SATELLITES ORBIT ELEMENTS : EPHEMERIS, Keplerian ELEMENTS, STATE VECTORS

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SATELLITES ORBIT ELEMENTS : EPHEMERIS, Keplerian ELEMENTS, STATE VECTORS

Satellite Ephemeris is Expressed either by 'Keplerian elements' or by 'State Vectors', that uniquely identify a specific orbit.

A satellite is an object that moves around a larger object. Thousands of Satellites launched into orbit around Earth.

First, look into the Preliminaries about 'Satellite Orbit', before moving to Satellite Ephemeris data and conversion utilities of the OM-MSS software.

(a) Satellite : An artificial object, intentionally placed into orbit. Thousands of Satellites have been launched into orbit around Earth.

A few Satellites called Space Probes have been placed into orbit around Moon, Mercury, Venus, Mars, Jupiter, Saturn, etc.

The Motion of a Satellite is a direct consequence of the Gravity of a body (earth), around which the satellite travels without any propulsion.

The Moon is the Earth's only natural Satellite, moves around Earth in the same kind of orbit.

(b) Earth Gravity and Satellite Motion : As satellite move around Earth, it is pulled in by the gravitational force (centripetal) of the Earth.

Contrary to this pull, the rotating motion of satellite around Earth has an associated force (centrifugal) which pushes it away from the Earth.

The centrifugal force equals the gravitational force and perfectly balance to maintain the satellite in its orbit.

For a satellite travelling in a circular orbit at altitude 'h' with velocity 'V', these two forces are expressed as :

- the centrifugal as F1 = (m * V^2) / (Re + h) and

- the gravitational force as F2 = (G * m * Me) /(Re + h)^2 .

Where G * Me = 3.99 x 10^14 m^3 / s^2, m is mass of satellite, G is gravitational constant, Me is mass of earth, Re is earth radius.

(c) Velocity equations : The two forces F1 and F2 are equal, therefore (m * V^2) / (Re + h) = (G * m * Me) / (Re + h)^2 .

Thus satellite velocity 'V' is related to its altitude 'h'; 'V' is constant at all circular orbit points, but vary at elliptical orbit points.

Assuming the orbit is Circular, the Satellite Velocty is expressed as V = ((G * Me)/(Re + h))^0.5 , which is

simply written as as V = ((G * Me) / r)^0.5, where 'r' = (Re + h) is the distance from satellite to earth centre.

Assuming that orbit is elliptical, the satellite Velocty is expressed as V = ((G * Me) * ((2/(Re + h)) - 1/a))^0.5 , which is

simply written as as V = ((G * Me) * (2/r - 1/a))^0.5, where 'r' = (Re + h) is distance from satellite to earth centre, and a = (rp + ra)/2

is semi-major axis, interpreted as orbit mean distance from earth center; variables rp & ra are perigee & apogee distances from earth center.

Note that the velocity of a satellite in circular or elliptical orbit depends on its altitude 'h' at that point;

secondly, the mass of satellite does not appear in its velocity equations; thus satellite velocity in its orbit is independent of its mass;

further, a satellite in elliptical orbit moves faster when closer to earth (near perigee) and moves slower when farther from earth (near apogee).

Examples of orbit altitude (km) Vs. velocity (meters/sec), for circular orbits. The typical values are :

(1) altitude 200 (km) corresponding velocity 7790 (meters/sec). (2) altitude 500 (km) corresponding velocity 7610 (meters/sec),

(3) altitude 800 (km) corresponding velocity 7450 (meters/sec). (4) altitude 35786 (km) corresponding velocity 3070 (meters/sec).

(5) altitude Moon 384000 (km) corresponding velocity 1010 (meters/sec).

(d) Attitude control : Satellite attitude is defined in terms of three axis - Roll, Pitch, & Yaw. These three axis form the local orbital reference system defined at each point of the orbit by three unit vectors. These vectors are derived from the satellite position and velocity vectors. is fully controlled relative to three axes, at each point of the orbit while the satellite moves in its orbit.

- the yaw axis is colinear with the earth's center and the satellite, called the 'geocentric direction'
- the roll axis is in the direction of the movement (velocity) of satellite, is perpendicular to geocentric axis,
- the pitch axis is perpendicular to the orbital plane.

Note, the roll axis does not coincide exactly with the velocity vector due to the eccentricity of the orbit.

All three axis pass through the center of gravity of the satellite.

The satellite attitude control is important. It helps the communication satellite antennas point towards the region of interest where ground stations are located. Similarly, by maintaining orientation, the solar panel are steered such that the panel surface is normal to Sun to generate full power. Futher, the remote sensing satellite is able to acquire images free from distortion and blurring effects.

(e) Time period : One orbit time is called time period of a satellite, calculated as the distance travelled by the satellite divided by its velocity.

For Circular Satellite Orbit : time period Pc = circumference of a circle of radius(Re + h) divided by velocity of satellite.

= 2 * pi * (((Re + h)^3) / (G * Me))^0.5 .

= 2 * pi * ((r^3) / (G * Me))^0.5 , where 'r' is radius of circular orbit from earth ceter.

For Elliptical Satellite Orbit : time period Pe = 2 * pi * ((a^3) / (G * Me))^0.5 , where 'a' is semi-major axis of the elliptical orbit.

Note : The equations show that the satellite orbit time period increases with increase in altitude. A satellite orbit altitude about 36000 km has its orbit time period roughly 1440 minutes or one sidereal day, ie (23hr,56 min,4sec) for earth to complete one orbit rotation.

Examples of orbit Time period (minutes) Vs. altitude (km), for circular orbits. The typical values are :

(1) altitude LEO 200 (km) corresponding Time period 88 (minutes). (2) altitude LEO 500 (km) corresponding Time period 94 (minutes) ,

(3) altitude LEO 800 (km) corresponding Time period 100 (minutes). (4) altitude Geo 35786 (km) corresponding Time period 1436 (minutes) .

(5) altitude Moon 384000 (km) corresponding Time period 40461 (minutes) ie roughly 28 days.

(f) Orbits : Two types of orbits, either Circular or Elliptical orbit, are assumed while formulating satellite velocity and its time period.

Any satellite can achieve orbit of any distance from the earth if its velocity is sufficient to keep it in orbit or prevent from falling to earth.

Thus, the altitudes at which satellites orbit the earth are split into three categories : Low earth orbit, Medium earth orbit, & High earth orbit.

Each category serve different purpose. Some orbits provide a constant view of one face of Earth, while others circle over different places in a day.

Some orbits are specifically named as Geosynchronous orbit (GSO), Geostationary (GEO), Equatorial Orbit, Polar Orbit, and Sunsynchronous.

The Orbit altitudes (km), Time period (minutes) and the usage/mission are briefly mentioned below.

(1) Low earth orbit (LEO) : Altitude 160-2,000 (km), Time period 87-127 (min), Used for most Earth Observation Remote Sensing satellites (EORSS), the International Space Station (ISS), the Space Shuttle, and the Hubble Space Telescope (HST).

A satellite at 300 (km) altitude has orbital period about 90 (min). In 90 (min), the earth at equator rotates about 2500 (km) .

Thus, the satellite after one time period, passes over equator a point/place 2500 (km) west of the point/place it passed over in its previous orbit.

To a person on the earth directly under the orbit, a satellite appears above horizon on one side of sky, crosses the sky, and disappears beyond the opposite horizon in about 10 (min). It reappears after 80 (min), but not over same spot, since the earth has rotated during that time.

