VALIDATION AND OPERATIONS EXPERIENCE RELATED TO ORBIT AND ATTITUDE SOFTWARE OF BHASKARA

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ABSTRACT

Bhaskara is a near-earth orbiting spin stabilised earth observation satellite with two TV cameras and microwave radiometers at two frequencies as primary payload. Orbit and attitude determination software has been developed in house to meet the position accuracy requirement of ± 4 Km and attitude accuracy requirements of 3 deg. for control purposes and better than 1 deg. for post facto data processing.

The software has been validated against actual and simulated tracking data and simulated attitude sensor data. Evaluation of orbit determination software has been carried out using interferometer data from CNES stations during the initial phase of the mission. The standard deviation in position by orbit determination using this data is around 500 m. Details of orbit determination using Indian tracking data are presented in this paper. Attitude accuracy requirements are met during the mission phase. Representative plots of attitude history of the satellite are also presented.

Keywords: Orbit and attitude determination, Software validation, Satellite operations, Emergency operations.

1. INTRODUCTION

Bhaskara, the first Indian experimental satellite for earth observations was launched on June 7, 1979 from the USSR, in a near circular orbit inclined at 50.67 deg. to the earth's equatorial plane at an altitude of 530 Km. The spin stabilised satellite weighed 444 Kg at launch. Its equivalent diameter is 1.59m. The primary objectives of the mission are to conduct earth observations using two camera systems capable of yielding a ground resolution of 1 Km and Ocean surface studies using Satellite Microwave Radiometers (SAMIR) at two different frequencies with ground resolution of 125 Km and 200 Km respectively. In order to achieve the above objectives, the accuracy requirement in the determination of the spacecraft position is ± 4 Km. Any relaxation in this leads to increased computational efforts for picture strip generation.

The spin rate of the satellite requires to be maintained within 6-11 rpm and the spin axis perpendicular to the orbital plane within 3 deg. The lower and the upper limits of spin rate are chosen to have sufficient picture overlap and to minimise smear effect during exposure period respectively. The spin axis orientation requirement is governed by the distortion caused to the photographic image. The spin axis orientation is to be corrected periodically mainly because of the orbit regression which is 4.8 deg/day. This was planned to be carried out automatically by the onboard control system using horizon sensors within an error of ± 3deg. Alternatively, this is done by generating ground commands which are to be executed in a manual mode. A Magnetic Bias Control (MBC) coil is used as a passive attitude control device. Apart from the "Normal mode" of spin axis orientation perpendicular to the orbital plane an "alternate mode", wherein the spin axis is kept in the orbital plane was planned, as it yields a wider coverage for SAMIR.

In order to achieve the above mission demands extensive support from orbit and attitude determination software was required. The software packages developed in-house, had to be put to quality control and other tests and validated before the start of the mission. The ensuing sections delineate the software requirements for the mission, the testing procedures adopted for validating the software and operational experience during the mission.

2. SOFTWARE REQUIREMENTS

The accuracy requirement for orbit determination is ± 4 Km in position for TV data processing. Network operations requirement is that the satellite position be predicted within 2 deg. of the beam centre of VHF antenna. Orbit determination was planned to be done on a weekly basis using 3 or 4 days of tracking data on either side of the epoch. The observation types are tone range and doppler and the specified errors including ionospheric effect are:

Tone range	: bias	: 300m	(1σ)
	random	: 150m	(1σ)
Two way doppler	: bias	: 3m/sec	(1σ)
	random	: 1m/sec	(1σ)

The program for orbit determination SOIP (Satellite Orbit Improvement Program) uses differential correction procedure for orbit determination. The orbit model used is based on R.H.Merson's (Ref.1) analytical approach. The program is designed to handle optical data, range, range rate direction cosines and any combination of these.

The main accuracy requirement for attitude determination is for TV experiment and needs to be better than 1 deg. The near real time mode accuracy requirement is 3 deg. The above accuracies are to be achieved using attitude data from horizon sensors, magnetometers and sun sensors. The following are the error specifications for attitude determination:

Sensor errors	: horizon sensor : 0.2 degree in earth pulse width	
Magnetometer	: 1.5 deg.	(3σ)
Sun sensor	: 0.5 deg.	(3σ)
Orbit error	: 50 Km in position	(3σ)
Time error	: 0.06 deg.	

