DECAY OF ESA SPACE OBJECTS

G. Janin and J. de Leeuw

European Space Operations Centre, 6100 Darmstadt, F.R.G.

ABSTRACT

Methods for orbital lifetime predictions are reviewed and the set up of the orbit generation software used at ESOC for lifetime estimation is outlined. After a discussion of the limitations in accuracy of lifetime predictions, a list of all ESRO/ESA objects launched up to now is given, together with their actual or predicted decay dates.

Keywords: Orbit generation, Lifetime estimation, Decay prediction, Satellite re-entry.

1. INTRODUCTION

Once a satellite has completed its operational mission, it does not cease to exist as a satellite and it keeps on orbiting around the Earth. Until when? This is the problem of the orbital lifetime, which will be discussed here.

First of all, it should be stressed that orbital lifetime estimations are not performed just to satisfy our curiosity, but that they are an important part of mission analysis.

Of major concern is, for instance, that the orbital lifetime is longer than the operational lifetime, so that the satellite does not decay before its nominal mission is completed. For some types of orbit, this leads to an essential mission constraint, reflected for instance in the satellite launch window.

Heavy satellites may not burn up completely during atmospheric re-entry, and some solid parts may hit the surface of the Earth. The exact time and place of re-entry has therefore to be predicted accurately in order to warn the inhabitants of the areas concerned about a possible danger. Such a situation has been illustrated in a spectacular way by the re-entries of COSMOS 954 (in January 1978), SKYLAB (in July 1979) and COSMOS 1402 (in February 1983) (for the re-entry of COSMOS, see G. Perry's contribution in these proceedings).

Satellites in geosynchronous orbits do not re-enter before several millions of years. This leads to the problem of crowding of the geostationary ring and the danger of a possible collision. An elegant solution to this problem is to manoeuvre geostationary satellites outside the geostationary ring once their mission is completed, as illustrated by the de-orbiting of ESA satellite GEOS-2 in an orbit about 300 km above the geostationary altitude (Ref. 1).

Even for a planetary orbiter, the orbital lifetime has to be considered. In order to limit a possible contamination of the planet's surface by Earth bacteria, the International Committee of Space Research has recommended a quarantine period of several decades for planetary orbiters before they should be allowed to hit the planetary surface.

2. METHODS FOR ORBITAL LIFETIME PREDICTIONS

Orbital lifetime predictions are estimated by means of an orbit generator. This is a mathematical tool which, given an initial position and velocity of the satellite, will propagate this state for future times, taking into account orbital perturbations.

Among all possible perturbations acting on the orbit, two are mainly responsible for orbital decay: air drag (lowering the apogee) and luni-solar effect (changing the eccentricity).

Depending on the requirements for the accuracy of lifetime predictions, several methods can be envisaged for orbit propagation.

If the accuracy requirement is not very high, the use of a purely analytical method is adequate and will give an output after a short computer time. Another fast way is to use precalculated tables or graphics as the ones developed by RAE Farnborough (see D.G. King-Hele's contribution in these proceedings).

Should analytical methods or graphics not be available or not be sufficiently accurate for the type of orbit considered, one should resort to semi-analytical methods, where only the short-periodic part of the perturbation is treated analytically. A long-term propagation is obtained by repeated applications of the formula on the updated orbital state. The *stroboscopic method*, advocated by E.A. Roth (Ref. 2) at ESOC, is based on such an idea. For the case of highly eccentric orbits, where a fully analytical treatment of the long-term effect of third-body perturbations is not feasible, such a semi-analytic method has revealed to be highly efficient. It is invaluable for launch window estimations requiring long-term trajectory calculations for a grid of launch dates and hours.

Precise orbit prediction resorts most of the time to numerical integration of the equations of motion, where a perturbation model as realistic as needed can be introduced.

This discussion shows that a large variety of tools has to be available to cover the various needs of orbit propagation. In order to ease the handling of the corresponding software packages, they have been included in a Unified System for Orbit Computation, which will be briefly described in the following section.

3. SOFTWARE FOR ORBIT GENERATION

By using the Uniform System for Orbit Computation (USOC, Ref. 3), a lifetime estimation for a satellite is performed by executing the three following steps:

1) *Preprocessing*. Through an interactive programme, the user is invited to choose among a large variety of available methods, and to input numerical values for the parameters. The preprocessor prepares a corresponding programme, tailor-made to the particular application of the user, by calling modules from a software modules' library, and by creating an input data file.