(2) Medium earth orbit (MEO) : Altitude 2000-35,780 (km), Time period 87-1435 (min), Used for navigation satellites of Global positioning called, GPS (20,200 kilometers), Glonass (19,100 kilometers) and Galileo (23,222 kilometers) constellations. The orbit time periods are about 12 hours.

The Communications satellites that cover the North and South Pole are also put in MEO.

(3) High earth orbit (HEO) : Altitude > 35,780, Time period > a sidereal day (23hr,56 min,4sec), Used to provide coverage over any point on globe, for astronomical work and for the communication in areas usually not possible from other type of orbits.

The orbits are highly inclined and highly elliptical, characterized by low-altitude perigee and extremely high-altitude apogee.

Such highly elliptical orbits, provide coverage over polar and near polar areas needed in countries, like USA & Russia.

Two satellites in any orbit can provide continuous coverage but disadvantage is, satellite position from a point on Earth does not remain same.

(4) Geosynchronous orbit (GSO) : Altitude about 36000 (km), Time period is same as the earth's rotational period (23hr,56 min,4sec).

The word 'synchronous' means, for an observer at a fixed location on earth, the satellite returns to exactly same place and same time each day.

GSO can be circular or elliptical and of any inclination. The inclination, 0 deg and 90 deg are the special cases of GEO.

(5) Geostationary (GEO) : A special case of a Circular GSO orbit in the equatorial plane (inclination zero deg), called Equatorial Orbit.

A satellite in this orbit stays fixed relative to earth surface directly over a point on equator. To an observer on ground, the satellite appears notion less. Only in an equatorial orbit, a satellite remain stationary over a point.

(6) Equatorial Orbit : Same as Geostationary (GEO); or a special case of a geosynchronous orbit (GSO) that is circular (or nearly circular) and inclination zero (or nearly zero) deg, ie directly above the equator. An orbit directly above equator will have an inclination of 0 deg or 180 deg .

(7) Polar Orbit : A strictly defined polar orbit means inclination is 90 deg. If inclination is within a few degrees of 90, then called a near polar orbit.

Usually, Polar Orbit refers to near-polar inclination and an altitude of 700 to 900 km.

Satellites in polar orbit pass over the equator and every latitude on the Earth's surface at the same local time each day.

This means the satellite passes overhead at any location is essentially at the same time throughout the year.

This orbit enables regular data collection at consistent times and is useful for long-term comparisons.

The weather, environmental and national security related monitoring satellites are placed in polar orbits.

(8) Sun-synchronous orbit : This orbit is a special case of the polar orbit, where the orbit inclination and the altitude combines in such a way that the satellite passes over any given point of on the earth surface at the same local solar time, meaning same sunlight.

In other words, the surface illumination angle is nearly the same every time. This consistent lighting is a useful characteristic for satellites that image the Earth's surface in visible or infrared wavelengths and for those remote sensing satellites carrying ocean and atmospheric sensing instruments that require sunlight. The Sun-synchronous orbits are useful for the imaging, spy, and weather satellites.

Summary of Orbits - Satellites arround Earth (values indicated are approximate)

Orbits type	Mission	Altitude (km)	Period	Tilt(deg)	Shape
LEO . Polar sun-sync	Remote sensing, Weather	150 - 900	98 - 104 minuts	98	circular
. Inclined non-polar	International Space Stn	340	91 minuts	51.8	circular
. Polar non-sun-sync	Earth observing, Scientific	450 - 800	90 - 101 minuts	80 - 94	circular

MEO	. Semi synchronous	Navigation, Communication, Space environment	20,000	12 hours	55	circular
GEO	. Geo-synchronous . Geo-stationary	Communication, Early warning Nuclear detection, weather	35,786	24 hrs (23hr,56 min,4sec)	0	circular
HEO	. Geo-synchronous	Communication	Varies from 400 to 35,587	12hrs (11hr,58 min)	63.4	long elipse

(g) Satellite Ephemeris data : Expressed either by 'Keplerian elements' or by 'State Vectors', that uniquely identify a specific orbit.

The Keplerian elements are descriptive of the size, shape, and orientation of an orbital ellipse.

The State Vectors represent the 3-D Position and Velocity components of the orbital trajectory at a certain time.

These two have their unique advantages. The State Vectors are excellent tool for pre-launch orbit predictions.

Keplerian elements, called NASA/NORAD 'Two-Line Elements'(TLE) Ephemeris data set, acquired on a particular day are applied as input to

tracking software for accurate predictions of satellite position in next 5 to 7 days. Thereafter a new set of Keplerian elements are acquired.

(h) Satellite Orbit Keplerian Element Set : The traditional orbital elements are the six Keplerian elements, that uniquely identify a specific orbit.

The Keplerian elements are distributed as NASA/NORAD 'Two-Line Elements'(TLE) Ephemeris data set.

The NASA/NORAD 'Two-Line Elements'(TLE) Ephemeris data set are explaned in next section.

(i) Satellite Orbit State Vectors Set : It is another common form of Satellite Orbital Element Set.

State Vectors represents, Position (X, Y, Z) and Velocity (Vx, Vy, Vz) of a Satellite orbital trajectory in time.

Vectors are excellent tool for a pre-launch or any past or future time, prediction of satellite position in orbit.

The conversion of Keplerian elements (NASA/NORAD 'Two-Line Elements') to state vectors are presented in next section .

(j) Ground Trace : A ground track or ground trace is the path on the surface of the Earth directly below an satellite.

It is the projection of the satellite's orbit onto the surface of the Earth.

Thus completed, the few preliminaries about 'Satellite Orbit' around earth.

Move on to Satellites Orbit Elements : NASA/NORAD 'Two-Line Elements, Keplerian Element and State Vectors. (Sections - 5.1 to 5.4).

First presented NASA/NORAD 'Two-Line Elements'(TLE) Ephemeris data set.

This is followed by Conversion of the Keplerian Element Set to State Vector Set and vice versa.

Later, Compute Keplerian Element at Perigee prior to Epoch, that can be used as new Ephemeris if perigee is start point for satellite pass.

Next Section - 5.1 NASA/NORAD 'Two-Line Elements'(TLE) Ephemeris data set.

NASA/NORAD 'Two-Line Elements'(TLE) Ephemeris Data Set

The Keplerian elements are encoded as text in different formats. The most common format is NASA/NORAD 'Two-Line Elements'(TLE).

TLE format consists 3 lines : First line call 'Line 0' contains satellite name, followed by standard two lines call 'Line 1' & 'Line 2' of Orbital Element.

TLE is distributed as NASA/NORAD 'Two-Line Elements'(TLE) Ephemeris data set.

Example of a TLE for the International Space Station 'ISS (ZARYA)'.

There are three lines; line 0 format AAAAAAAAAAA is title 'satellite name', where a disagreement exists about how wide name may be, 11 or 12, or 24 Columns.

```
ISS (ZARYA)

1 NNNNNU NNNNNAAA NNNNN.NNNNNNNN +.NNNNNNNN +NNNNN-N +NNNNN-N N NNNNN      line 1  format Total Columns 67,   blanks  2  9  18  33   44   53   62  64 .

1 25544U 98067A   08264.51782528 -.00002182  00000-0 -11606-4 0  2927      line 1  with Orbital Element put/filled in.

2 NNNNN NNN.NNNN NNN.NNNN NNNNNNN NNN.NNNN NNN.NNNN NN.NNNNNNNNNNNNNN      line 2  format Total Columns 67,   blanks  2  8  17  26   34   43   52 .

2 25544  51.6416 247.4627 0006703 130.5360 325.0288 15.72125391563537      line 2  with Orbital Element put/filled in .
```

The format of line 1 & line 2 are explained below.