The attitude determination of Bhaskara is carried out in two phases. One, in near real time between consecutive visible passes, for satellite control; the other, post-facto for data mapping with a turn around time of seven days using accurate orbital data. Attitude determination in near real time mode is a critical program segment, and is designed to provide sufficiently accurate attitude information even with limited sensor data and also in sensor contingency modes. This is an important area where the software support should have enough redundancy to increase the confidence of the operation. NRTATD is the primary program which uses an 'Extended Kalman Filter' algorithm for attitude refinement in near real time and NRTLIM is the back up program which employs direct search method to find the unique solution of the spin axis vector. PFTATD, the post-facto attitude computation program, is a modified version of NRTATD, to give more accurate results, as it uses large sensor data and also accurate orbit data. Both NRTATD and PFTATD use a geometrical method for initial attitude estimation.

3. TESTING PROCEDURE

The software packages have been mission qualified by subjecting them to selected representative tests, checking boundary conditions etc. The critical conditions for the orbit determination software were identified to be those which are related to its capability to handle cases like near circular orbits, large deviations in the initial estimates of the orbit, the minimum data availability situations and varied observation types. In the case of attitude determination, contingencies could arise due to large nutation angle, different types of sensor failures, partial loss of telemetry data and large orbit errors. For testing purposes, wherever it was possible input data has been taken from a real case, else simulated input data is used.

The orbit determination program is tested for its two parts, orbit prediction testing for its mathematical modelling of equations of motion and orbit improvement part for its convergence aspects. Final quality assurance checks before the software use in operations have been carried out subjecting the programs to the tests summarised in Table 1.

Table 1.

Set No.	Test	Test orbit considered	Evaluation criteria
1	Orbit prediction accuracy check	GEOS-1 Apogee height = 2278 Km Perigee height=1113Km	Comparison with SAO optical tracking data at the end of six days.
2	Orbit prediction accuracy check	ALPHA-1 Apogee height = 3321 Km Perigee height=558 Km	Comparison with SAO optical tracking data at the end of six days.
3	Orbit determination	SIGNE-3 near circular 500 Km altitude inclination = 51 deg.	Comparison with orbit determination with CNES program results.
4	Orbit determination	Simulated tracking data with random and bias errors.	Comparison with nominal values

Test set No.1 was considered because of high perigee and apogee heights to check the accuracy of the program for the earth's oblateness effect, isolating drag effect. The absolute difference in position as compared to SAO results are 1.84 Km in X component, 0.237 Km in Y component and 2.209 Km in Z component. The model accuracy mainly due to earth's oblateness effect is acceptable to the mission requirement.

Test set No.2 was considered because it gave a testing opportunity for the accuracy of the model as a whole, considering drag as well. The prediction error upto six days is less than 3 Km in position.

The orbit determination part is then examined using real tracking data of SIGNE-3, a satellite similar to Bhaskara. Interferometer data over a week from CNES tracking stations was considered and orbit determination results at an epoch in the centre of the tracking interval were obtained from CNES. SOIP was run with the above tracking data and the orbital elements were determined. The residual observation was very small (of the order of 4' of an arc) and the converged value of the satellite position is within 0.77 Km of the CNES results. The difference along the track is 0.61 Km and across the track is 0.46 Km. Since CNES software is operationally used for a number of missions and is guaranteed to have an along track error not exceeding ± 5 Km and across track error not exceeding ± 1 Km it can be presumed that SOIP also gives the same order of accuracy. When the initial guess of satellite position was perturbed by 30-40 Km, convergence was still achieved.

Test set No.4 has been considered to test the software for all types of observation. The program has been checked by generating simulated tracking data for the nominal orbit of Bhaskara, corrupting these data by the addition of random and bias errors and recomputing the orbit. The results are quite satisfactory. Specimen results of these tests are presented in Table 2.

