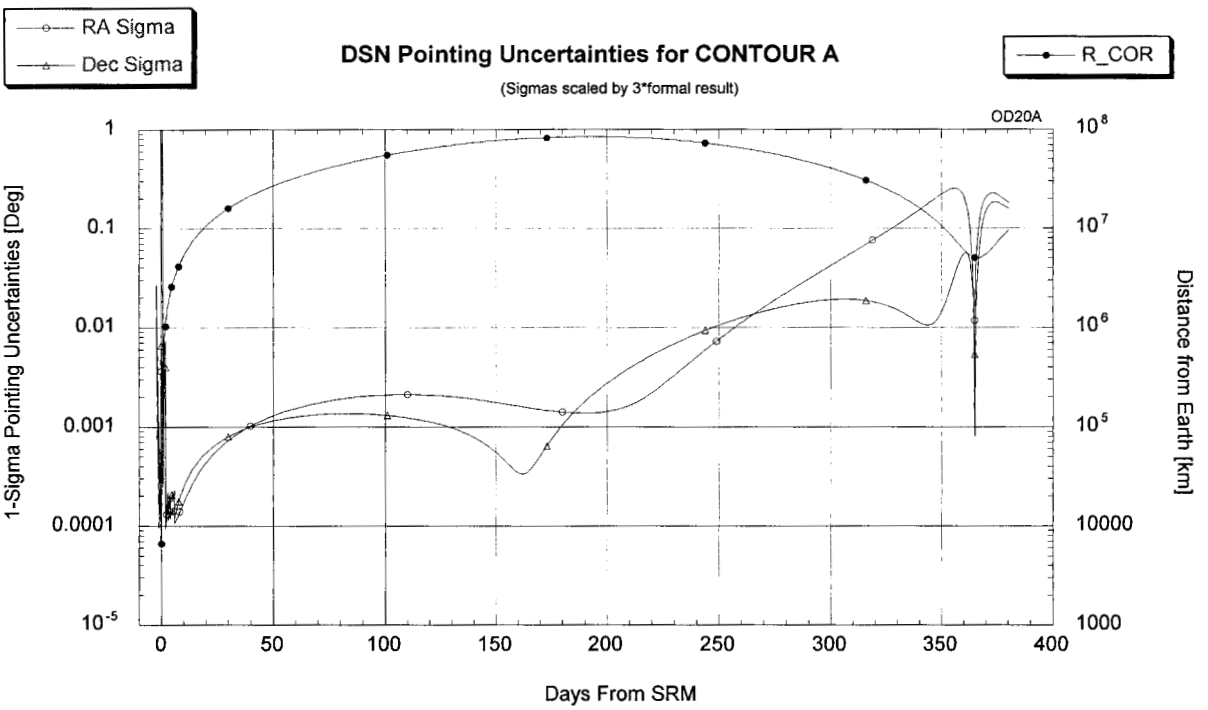


**Figure 10: Pointing Uncertainties**

**Table 1: Ranging Parameter Settings**

| | Launch Settings | After July 11 17:38 |
|---|---|---|
| Clock | 5 | 4 |
| Chop | 8 | 4 |
| High | 17 | 20 |
| T1 (sec) | 2 | 2 |
| T2 (sec) | 1 | 1 |
| Cycle Time (sec) | 29 | 37 |
| Ambiguity (km) | 1187 | 9494 |

**Table 2: Filter Parameters**

| **Parameters** | **A Priori Uncertainty (1-□igma)** | **Comments** |
|---|---|---|
| Estimated as bias: | | |
| Epoch state (6) | 100 km, 0.1 m/s per axis | |
| Solar pressure acceleration (3) | 50% of a priori value for each surface | Scale factors for cylinder and two flat plates (A priori values = 1) |
| Drag coefficients, constant part (3) | 50% of a priori value for each surface | Scale factors for cylinder and two flat plates. (A priori values =1) |
| Impulsive events | 20 mm/s per axis | All thrusting events other than planned maneuvers. (A priori values = 0) |
| Planned maneuvers (3) | 5° RA, 5° Dec, 5% ΔV | |
| Ranging bias, constant part | 5 m | Constant bias, all ranging points. |
| Estimated as stochastic: | | |
| Drag coefficients, stochastic part (3) | 0.5 each surface | Scale factors for cylinder and two flat plates. (A priori values = 0). □ = 0, updated each apoapsis. |
| Ionosphere (day) | 0.05 | |
| Ionosphere (night) | 0.01 | |
| Ranging bias, stochastic part | 2 m | Independent bias each ranging pass. |
| Considered: | | |
| DSN station | 15 cm per axis | Sufficiently conservative to |

| | | |
|---|---|---|
| locations (9) | | cover media errors as well. |
| Earth ephemeris | | |
| Station locations | | |
| Troposphere (dry) | | |
| Troposphere (wet) | | |
| Ionosphere (day) | | |
| Ionosphere (night) | | |
| Polar motion | | |
| UT1 | | |

**Table 3: Summary of Optical Data**

| | **16 Aug** | **17 Aug** | **18 Aug** | **19 Aug** | **20 Aug** | **21 Aug** | **Total** |
|---|---|---|---|---|---|---|---|
| **A** | 2S,2L | 7M,4T | 4M | 2M | 1M | 4M | 26 |
| **B** | 2S,3L | 3M,3T | 4M | 2M | 0 | 0 | 17 |
| **C** | 2M | 0 | 0 | 0 | 0 | 0 | 2 |

S = Space Watch, L = Linear, M = Mauna Kea, T = Table Mountain

**Table 4: Estimated SRM**

| | **CONTOUR-A** | **CONTOUR-B** | **CONTOUR-C** | **NOMINAL** |
|---|---|---|---|---|
| **RA** (deg.) | 259.046 ±0.006 | 258.780 ±0.006 | 259. ±1.0 | 258.731 |
| **Dec** (deg.) | 29.350 ±0.001 | 29.583 ±0.001 | 28.8 ±0.2 | 29.295 |
| **ΔV** (m/sec) | 1849.8 ±0.1 | 1850.3 ±0.1 | 1896. ±20 | 1920.1 |
| **ΔV** (% SRM) | 96 | 96 | 98 | 100 |

**Table 5: Solar Pressure Results**

| | Relative acceleration | 1-sigma uncertainty | Mass limits for 1-sigma errors and 4.3 m^2 area [kg] |
|---|---|---|---|
| **NOMINAL** | 1 | ----- | ----- |
| **A** | 3.6 | 1.0 | 190/108 |
| **B** | 149 | 2.0 | 3.4/3.3 |
| **C** | 2.9 | 100 | 495/5 |

**Table 6: Earth Flyby B-Plane Results**

| | B•R [km] | B•T [km] | Date/Time [UTC] |
|---|---|---|---|
| A | 435,000 ±1,400 | 4,947,000 ±16,000 | 16 Aug 13:47:39 ±330 sec |
| B | 580,000 ±2,700 | 6,985,000 ±32,800 | 17 Aug 01:36:55 ±670 sec |
| C | 93,700 ±169,000 | 721,000 ±1,564,000 | 15 Aug 13:24:41 ± 9.0 hour |