LINE 1	Field	Columns	Blank	Content	Values from the above example
	1	01-01	02	Line number	1
	2	03-07		Satellite number	25544
	3	08-08	09	Classification (U = Unclassified)	U
	4	10-11		International Designator (Last two digits of launch year)	98
	5	12-14		International Designator (Launch number of the year)	067
	6	15-17	18	International Designator (Piece of the launch)	A
	7	19-20		Epoch Year (Last two digits of year)	08
	8	21-32	33	Epoch (Day of the year and fractional portion of the day)	264.51782528
	9	34-43	44	First Time Derivative of the Mean Motion divided by two	-.00002182
	10	45-52	53	Second Time Derivative of Mean Motion divided by six	00000-0
	11	54-61	62	BSTAR drag term	-11606-4

	Field	Columns	Blank	Content	Values from the above example
	12	63-63	64	number 0 (Originally Ephemeris type)	0
	13	65-68		Element set number. incremented when a new TLE is generated	292
	14	69-69		Checksum (Modulo 10)	7

LINE 2	Field	Columns	Blank	Content	Values from the above example
	1	01-01	02	Line number	2
	2	03-07	08	Satellite number	25544
	3	09-16	17	Inclination deg.	51.6416
	4	18-25	26	Right Ascension of the Ascending Node (deg)	247.4627
	5	27-33	34	Eccentricity (decimal point assumed)	0006703
	6	35-42	43	Argument of Perigee (deg)	130.5360
	7	44-51	52	Mean Anomaly (deg)	325.0288
	8	53-63		Mean Motion (Revs per day)	15.72125391
	9	64-68		Revolution number at epoch (Revs)	56353
	10	69-69		Checksum (Modulo 10)	7

Note :

1. Where decimal points are assumed, it is leading decimal points, for example Line 1, Field 11 (-11606-4) translates to -0.11606E-4.
2. The orbit number at the epoch time is counted at ascending equator crossings or at perigee.
3. Checksum is computed as follows : Start with zero. For each digit in the line, add the value of the digit.
   For each minus sign, add 1. For each plus sign, add 2. For each letter, blank, or period, don't add anything.
   Take the last decimal digit of the result (ie, take the result modulo 10) as the check digit.
4. The International Designator, Line 1 fields (4, 5, 6) are usually blank in the NASA Prediction Bulletins issued.

A credible, regular, updated free service source for NASA/NORAD 'Two-Line Elements'(TLE) Bulletins :

The Center for Space Standards & Innovation (CSSI) provide worldwide standard data & educational materials to space community.

The Data services are offered from CSSI's freely, which includes Two-Line element sets, precision orbit ephemerides, solar weather data, etc.

However, for all satellites, you can download the NASA/NORAD 'two-line elements'(TLE) from CelesTrak Web site, URL **http://celestrak.com/NORAD/elements/** .

See below, for five satellites, the NASA/NORAD 'two-line elements'(TLE) download on May 28, 2014, from site URL **http://celestrak.com/NORAD/elements/**

The NASA/NORAD 'two-line elements'(TLE) for five satellites download on May 28, 2014, from site URL http://celestrak.com/NORAD/elements/

(a). American Remote Sensing satellite launched on February 11, 2013, the NASA/NORAD TLE down load on May 28, 2014, 18:13 hrs IST

```
LANDSAT 8
1 39084U 13008A   14148.14086282  .00000288  00000-0  73976-4 0  4961
2 39084  98.2215 218.5692 0001087  96.5686 263.5699 14.57098925 68534
```

(b). French Remote Sensing satellite launched on September 9, 2012, the NASA/NORAD TLE down load on May 28, 2014, 18:13 hrs IST

```
SPOT 6
1 38755U 12047A   14148.14295346  .00000295  00000-0  73402-4 0  9574
2 38755  98.1987 215.8134 0001368  80.3963 279.7434 14.58528066 91251
```

(c). Indian Remote Sensing satellite launched on July 12, 2010, the NASA/NORAD TLE down load on May 28, 2014, 18:13 hrs IST

```
CARTOSAT 2B
1 36795U 10035A   14148.12955979  .00000641  00000-0  94319-4 0  3461
2 36795  97.9448 207.1202 0016257  44.4835 315.7690 14.78679483209252
```

(d). International Space Stn launched on Nov. 20, 1998, the NASA/NORAD TLE down load on May 28, 2014, 18:13 hrs IST

```
ISS (ZARYA)
1 25544U 98067A   14148.25353351  .00006506  00000-0  11951-3 0  3738
2 25544  51.6471 198.4055 0003968  47.6724  33.3515 15.50569135888233
```

(e). Indian Geo Comm. Sat launched on Jan. 05, 2014, the NASA/NORAD TLE down load on May 28, 2014, 18:13 hrs IST

```
GSAT-14
1 39498U 14001A   14146.03167358 -.00000092  00000-0  00000+0 0  1238
2 39498   0.0049 223.9821 0002051 110.2671 354.6468  1.00272265  1407
```

(f) Natural satellite Moon moves around Earth in the same kind of orbit as the artificial satellites.

For Moon, the NASA/NORAD 'Two-Line Elements'(TLE) Bulletins are not easily available or offered regularly in public domain. Giving Keplerian elements for the Moon is much more difficult. The Moon's orbital plane wobbles around that changes inclination about 18 to 28 degrees. The Moon's orbit is also severely perturbed. The perturbations come from Sun, Earth (not being exact sphere) and from major planets. (Ref. **http://www.amsat.org/amsat/archive/amsat-bb/200107/msg00247.html**).

However, for the natural satellite Moon, the Keplerian elements set down load on Jun 14, 2014, 16:14 hrs IST,

from site URL **http://www.logsat.com/lsp-keplerian-elements.asp** & **http://www.logsat.com/pub/Jun14.txt** is :

Satellite: Moon , Catalog number: 000000 , Epoch time: 14143.16621081682 , Element set: 357 ,

Inclination: 18.7965 deg , RA of node: 352.4777 deg , Eccentricity: 0.0512

Arg of perigee: 316.1136 deg , Mean anomaly: 40.2074 deg , Mean motion: 0.036600996000 rev/day ,

Decay rate: 0 , Epoch rev: 0 , Checksum: 000

This is rewritten in NASA/NORAD 'Two-Line Elements'(TLE) Bulletins form :

```
MOON  (natural satellite of earth)
1 00000U 00000A   14143.16621081  .00000000  00000-0  00000-0 0  3574
2 00000  18.7965 352.4777 0512000 316.1136  40.2074 00.036600996 0006
```

Thus, explained with examples, the format of NASA/NORAD 'Two-Line Elements'(TLE) as the satellite orbital parameters called Keplerian elements and the source for obtaining them for the satellites launched by any country.

Move on to Conversion of Keplerian Element Set to State Vector Set and vice versa, applied to six satellites.

These six satellites are - LANDSAT 8, SPOT 6, CARTOSAT-2B, ISS (ZARYA), GSAT-14, and Moon .

Next Section - 5.2 Conversion of Keplerian Element Set to State Vector Set and vice versa.

Conversion of Keplerian Element Set to State Vector Set and Vice versa.

This utility is applied to six satellites, LANDSAT 8, SPOT 6, CARTOSAT-2B, ISS (ZARYA), GSAT-14, and Moon .

The input is respective Satellite's NASA/NORAD 'Two-Line Elements'(TLE).

The output is corrsponding State Vector Set for the respective Satellites.