In testing the near real time attitude determination programs, the emphasis has been on the reliability of this segment of the program even under adverse conditions of limited data and sensor contingency modes as this has been identified as mission critical. While testing the program PFTATD the emphasis has been on the program meeting the accuracy requirement as specified by the mission. Testing of the program was done in two stages. In the first stage, mathematical formulation, flow chart and coding of the source program were reviewed. In the second stage appropriate test runs were selected and after execution, these test results were analysed.

In the mathematical formulation modelling of geomagnetic field vector, sun ephemeris generation and validity of all assumptions made were examined. The entire structure of the program was analysed for the different functional segments of the program and the interfaces joining them. Since it was not possible to get a spacecraft with both orbit and sensor configuration similar to Bhaskara, extensive testing of the program was carried out with simulated data. The basic dynamics assumed for the spacecraft is that of the spinning satellite with nutation. When the nutation angle is made equal to zero, pure spinning spacecraft motion results. A nutation damper is provided in the spacecraft and so the residual nutation angle of 0.25 deg. is considered for simulation. Sensor data is simulated over a spin cycle and over this bias and random errors are introduced. Cases of attitude commands resulting in abrupt changes in attitude can also be handled by this program.

Table 2. Sample results of orbit determination with simulated tracking data.

	Range (m)	Range rate (m/sec)	Direction cosines (minutes of an arc)
Random error	150	1	6
Bias	300	3	6

	X (Km)	Y (Km)	Z (Km)	$\dot{X}$ (Km/sec)	$\dot{Y}$ (Km/sec)	$\dot{Z}$ (Km/sec)	ΔX (Km)	ΔY (Km)	ΔZ (Km)
Nominal parameters	1968.31	6623.49	22.87	-4.584	1.348	5.904	-	-	-
SOIP input	1990.67	6616.87	4.37	-4.574	1.379	5.904	22.36	6.62	18.50
Orbit determination with tracking data:									
Range	1968.03	6623.85	23.06	-4.583	1.348	5.904	0.28	0.36	0.18
Range rate	1968.02	6623.51	23.11	-4.584	1.348	5.904	0.30	0.02	0.24
Direction cosines	1964.73	6624.09	22.66	-4.5794	1.345	5.909	3.58	0.60	0.21
Range from single station	1969.62	6622.45	21.22	-4.583	1.350	5.905	1.31	1.04	1.65
Range & Range rate from single station	1968.66	6623.42	22.66	-4.583	1.349	5.904	0.35	0.07	0.22
Range rate from single station	1968.16	6623.22	23.56	-4.858	1.348	5.903	0.16	0.27	0.69

Telemetry transmission errors are considered introducing parity, occasional wrong reading of onboard clock, and frame synchronisation unlocking.

With these simulated data many test sets were generated and attitude determination programs were tested with them. Four typical tests considered are shown in Table 3.

Table 3.

Set No.	Program tested	Test condition
1	NRTATD	Various combinations of sensors and 100 Km error in subsatellite point as input.
2	NRTATD	Commands at spin cycles 20 and 30 after the spacecraft is visible. Attitude determination before the first command and after the second command.
3	PFTATD	Unfavourable attitude geometry with sun-satellite vector and earth-satellite vector being nearly parallel.
4	PFTATD	Spacecraft spinning with nutation.

Test set No.1 was considered to study the sensitivity of the final accuracy of NRTATD for subsatellite point error. The results showed that even when the sub-satellite point error is twice the nominal value the maximum error of spin axis orientation was 2.15 deg. with a maximum of 2 minutes of data needed for satisfactory convergence.

Test set No.2 considered the case of two SAOC commands in a single pass, attitude determination before the first command and also after the second command. This is to confirm SAOC execution and also to compare the attitude change due to given commands.

Test set No.3 has been considered because it corresponds to an unfavourable geometry of reference vectors. The telemetry data here have been generated for a pass over Sriharikota (SHAR) around noon, so that the sun sensor and horizon sensor combination form an unfavourable geometry of parallel reference vectors. From sufficient amount of data the program screens them until a favourable geometry is reached to compute the initial estimate.

Test set No.4 has been considered as it determines the attitude in presence of nutation which is a critical condition for the mission. The maximum error is less than 0.3 deg. for a constant nutation of 0.25 deg.

A summary of the results of some of the above test cases are presented in table 4. The satisfactory results of all these test cases increased the confidence in the programs, thus validating the software capabilities.

4. OPERATIONAL EXPERIENCE

Interferometer tracking data from two CNES stations was available for a period of 45 days since the launch to assess the orbit determination software capabilities and to evaluate the quality of Indian tracking data. Orbit determination with SOIP was carried out at 8 epochs using the CNES interferometer data. The standard deviation in position of orbit determination was around 500 meters. Tone range data during the period of CNES tracking support was analysed to assess the performance of the orbit determination system. Orbit determination with tone range data from single station was carried out for four data sets. For the best orbit determination set the difference in position, when compared with results using CNES data was around 4 Km. However, orbit determination using the other sets showed larger deviation from the results of CNES data, mainly because of poor quality of data. A comparison of orbital elements and their standard deviations obtained during the initial phase of the Mission for a good specimen of tone ranging data with CNES interferometer data is given in table 5. For the subsequent period successful orbit determination using tone range data was not always possible either because of poor tracking data or operational difficulties. Orbit determination using Tone range data was good enough to meet tracking requirement, but the accuracies achieved did not meet the data processing requirements.

Orbit determination with the other main tracking source, two way doppler could not be carried out because of poor quality of data. However, orbit determination was carried out successfully on a routine basis using one way doppler data. Prediction accuracies were evaluated from the difference in Time of Closest Approach (TCA) as measured by one way doppler system and the predicted value which is directly related to the positional error. Of the 34 sets of one way doppler data processed since launch (as on March 2, 1981) for orbit determination, in 17 cases, prediction at the end of 7 days from the epoch were accurate to 2 seconds or less which corresponds to a positional accuracy of nearly 14 Km or less. Only on 3 occasions prediction accuracy at the end of a week from the epoch was greater than 5 seconds in TCA error. Orbit determination using one way doppler data was carried out once in the initial mission phase and compared with CNES tracking data results. For this case the difference in position is around 12 Km, when compared to orbit determination using CNES data.

Table 4. Sample results of simulation studies of attitude determination programs.

Test set No.	Epoch and simulated attitude	Sensors used	Attitude: Initial α Deg.	Initial δ Deg	Initial W rpm	Final α Deg	Final δ Deg	Final W rpm	Error $\Delta\alpha$ Deg	Error $\Delta\delta$ Deg	Data required. (min)
1	4.10.1978 $6^h\ 30^m\ 0^s$ α = 150.98 deg. δ = 124.96 deg. W = 3 rpm	DSS MAG	151.63	125.81	3	152.67	127.05	3	1.69	2.09	2
		TWSLT MAG	117.63	125.81	3	152.67	127.06	3	1.69	2.10	2
2	4.10.1978 $10^h\ 30^m\ 0.1^s$ α = 170.96 δ = 135.67 W = 4 rpm	HSS SS	171.44	135.48	4	171.39	135.5	4	0.43	0.17	1.5
			183.21	105.76	4	183.21	105.77	4	0.3	0.18	2
	After second command α = 182.91 δ = 105.95 W = 4 rpm	HSS SS MAG	171.44	135.46	4	171.45	135.61	4	0.49	0.06	1
			183.27	105.76	4	183.24	105.83	4	0.33	0.12	2
5	4.10.1978 $6^h\ 30^m\ 0^s$ α = 284.49 δ = 131.33 W = 4 rpm Nutation angle = 0.25 deg.	HSS TWLST	284.27	131.23	6	284.34	131.44	6	0.15	0.11	4.7
		HSS DSS	284.25	131.08	6	284.33	131.47	6	0.16	0.14	4.7

HSS - Horizon sensors; TWSLT - Twinslit Sun sensors; MAG - Magnetometer
DSS - Digital Sun sensor; SS - Twinslit & Digital Sun sensor
α - Right ascension δ - Co-declination W - spin rate.