Satellite LANDSAT 8 : Conversion of Keplerian Element Set to State Vector Set and vice versa.

(a) LANDSAT 8 'Two-Line Elements'(TLE) downloaded on May 28, 2014, 18:13 hrs IST, Satellite launched on February 11, 2013

```
1 39084U 13008A   14148.14086282  .00000288  00000-0  73976-4 0  4961
2 39084  98.2215 218.5692 0001087  96.5686 263.5699 14.57098925 68534
```

From this TLE, the data relevant for the purpose are manually interpreted & extracted :

Satellite number 39084, LANDSAT 8 ,

(i) Conversion Utility : Keplerian Elements to State Vectors (Forward Conversion)

Input : The Keplerian Elements Set at Epoch, extracted from 'Two-Line Elements'(TLE) .

```
EPOCH_year  =  2014    EPOCH_days_decimal_of_year   =  147.1408628200

EPOCH_inclination_deg          =   98.2215000000        EPOCH_right_asc_acnd_node_deg   =  218.5692000000
EPOCH_eccentricity             =    0.0001087000        EPOCH_argument_of_perigee_deg   =   96.5686000000
EPOCH_mean_anomaly_deg         =  263.5699000000        EPOCH_mean_motion_rev_per_day   =   14.5709892500
EPOCH_revolution               =     6853               EPOCH_node_condition            =  1
```

Output : The Computed corresponding State Vectors Set, Position vector(X, Y, Z) and Velocity vector(Vx, Vy, Vz) at Epoch.

```
EPOCH_year   =  2014    EPOCH_days_decimal_of_year    =  147.1408628200

X  =   -5535.2447229896      Y  =   -4411.0085700927      Z  =      15.4200278230      R  =    7077.8646869006
Vx =    -655.5016695670      Vy =     849.8345806371      Vz =    7427.2400585557      V  =    7504.3851274216
EPOCH_revolution               =     6853               EPOCH_node_condition            =  1
```

(ii) Conversion Utility : State Vectors to Keplerian Elements (Backward Conversion)

Input : The State Vectors Set at Epoch, computed just above from 'Two-Line Elements'(TLE).

EPOCH_year = 2014 EPOCH_days_decimal_of_year = 147.1408628200

X =	-5535.2447229896	Y =	-4411.0085700927	Z =	15.4200278230	R =	7077.8646869006
Vx =	-655.5016695670	Vy =	849.8345806371	Vz =	7427.2400585557	V =	7504.3851274216

EPOCH_revolution = 6853 EPOCH_node_condition = 1

Output : The Computed corresponding Keplerian Elements Set at Epoch.

EPOCH_year = 2014 EPOCH_days_decimal_of_year = 147.1408628200

EPOCH_inclination_deg	= 98.2215000000	EPOCH_right_asc_acnd_node_deg	= 218.5692000000
EPOCH_eccentricity	= 0.0001087000	EPOCH_argument_of_perigee_deg	= 96.5686000000
EPOCH_mean_anomaly_deg	= 263.5698999999	EPOCH_mean_motion_rev_per_day	= 14.5709892500
EPOCH_revolution	= 6853	EPOCH_node_condition	= 1

Continue To next Satellite

Satellite SPOT 6 : Conversion of Keplerian Element Set to State Vector Set and vice versa.

(b) SPOT 6 'Two-Line Elements'(TLE) downloaded on May 28, 2014, 18:13 hrs IST, Satellite launched on September 9, 2012

```
1 38755U 12047A   14148.14295346  .00000295  00000-0  73402-4 0  9574
2 38755  98.1987 215.8134 0001368  80.3963 279.7434 14.58528066 91251
```

From this TLE, the data relevant for the purpose are manually interpreted & extracted :

Satellite number 38755, SPOT 6 ,

(i) Conversion Utility : Keplerian Elements to State Vectors (Forward Conversion)

Input : The Keplerian Elements Set at Epoch, extracted from 'Two-Line Elements'(TLE) .

```
EPOCH_year  =  2014    EPOCH_days_decimal_of_year   =  147.1429534600
EPOCH_inclination_deg          =   98.1987000000        EPOCH_right_asc_acnd_node_deg   =  215.8134000000
EPOCH_eccentricity             =    0.0001368000        EPOCH_argument_of_perigee_deg   =   80.3963000000
EPOCH_mean_anomaly_deg         =  279.7434000000        EPOCH_mean_motion_rev_per_day   =   14.5852806600
EPOCH_revolution               =     9125               EPOCH_node_condition            =  1
```

Output : The Computed corresponding State Vectors Set, Position vector(X, Y, Z) and Velocity vector(Vx, Vy, Vz) at Epoch.

```
EPOCH_year   =  2014    EPOCH_days_decimal_of_year     =  147.1429534600
X  =   -5736.9414700815      Y  =   -4136.9553443077      Z  =      15.1814434008      R  =    7072.9857505978
Vx =    -612.4123815408      Vy =     878.2634599352      Vz =    7430.3591738511      V  =    7507.1055062891
EPOCH_revolution                =     9125               EPOCH_node_condition            =  1
```

(ii) Conversion Utility : State Vectors to Keplerian Elements (Backward Conversion)

Input : The State Vectors Set at Epoch, computed just above from 'Two-Line Elements'(TLE).

```
EPOCH_year   =  2014    EPOCH_days_decimal_of_year     =  147.1429534600
X  =   -5736.9414700815      Y  =   -4136.9553443077      Z  =      15.1814434008      R  =    7072.9857505978
Vx =    -612.4123815408      Vy =     878.2634599352      Vz =    7430.3591738511      V  =    7507.1055062891
EPOCH_revolution                =     9125               EPOCH_node_condition            =  1
```

Output : The Computed corresponding Keplerian Elements Set at Epoch.

EPOCH_year = 2014		EPOCH_days_decimal_of_year = 147.1429534600	
EPOCH_inclination_deg	= 98.1987000000	EPOCH_right_asc_acnd_node_deg	= 215.8134000000
EPOCH_eccentricity	= 0.0001368000	EPOCH_argument_of_perigee_deg	= 80.3963000000
EPOCH_mean_anomaly_deg	= 279.7434000000	EPOCH_mean_motion_rev_per_day	= 14.5852806600
EPOCH_revolution	= 9125	EPOCH_node_condition	= 1

Continue To next Satellite

Satellite CARTOSAT 2B : Conversion of Keplerian Element Set to State Vector Set and vice versa.

(c) CARTOSAT 2B 'Two-Line Elements'(TLE) downloaded on May 28, 2014, 18:13 hrs IST, Satellite launched on July 12, 2010

```
1 36795U 10035A   14148.12955979  .00000641  00000-0  94319-4 0  3461
2 36795  97.9448 207.1202 0016257  44.4835 315.7690 14.78679483209252
```

From this TLE, the data relevant for the purpose are manually interpreted & extracted :

Satellite number 36795, CARTOSAT 2B ,

(i) Conversion Utility : Keplerian Elements to State Vectors (Forward Conversion)

Input : The Keplerian Elements Set at Epoch, extracted from 'Two-Line Elements'(TLE) .

EPOCH_year = 2014 EPOCH_days_decimal_of_year = 147.1295597900

EPOCH_inclination_deg	= 97.9448000000	EPOCH_right_asc_acnd_node_deg	= 207.1202000000
EPOCH_eccentricity	= 0.0016257000	EPOCH_argument_of_perigee_deg	= 44.4835000000
EPOCH_mean_anomaly_deg	= 315.7690000000	EPOCH_mean_motion_rev_per_day	= 14.7867948300
EPOCH_revolution	= 20925	EPOCH_node_condition	= 1

Output : The Computed corresponding State Vectors Set, Position vector(X, Y, Z) and Velocity vector(Vx, Vy, Vz) at Epoch.