Table 5. Comparison of Orbit determination using CNES and Indian tracking data

Epoch: 1.7.1979

Data used for improvement	$a(\sigma_a)$ Km	$e(\sigma_e)$	$i(\sigma_i)$ Deg	$\omega(\sigma_\omega)$ Deg	$\Omega(\sigma_\Omega)$ Deg	$M_0(\sigma_{M_0})$ Deg	$M_1(\sigma_{M_1})$ Deg/day	$M_2(\sigma_{M_2})$ Deg/day^2
CNES	6906.322 (1.2×10^{-5})	0.00214 (1.7×10^{-5})	50.667 (0.00176)	119.825 (0.311)	234.38 (0.00106)	164.021 (0.312)	5443.903 (0.0008)	0.0136 (0.003)
SHAR Tones	6907.292 (4.6×10^{-5})	0.00193 (2.7×10^{-5})	50.667 (0.0109)	104.807 (2.235)	234.414 (0.0073)	179.041 (2.234)	5443.938 (0.00309)	0.0044 (0.00042)

A contingency method of orbit determination using time of closest approach measurements only has been evolved to meet the operational needs, when successful and accurate orbit determination with any tracking data is not possible. Orbit perturbations for low earth satellites are essentially due to drag, and earth's oblateness. Earth's oblateness effect is properly accounted for in the mathematical modelling. Using the difference in predicted and observed TCA times from one way doppler system, appropriate corrections could be applied to the orbital parameters affected by drag i.e. Mean anomaly (M_0), mean motion (M_1) and rate of mean motion (M_2). Constructing a difference table on TCA errors, corrections for these parameters are applied iteratively as:

$$M_0 = \frac{\tau}{T} \times 360 \text{ deg.} \qquad (1)$$

$$M_1 = \frac{\Delta\tau}{T} \times 360 \text{ deg/day} \qquad (2)$$

$$M_2 = \frac{\Delta^2\tau}{T} \times 360 \text{ deg/day}^2 \qquad (3)$$

Where 'T' is the period of the satellite orbit, τ is the TCA error, $\Delta\tau$ is its first difference and $\Delta^2\tau$ is the second difference at the correction epoch. In many cases, position accuracy of the order of 10 Km was obtained after a couple of iterations. A set of results for updating orbital elements by this method are presented in table 6.

Orbital elements of Bhaskara since launch are presented in table 7. It is interesting to note the semi-annual variation in drag parameter M_2, presented in Figure 1.

have been more consistent. Attitude computation was possible with NRTLIM, even when horizon sensor data was not available whereas NRTATD did not converge in such a case.

The attitude control system has three independent mechanical systems having two gas bottles each, which are used for SAOC or Spin Rate Control (SRC) operation. An emergency situation arose during November 1980, when gas in one of the mechanical systems leaked out. The attitude had to be managed with the gas left over in the