EPOCH_year = 2014 EPOCH_days_decimal_of_year = 147.1295597900

X = -6231.7560551250	Y = -3189.4018492384	Z = 14.8069953230	R = 7000.5204759091
Vx = -453.7396013124	Vy = 940.0898291212	Vz = 7477.6527638575	V = 7550.1615459169

EPOCH_revolution = 20925 EPOCH_node_condition = 1

(ii) Conversion Utility : State Vectors to Keplerian Elements (Backward Conversion)

Input : The State Vectors Set at Epoch, computed just above from 'Two-Line Elements'(TLE).

EPOCH_year = 2014 EPOCH_days_decimal_of_year = 147.1295597900

X = -6231.7560551250	Y = -3189.4018492384	Z = 14.8069953230	R = 7000.5204759091
Vx = -453.7396013124	Vy = 940.0898291212	Vz = 7477.6527638575	V = 7550.1615459169

EPOCH_revolution = 20925 EPOCH_node_condition = 1

Output : The Computed corresponding Keplerian Elements Set at Epoch.

EPOCH_year	= 2014	EPOCH_days_decimal_of_year	=	147.1295597900	
EPOCH_inclination_deg	=	97.9448000000	EPOCH_right_asc_acnd_node_deg	=	207.1202000000
EPOCH_eccentricity	=	0.0016257000	EPOCH_argument_of_perigee_deg	=	44.4835000000
EPOCH_mean_anomaly_deg	=	315.7690000000	EPOCH_mean_motion_rev_per_day	=	14.7867948300
EPOCH_revolution	=	20925	EPOCH_node_condition	=	1

Continue To next Satellite

Satellite ISS (ZARYA) : Conversion of Keplerian Element Set to State Vector Set and vice versa.

(d) ISS (ZARYA) 'Two-Line Elements'(TLE) downloaded on May 28, 2014, 18:13 hrs IST, Satellite launched on November 20, 1998

```
1 25544U 98067A   14148.25353351  .00006506  00000-0  11951-3 0  3738
2 25544  51.6471 198.4055 0003968  47.6724  33.3515 15.50569135888233
```

From this TLE, the data relevant for the purpose are manually interpreted & extracted :

Satellite number 25544, ISS (ZARYA) ,

(i) Conversion Utility : Keplerian Elements to State Vectors (Forward Conversion)

Input : The Keplerian Elements Set at Epoch, extracted from 'Two-Line Elements'(TLE) .

EPOCH_year = 2014 EPOCH_days_decimal_of_year = 147.2535335100

EPOCH_inclination_deg	= 51.6471000000	EPOCH_right_asc_acnd_node_deg	= 198.4055000000
EPOCH_eccentricity	= 0.0003968000	EPOCH_argument_of_perigee_deg	= 47.6724000000
EPOCH_mean_anomaly_deg	= 33.3515000000	EPOCH_mean_motion_rev_per_day	= 15.5056913500
EPOCH_revolution	= 88823	EPOCH_node_condition	= 1

Output : The Computed corresponding State Vectors Set, Position vector(X, Y, Z) and Velocity vector(Vx, Vy, Vz) at Epoch.

EPOCH_year = 2014 EPOCH_days_decimal_of_year = 147.2535335100

X = 311.7253734371	Y = -4283.4907194611	Z = 5261.0200081909	R = 6791.4502853764
Vx = 7415.4532574405	Vy = 1686.8647169809	Vz = 936.2139516379	V = 7662.3074951298

EPOCH_revolution = 88823 EPOCH_node_condition = 1

(ii) Conversion Utility : State Vectors to Keplerian Elements (Backward Conversion)

Input : The State Vectors Set at Epoch, computed just above from 'Two-Line Elements'(TLE).

EPOCH_year = 2014 EPOCH_days_decimal_of_year = 147.2535335100

X = 311.7253734371	Y = -4283.4907194611	Z = 5261.0200081909	R = 6791.4502853764
Vx = 7415.4532574405	Vy = 1686.8647169809	Vz = 936.2139516379	V = 7662.3074951298

EPOCH_revolution = 88823 EPOCH_node_condition = 1

Output : The Computed corresponding Keplerian Elements Set at Epoch.

EPOCH_year	= 2014	EPOCH_days_decimal_of_year	= 147.2535335100		
EPOCH_inclination_deg	=	51.6471000000	EPOCH_right_asc_acnd_node_deg	=	198.4055000000
EPOCH_eccentricity	=	0.0003968000	EPOCH_argument_of_perigee_deg	=	47.6724000000
EPOCH_mean_anomaly_deg	=	33.3515000000	EPOCH_mean_motion_rev_per_day	=	15.5056913500
EPOCH_revolution	=	88823	EPOCH_node_condition	=	1

Continue To next Satellite

Satellite GSAT-14) : Conversion of Keplerian Element Set to State Vector Set and vice versa.

(e) GSAT-14 'Two-Line Elements'(TLE) downloaded on May 28, 2014, 18:14 hrs IST, Satellite launched on January 05, 2014

```
1 39498U 14001A   14146.03167358 -.00000092  00000-0  00000+0 0  1238
2 39498   0.0049 223.9821 0002051 110.2671 354.6468  1.00272265  1407
```

From this TLE, the data relevant for the purpose are manually interpreted & extracted:

Satellite number 39498, GSAT-14 ,

(i) Conversion Utility : Keplerian Elements to State Vectors (Forward Conversion)

Input : The Keplerian Elements Set at Epoch, extracted from 'Two-Line Elements'(TLE) .

```
EPOCH_year  =  2014    EPOCH_days_decimal_of_year   =  145.0316735800
EPOCH_inclination_deg          =    0.0049000000        EPOCH_right_asc_acnd_node_deg   =  223.9821000000
EPOCH_eccentricity             =    0.0002051000        EPOCH_argument_of_perigee_deg   =  110.2671000000
EPOCH_mean_anomaly_deg         =  354.6468000000        EPOCH_mean_motion_rev_per_day   =    1.0027226500
EPOCH_revolution               =      140               EPOCH_node_condition            =  1
```

Output : The Computed corresponding State Vectors Set, Position vector(X, Y, Z) and Velocity vector(Vx, Vy, Vz) at Epoch.

```
EPOCH_year   =  2014    EPOCH_days_decimal_of_year    =  145.0316735800
X  =   36095.3223130873      Y  =  -21779.4122304999      Z  =       3.4839025250      R  =   42157.0290951495
Vx =    1588.6953038064      Vy =    2633.0807004007      Vz =      -0.0676820819      V  =    3075.2344215913
EPOCH_revolution               =      140               EPOCH_node_condition            =  1
```

(ii) Conversion Utility : State Vectors to Keplerian Elements (Backward Conversion)

Input : The State Vectors Set at Epoch, computed just above from 'Two-Line Elements'(TLE).

```
EPOCH_year   =  2014    EPOCH_days_decimal_of_year    =  145.0316735800
X  =   36095.3223130873      Y  =  -21779.4122304999      Z  =       3.4839025250      R  =   42157.0290951495
Vx =    1588.6953038064      Vy =    2633.0807004007      Vz =      -0.0676820819      V  =    3075.2344215913
EPOCH_revolution               =      140               EPOCH_node_condition            =  1
```

Output : The Computed corresponding Keplerian Elements Set at Epoch.