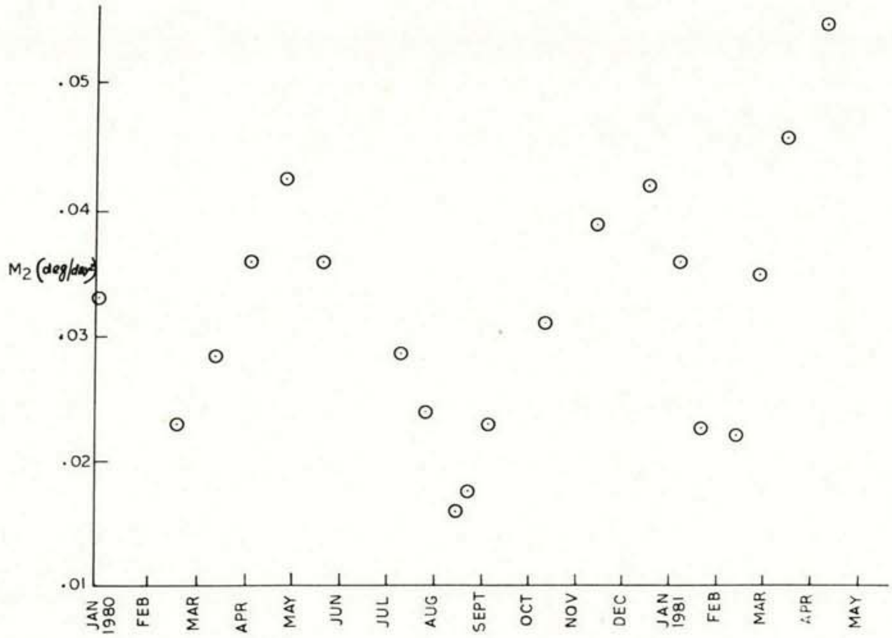

Figure 1 Mean half-acceleration of Bhaskara

Attitude management of the satellite in near real time was carried out by the manual mode of issuing Spin Axis Orientation Control (SAOC) commands from the ground, whenever there is an attitude deviation. Though automode of attitude correction was successfully tested in the initial phase, attitude correction could not be carried out in this mode on a routine basis, because of onboard switching interference changing the threshold in the Horizon Crossing Indicator (HCI). MBC was used effectively as a passive attitude control device to correct for attitude changes due to orbit regression. The attitude of the satellite is maintained fairly accurately to meet the mission goals. The attitude evolution of Bhaskara during August 1980 is presented in Figure 2 and Figure 3 as a representative sample of good attitude maintenance. It can be seen that the angle between the spin axis and the orbit normal is maintained within 5 degrees, and the spin rate between 6-8 rpm. A relaxation has been made on the attitude angle maintenance to be within 5 degree, instead of the planned 3 degree to conserve gas because of operational constraints. Though both NRTATD and NRTLIM programs are used, the results of NRTLIM

other two mechanical systems, which have been used up almost fully. This called for strict watch on the attitude of the satellite and the requirement for MBC correction became critical. MBC has been effectively used to manage the attitude by current manipulation correcting for the right ascension and issuing SAOC commands for declination correction. The attitude is maintained within 10 degrees and a case history of attitude angle during March 1981 is given in figure 4.

Post facto attitude computation was carried out to an accuracy of 0.5 deg.

5. CONCLUDING REMARKS

Bhaskara is an experimental earth observation satellite with two TV cameras and Microwave radiometers at two frequencies as primary payload. The mission requirement in post-facto orbit and attitude accuracies for image processing are ± 4 Km in position and 1 deg. in attitude. The near real time mode accuracy requirement for spacecraft control is 3 deg. Bhaskara is the first Indian Satellite mission with such requirements. Orbit

Table 6. Orbit refinement by time correction

Satellite: Bhaskara

Orbit: 9554

Epoch: 1981 02 26 00 00 00

Element	Before correction	After correction
e	0.0007355	0.0007355
i	50.667852	50.667852
w	338.87067	338.87067
	199.97071	199.97071
M_0	141.49301	141.49301
M_1	5476.4415	5476.5681
M_2	0.0245	0.043

Prediction accuracy:

Date	Orbit	TCA error Before correction.	TCA error After correction.
81.02.26	9567	2	0
81.02.27	9582	5	1
81.02.28	9597	9	0
81.03.01	9604	10	-1
81.03.02	9627	19	1
81.03.03	9642	20	-2
81.03.04	9649	24	-1
81.03.05	9672	26	-3
81.03.06	9687	40	-3
81.03.07	9702	48	-1
81.03.08	9717	56	-2
81.03.10	9747	77	-3
81.03.11	9762	87	-2
81.03.12	9769	91	1
81.03.13	9784	104	3

and attitude determination software packages have been developed in-house to meet these demands, with sufficient redundancy to meet contingencies. The computer programs have been subjected to extensive tests with real and simulated data and validated for their capabilities before the mission. These programs have also been used successfully meeting the required specifications in the mission phase. Contingency demands both planned and unplanned have been successfully met. The operational experience has given confidence in the software area to meet similar demands in the future missions.

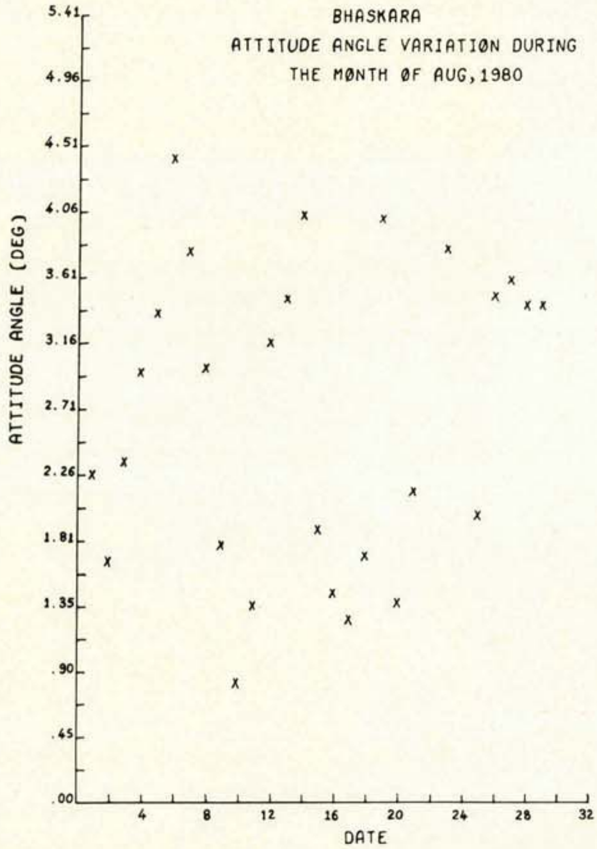

Figure 2.

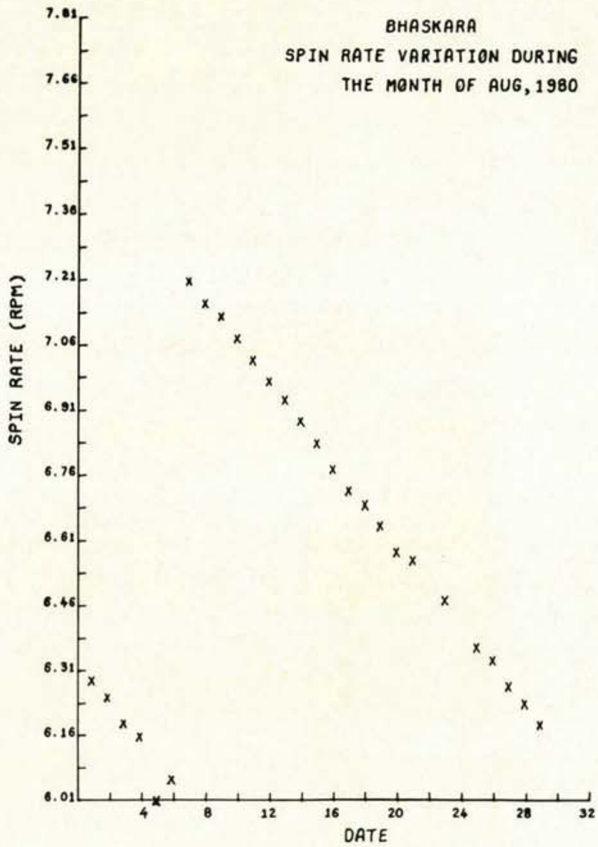

Figure 3.