EPOCH_year	= 2014	EPOCH_days_decimal_of_year	= 145.0316735800		
EPOCH_inclination_deg	=	0.0049000000	EPOCH_right_asc_acnd_node_deg	=	223.9821000000
EPOCH_eccentricity	=	0.0002051000	EPOCH_argument_of_perigee_deg	=	110.2671000000
EPOCH_mean_anomaly_deg	=	354.6468000000	EPOCH_mean_motion_rev_per_day	=	1.0027226500
EPOCH_revolution	=	140	EPOCH_node_condition	=	1

Continue To next Satellite

Satellite MOON : Conversion of Keplerian Element Set to State Vector Set and vice versa.

(f) MOON 'Two-Line Elements'(TLE) downloaded on Jun 14, 2014, 16:14 hrs IST, Natural Satellite

```
1 00000U 00000A   14143.16621081  .00000000  00000-0  00000-0 0  3574
2 00000  18.7965 352.4777 0512000 316.1136  40.2074 00.036600996 0006
```

From this TLE, the data relevant for the purpose are manually interpreted & extracted :

Satellite number 00000, MOON ,

(i) Conversion Utility : Keplerian Elements to State Vectors (Forward Conversion)

Input : The Keplerian Elements Set at Epoch, extracted from 'Two-Line Elements'(TLE) .

```
EPOCH_year  =  2014    EPOCH_days_decimal_of_year   =  142.1662108168
EPOCH_inclination_deg       =   18.7965000000     EPOCH_right_asc_acnd_node_deg  =  352.4777000000
EPOCH_eccentricity          =    0.0512000000     EPOCH_argument_of_perigee_deg  =  316.1136000000
EPOCH_mean_anomaly_deg      =   40.2074000000     EPOCH_mean_motion_rev_per_day  =    0.0366009960
EPOCH_revolution            =        0            EPOCH_node_condition           =  1
```

Output : The Computed corresponding State Vectors Set, Position vector(X, Y, Z) and Velocity vector(Vx, Vy, Vz) at Epoch.

```
EPOCH_year   =  2014    EPOCH_days_decimal_of_year    =  142.1662108168
X  =  365705.5648844948     Y  =  -46450.6213911481     Z  =     620.9529484744     R  =  368644.2811134813
Vx =     161.8765603889     Vy =     989.7819390712     Vz =     341.1953415596     V  =    1059.3802758296
EPOCH_revolution            =        0            EPOCH_node_condition           =  1
```

(ii) Conversion Utility : State Vectors to Keplerian Elements (Backward Conversion)

Input : The State Vectors Set at Epoch, computed just above from 'Two-Line Elements'(TLE).

```
EPOCH_year   =  2014    EPOCH_days_decimal_of_year    =  142.1662108168
X  =  365705.5648844948     Y  =  -46450.6213911481     Z  =     620.9529484744     R  =  368644.2811134813
Vx =     161.8765603889     Vy =     989.7819390712     Vz =     341.1953415596     V  =    1059.3802758296
EPOCH_revolution            =        0            EPOCH_node_condition           =  1
```

Output : The Computed corresponding Keplerian Elements Set at Epoch.

EPOCH_year = 2014		EPOCH_days_decimal_of_year	= 142.1662108168
EPOCH_inclination_deg	= 18.7965000000	EPOCH_right_asc_acnd_node_deg	= 352.4777000000
EPOCH_eccentricity	= 0.0512000000	EPOCH_argument_of_perigee_deg	= 316.1136000000
EPOCH_mean_anomaly_deg	= 40.2074000000	EPOCH_mean_motion_rev_per_day	= 0.0366009960
EPOCH_revolution	= 0	EPOCH_node_condition	= 1

End of the Conversion of the Keplerian Element Set to State Vector Set and vice versa for six Satellites.

If any need arise, the above utility can accurately convert the Keplerian Element set into State Vector Set, or the reverse.

The source for Keplerian Element set, is NASA/NORAD 'Two-Line Elements'(TLE), mentioned before.

Next Section - 5.3 Satellite Orbit Keplerian element at Perigee, prior to Epoch.

Satellite Orbit Keplerian Element set at Perigee, prior to Epoch.

Compute Satellite Orbit Keplerian element set at Perigee prior to Epoch.

This utility is applied to six satellites, LANDSAT 8, SPOT 6, CARTOSAT-2B, ISS (ZARYA), GSAT-14, and Moon .

The input is of the respective Satellite's NASA/NORAD 'Two-Line Elements'(TLE).

The outputs is corrsponding Keplerian element set at Perigee prior to Epoch, for the respective Satellites.

(a) LANDSAT 8 'Two-Line Elements'(TLE) downloaded on May 28, 2014, 18:13 hrs IST, Satellite launched on February 11, 2013

```
1 39084U 13008A   14148.14086282  .00000288  00000-0  73976-4 0  4961
2 39084  98.2215 218.5692 0001087  96.5686 263.5699 14.57098925 68534
```

From this TLE, the Keplerian Element manually interpreted & extracted :

Satellite number 39084, LANDSAT 8 ,

```
EPOCH_year  =  2014           EPOCH_days_decimal_of_year   =  147.1408628200
EPOCH_inclination_deg          =   98.2215000000            EPOCH_right_asc_acnd_node_deg   =  218.5692000000
EPOCH_eccentricity             =    0.0001087000            EPOCH_argument_of_perigee_deg   =   96.5686000000
EPOCH_mean_anomaly_deg         =  263.5699000000            EPOCH_mean_motion_rev_per_day   =   14.5709892500
EPOCH_revolution               =   6853                     EPOCH_node_condition            =  1
```

Computed Satellite Orbit Keplerian Element Set at PERIGEE prior to Epoch from NORAD TLE as Epoch.

```
PERIGEE_year   =  2014        PERIGEE_days_decimal_of_year   =  147.0905854502
PERIGEE_inclination_deg        =   98.2215000000            PERIGEE_right_asc_acnd_node_deg =  218.5196064188
PERIGEE_eccentricity           =    0.0001087000            PERIGEE_argument_of_perigee_deg =   96.7242739798
PERIGEE_mean_anomaly_deg       =  359.9999999029            PERIGEE_mean_motion_rev_per_day =   14.5619910304
PERIGEE_revolution             =    6853                    PERIGEE_node_condition          =  1
```

Move on to next Satellite computing values of 'Keplerian element set at Perigee prior to Epoch'

Continue To next Satellite

(b) SPOT 6 'Two-Line Elements'(TLE) downloaded on May 28, 2014, 18:13 hrs IST, Satellite launched on September 9, 2012

```
1 38755U 12047A   14148.14295346  .00000295  00000-0  73402-4 0  9574
2 38755  98.1987 215.8134 0001368  80.3963 279.7434 14.58528066 91251
```

From this TLE, the Keplerian Element manually interpreted & extracted :

Satellite number 38755, SPOT 6 ,

```
EPOCH_year  =  2014            EPOCH_days_decimal_of_year   =  147.1429534600
EPOCH_inclination_deg          =   98.1987000000            EPOCH_right_asc_acnd_node_deg   =  215.8134000000
EPOCH_eccentricity             =    0.0001368000            EPOCH_argument_of_perigee_deg   =   80.3963000000
EPOCH_mean_anomaly_deg         =  279.7434000000            EPOCH_mean_motion_rev_per_day   =   14.5852806600
EPOCH_revolution               =   9125                     EPOCH_node_condition            =  1
```

Computed Satellite Orbit Keplerian Element Set at PERIGEE prior to Epoch from NORAD TLE as Epoch.

```
PERIGEE_year   =  2014         PERIGEE_days_decimal_of_year   =  147.0896431408
PERIGEE_inclination_deg        =   98.1987000000            PERIGEE_right_asc_acnd_node_deg =  215.7608394403
PERIGEE_eccentricity           =    0.0001368000            PERIGEE_argument_of_perigee_deg =   80.5618466274
PERIGEE_mean_anomaly_deg       =    0.0000002622            PERIGEE_mean_motion_rev_per_day =   14.5762585790
PERIGEE_revolution             =    9125                    PERIGEE_node_condition          =  1
```

Move on to next Satellite computing values of 'Keplerian element set at Perigee prior to Epoch'

Continue To next Satellite

(c) CARTOSAT 2B 'Two-Line Elements'(TLE) downloaded on May 28, 2014, 18:13 hrs IST, Satellite launched on July 12, 2010

```
1 36795U 10035A   14148.12955979  .00000641  00000-0  94319-4 0  3461
2 36795  97.9448 207.1202 0016257  44.4835 315.7690 14.78679483209252
```

From this TLE, the Keplerian Element manually interpreted & extracted :

Satellite number 36795, CARTOSAT 2B ,

```
EPOCH_year  =  2014            EPOCH_days_decimal_of_year   =  147.1295597900
EPOCH_inclination_deg          =   97.9448000000              EPOCH_right_asc_acnd_node_deg   =  207.1202000000
EPOCH_eccentricity             =    0.0016257000              EPOCH_argument_of_perigee_deg   =   44.4835000000
EPOCH_mean_anomaly_deg         =  315.7690000000              EPOCH_mean_motion_rev_per_day   =   14.7867948300
EPOCH_revolution               =  20925                       EPOCH_node_condition            =  1
```

Computed Satellite Orbit Keplerian Element Set at PERIGEE prior to Epoch from NORAD TLE as Epoch.

```
PERIGEE_year   =  2014         PERIGEE_days_decimal_of_year   =  147.0702033672
PERIGEE_inclination_deg        =   97.9448000000              PERIGEE_right_asc_acnd_node_deg =  207.0616328473
PERIGEE_eccentricity           =    0.0016257000              PERIGEE_argument_of_perigee_deg =   44.6751258926
PERIGEE_mean_anomaly_deg       =  359.9999995488              PERIGEE_mean_motion_rev_per_day =   14.7774423308
PERIGEE_revolution             =   20925                      PERIGEE_node_condition          =  1
```

Move on to next Satellite computing values of 'Keplerian element set at Perigee prior to Epoch'

Continue To next Satellite

(d) ISS (ZARYA) 'Two-Line Elements'(TLE) downloaded on May 28, 2014, 18:13 hrs IST, Satellite launched on November 20, 1998

```
1 25544U 98067A   14148.25353351  .00006506  00000-0  11951-3 0  3738
2 25544  51.6471 198.4055 0003968  47.6724  33.3515 15.50569135888233
```

From this TLE, the Keplerian Element manually interpreted & extracted :

Satellite number 25544, ISS (ZARYA) ,

EPOCH_year = 2014	EPOCH_days_decimal_of_year = 147.2535335100		
EPOCH_inclination_deg	= 51.6471000000	EPOCH_right_asc_acnd_node_deg	= 198.4055000000
EPOCH_eccentricity	= 0.0003968000	EPOCH_argument_of_perigee_deg	= 47.6724000000
EPOCH_mean_anomaly_deg	= 33.3515000000	EPOCH_mean_motion_rev_per_day	= 15.5056913500
EPOCH_revolution	= 88823	EPOCH_node_condition	= 1

Computed Satellite Orbit Keplerian Element Set at PERIGEE prior to Epoch from NORAD TLE as Epoch.

PERIGEE_year = 2014	PERIGEE_days_decimal_of_year = 147.2475593942		
PERIGEE_inclination_deg	= 51.6471000000	PERIGEE_right_asc_acnd_node_deg	= 198.4350551515
PERIGEE_eccentricity	= 0.0003968000	PERIGEE_argument_of_perigee_deg	= 47.6503677902
PERIGEE_mean_anomaly_deg	= 0.0000008019	PERIGEE_mean_motion_rev_per_day	= 15.5074083546
PERIGEE_revolution	= 88823	PERIGEE_node_condition	= 1

Move on to next Satellite computing values of 'Keplerian element set at Perigee prior to Epoch'

Continue To next Satellite

(e) GSAT-14 'Two-Line Elements'(TLE) downloaded on May 28, 2014, 18:14 hrs IST, Satellite launched on January 05, 2014

```
1 39498U 14001A   14146.03167358 -.00000092  00000-0  00000+0 0  1238
2 39498   0.0049 223.9821 0002051 110.2671 354.6468  1.00272265  1407
```

From this TLE, the Keplerian Element manually interpreted & extracted :

Satellite number 39498, GSAT-14 ,

```
EPOCH_year  =  2014            EPOCH_days_decimal_of_year   =  145.0316735800
EPOCH_inclination_deg          =    0.0049000000               EPOCH_right_asc_acnd_node_deg   =  223.9821000000
EPOCH_eccentricity             =    0.0002051000               EPOCH_argument_of_perigee_deg   =  110.2671000000
EPOCH_mean_anomaly_deg         =  354.6468000000               EPOCH_mean_motion_rev_per_day   =    1.0027226500
EPOCH_revolution               =    140                        EPOCH_node_condition            =  1
```

Computed Satellite Orbit Keplerian Element Set at PERIGEE prior to Epoch from NORAD TLE as Epoch.

```
PERIGEE_year   =  2014         PERIGEE_days_decimal_of_year   =  144.0492548835
PERIGEE_inclination_deg        =    0.0049000000               PERIGEE_right_asc_acnd_node_deg =  223.9952481674
PERIGEE_eccentricity           =    0.0002051000               PERIGEE_argument_of_perigee_deg =  110.2408036654
PERIGEE_mean_anomaly_deg       =    0.0000000029               PERIGEE_mean_motion_rev_per_day =    1.0027598249
PERIGEE_revolution             =     140                       PERIGEE_node_condition          =  1
```

Move on to next Satellite computing values of 'Keplerian element set at Perigee prior to Epoch'

Continue To next Satellite

(f) MOON 'Two-Line Elements'(TLE) downloaded on Jun 14, 2014, 16:14 hrs IST, Natural Satellite

1 00000U 00000A 14143.16621081 .00000000 00000-0 00000-0 0 3574

2 00000 18.7965 352.4777 0512000 316.1136 40.2074 00.036600996 0006

From this TLE, the Keplerian Element manually interpreted & extracted :

Satellite number 00000, MOON ,

EPOCH_year = 2014 EPOCH_days_decimal_of_year = 142.1662108168

EPOCH_inclination_deg	= 18.7965000000	EPOCH_right_asc_acnd_node_deg	= 352.4777000000
EPOCH_eccentricity	= 0.0512000000	EPOCH_argument_of_perigee_deg	= 316.1136000000
EPOCH_mean_anomaly_deg	= 40.2074000000	EPOCH_mean_motion_rev_per_day	= 0.0366009960
EPOCH_revolution	= 0	EPOCH_node_condition	= 1

Computed Satellite Orbit Keplerian Element Set at PERIGEE prior to Epoch from NORAD TLE as Epoch.