Table 7. Orbital elements of Bhaskara

Epoch at mid-night	Semima-jor axis a (Km.)	Eccentri-city $e \times 10^2$	Inclina-tion i (deg)	Arg. of perigee w (deg)	Long. Ascn. Node Ω (deg.)	Mean ano-maly M_o (deg)	Mean mo-tion M_1 (deg/day)	Rate of mean mo-tion M_2 (deg/day^2)	Altitu-de (Km)
79.06.07	6907.75	0.1256	50.670	108.135	348.774	570.610	5443.333	0.0060	529.587
79.06.26	6907.75	0.2256	50.667	107.481	258.266	298.200	5443.773	0.0136	529.585
79.07.15	6906.95	0.1599	50.677	162.500	167.487	71.660	5444.254	0.0087	528.785
79.08.19	6906.97	0.0804	50.677	307.681	48.053	53.248	5444.762	0.0087	528.805
79.09.10	6906.50	0.2257	50.677	63.945	255.176	57.479	5444.258	0.0087	528.335
79.10.14	6905.93	0.1749	50.677	151.303	91.695	240.899	5444.927	0.0087	527.765
79.11.13	6902.60	0.0748	50.668	320.248	308.774	219.670	5449.873	0.0133	524.435
79.12.14	6900.50	0.2136	50.667	65.085	160.587	19.007	5452.343	0.0333	522.335
80.01.13	6898.87	0.1738	50.667	137.378	167.142	219.370	5454.282	0.0332	520.705
80.02.18	6897.54	0.0881	50.717	335.966	203.616	336.865	5455.859	0.0228	519.375
80.03.12	6896.65	0.1888	50.666	56.753	92.897	201.925	5456.928	0.0288	518.485
80.03.22	6894.35	0.1926	50.506	134.818	255.989	148.421	5459.666	0.0390	516.185
80.05.20	6892.57	0.0946	50.724	357.718	97.605	259.559	5461.777	0.0364	514.405
80.06.11	6891.85	0.1768	50.724	47.444	15.865	253.384	5462.616	0.0364	513.685
80.07.13	6890.91	0.1853	50.723	121.133	221.913	168.377	5463.748	0.0288	512.745
80.08.13	6889.99	0.0582	50.636	228.778	71.836	16.685	5464.844	0.0160	511.825
80.09.10	6889.06	0.2029	50.762	33.361	296.831	350.287	5465.951	0.0213	510.895
80.10.13	6887.65	0.2001	50.663	121.479	137.679	68.913	5467.627	0.0309	509.485
80.11.15	6885.54	0.0442	50.675	240.629	338.319	190.051	5470.138	0.0386	507.375
80.12.16	6883.72	0.1688	50.700	52.031	188.437	185.350	5472.291	0.0422	505.555
81.01.09	6882.33	0.2162	50.709	113.024	72.335	174.232	5473.964	0.0358	504.165
81.01.20	6881.96	0.1657	50.643	126.952	19.137	298.444	5474.402	0.0225	503.795

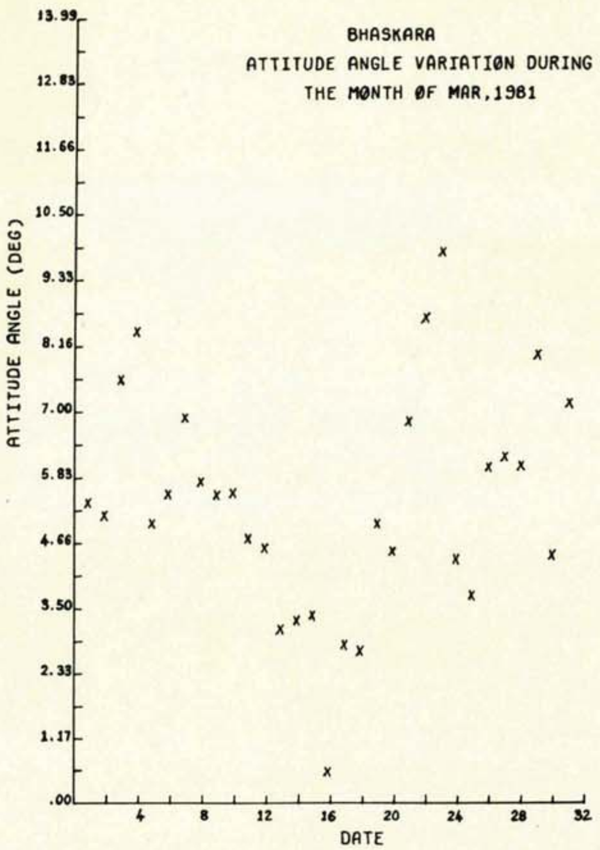

Figure 4.

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