PERIGEE_year = 2014 PERIGEE_days_decimal_of_year = 139.1147315693

PERIGEE_inclination_deg	= 18.7965000000	PERIGEE_right_asc_acnd_node_deg	= 352.4777171774
PERIGEE_eccentricity	= 0.0512000000	PERIGEE_argument_of_perigee_deg	= 316.1135684193
PERIGEE_mean_anomaly_deg	= 359.9999999986	PERIGEE_mean_motion_rev_per_day	= 0.0366010099
PERIGEE_revolution	= 0	PERIGEE_node_condition	= 1

End of the computing values of 'Keplerian element set at Perigee prior to Epoch' for the six Satellites.

The Epoch corresponds to NORAD TLE (epoch year & epoch day fraction of the year) of the respective satellites.

The Keplerian element at Perigee are often adopted as start point for 'Satellite Pass - Prediction of Ground Trace'.

Next Section - 5.4 Concluding NASA/NORAD 'Two-Line Elements, Keplerian Element & State Vectors.

OM-MSS Section - 5.4 --47

Concluding Satellite Ephemeris Data, NASA/NORAD 'Two-Line Elements, Keplerian Element & State Vectors. (Sections 5.0 to 5.3)

In previous Sections (5.0 to 5.3), the following were presented :

1. Few preliminaries about Satellite Orbit around earth, that included Earth Gravity and Satellite Motion, Velocity equations, Attitude control, Orbit Time period, Orbits types and categories, Ephemeris data and Ground Trace.
2. Explained with examples, the format of NASA/NORAD 'Two-Line Elements'(TLE) as the satellite orbital parameters for six satellites - LANDSAT 8, SPOT 6, CARTOSAT 2B, ISS (ZARYA), GSAT-14 and Moon.
3. Conversion of Keplerian Element Set to State Vector Set and Vice versa, applied to all six satellites mentioned above.
4. Computed Satellite Orbit Keplerian element set at Perigee prior to Epoch for all six satellites, using respective satellite's TLE.

End of preliminaries about 'Satellite Orbit' around earth and Satellite Ephemeris data and convesion utilities of the OM-MSS software.

Next Section - 6 Satellites Motion in Orbit around Earth.

REFERENCES : TEXT BOOKS & INTERNET WEB LINKS.

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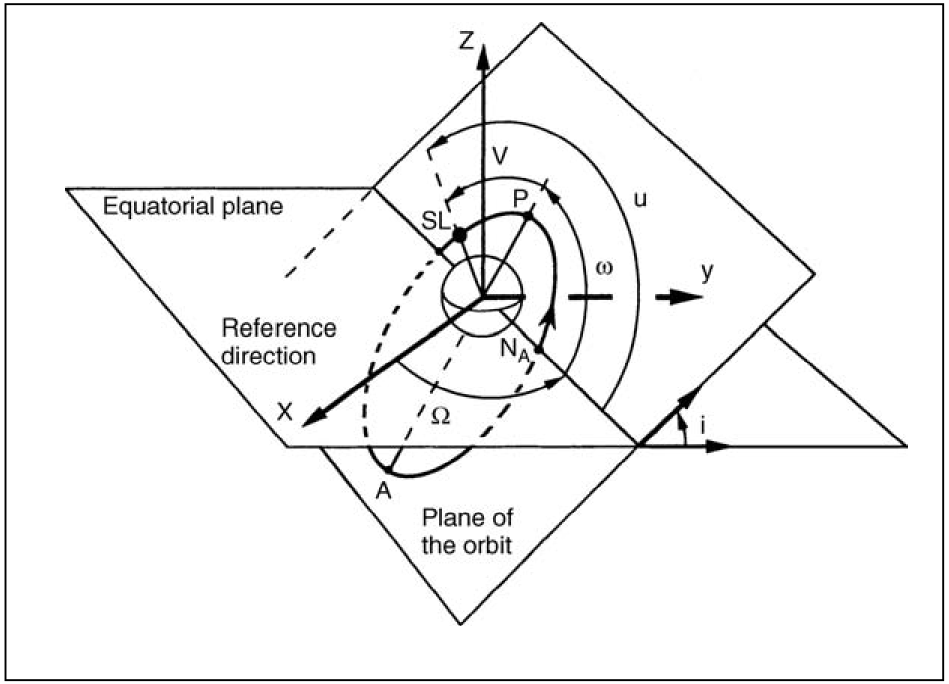

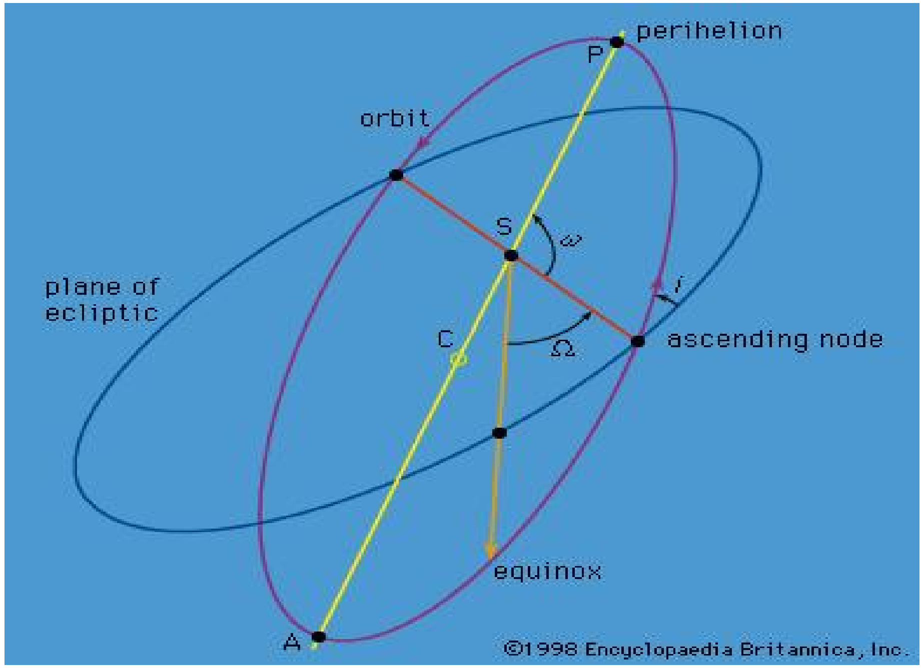

Fig- 4 & 5 Positioning of Orbit in Space

Orbit Position in Space at Epoch is defined by the Values of Kepler Orbit elements : (definations apply to both planets & Satellits)

1. Inclination 'i' of the orbit of a planet, is angle between the plane of planet's orbit and the plane containing Earth's orbital path (ecliptic).
2. Right ascension 'Ω' of the ascending node is the angle taken positively from 0 to 360 deg in the forward direction, between the reference direction and the ascending node of the orbit (the intersection of the orbit with the plane of the equator crossing this plane from south to north).
3. Argument of Perigee 'ω', specify angle between orbit's perigee and orbit's ascending node, measured in orbital plane and direction of motion.
4. Eccentricity '*e*' of an orbit shows how much the shape of an object's orbit is different from a circle;
5. Mean Anomaly '*v*' relates the position and time for a body moving in a Kepler orbit. The mean anomaly of an orbiting body is the angle through which the body would have traveled about the center of the orbit's auxiliary circle. 'M' grows linearly with time.

A knowledge of above five parameters completely defines the trajectory of an object or satellite in space. However, the Nodal angular elongation '*u*' can also be used to define the position of the satellite in its orbit. This is the angle taken positively in the direction of motion from 0 to 360 deg between the direction of the ascending node and the direction of the satellite **($u = \omega + v$).**

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