2) *Processing*. The programme created during preprocessing is executed and an output file is produced.

3) *Postprocessing*. A detailed print-out of the output file is made, or a set of graphical representations is offered.

The programme built up during preprocessing is composed of five parts:

1) definition of the coordinate system and the initial state,

2) formulation of the differential equations of the motion,

3) integration method,

4) force model,

5) stopping conditions.

Each of these parts is represented in the computer by a choice of functional modules corresponding to different cases, methods, models, etc. Modules accomplishing the same function are interchangeable, they possess the same FORTRAN name and identical calling sequences. They are homonymes.

For numerical integration in particular there are homonyme modules for:

- the formulation of the equations of motion, which can be in cartesian coordinates, with or without time transformation, or in regularized variables, or in orbital elements, etc.

- the integration method, which can be single-step, with or without automatic stepsize control, or multi-step,

- the perturbation model,

- the stopping condition, which can be defined by any arbitrary function of the state (given time, given altitude, sphere of influence, etc.).

In this flexible way, the most appropriate functional module can be taken out of the library and plugged into the programme for maximal efficiency. The library of functional modules can be extended indefinitely without complicating the structure of the application programmes, which is frozen.

The choice of the proper method or model for a given application is left to the experience of the user. However, some general advice can be given. For instance, when integrating satellite orbits numerically

- it is preferable to have a mathematically elaborated formulation of the equations of motion combined with a straightforward constant stepsize integration method, rather than having a simple formulation in cartesian coordinates associated with a sophisticated stepsize control;

- in order to prevent the accumulation of round-off errors, long-term orbit propagation should be performed with a large stepsize multistep method and a refined formulation of the equations of motion;

- short-arc propagations are better accommodated with a singlestep method;

- when a small stepsize is needed for high precision propagation, the importance of the formulation of the differential equations of motion vanishes and a simple formulation should be chosen in order to reduce the computation overhead.

More details can be found in the CNES course on Space Technology (Ref. 4).

4. HIGH ECCENTRICITY ORBITS

When the orbital eccentricity is close to 0.9, the orbit is considered as highly eccentric. Such orbits are popular for scientific satellites aimed at exploring the Earth magnetosphere at 20 or more Earth radii. ESA has already launched five such satellites (HEOS-A1, HEOS-A2, COSB, ISEE-2 and EXOSAT). In the early nineties it is planned to launch a group of four satellites (CLUSTER) also in a high eccentricity orbit.

The dominant perturbation for such orbits is the gravitational disturbance by third-body, for an Earth orbiter: the luni-solar perturbation. Its long-periodic effect can hardly be treated analytically and one has to resort to a semi-analytical method or to numerical integration.

On the other hand, there are practically no uncertainties associated with third-body perturbations: third bodies can be treated as point masses and their ephemerides are known to a high accuracy. The third-body effect on the satellite orbit can therefore be predicted with high precision and so can the satellite's decay. The major third-body influence on the satellite's lifetime is an eccentricity change, affecting the perigee height. Should this height decrease below the Earth surface, the satellite has ceased to exist.

For most of the satellites orbiting in high eccentricity orbits, a re-entry prediction accuracy of a few days after a lifetime of several years is realistic. This has been confirmed by the re-entry of HEOS-A2 (Ref. 5) and HEOS-A1 (Ref. 6). The re-entries of COSB, ISEE-2 and EXOSAT are briefly discussed in Ref. 7. The perigee history prediction for COSB and ISEE-2 is shown in Fig. 1.

However, as explained in Ref. 6, there are cases when a last minute surprise can occur due to the atmospheric drag when the perigee height is low enough in the atmosphere.

The dissipative effect occuring during the brief passage of the satellite in the Earth atmosphere may reduce the orbital energy significantly and lower the apogee height. The luni-solar effect is reduced and the perigee height stabilized. As a consequence, the lifetime is extended. This paradoxical situation - the drag causes an increase in the lifetime - occurs particularly for light objects such as rocket third stages, dual launch structures, etc.

Objects in geostationary transfer orbits (eccentricity e = 0.73), being less subjected to third-body perturbations than scientific satellites, are particularly vulnerable to perigee drag, and it is not unusual that the orbit of an Ariane third stage or Sylda becomes almost circular at an altitude between 100 and 500 km before decay.

As will be recalled in the next section, drag perturbation cannot be exactly modelled, therefore the decay prediction of light objects in geostationary transfer orbits is very uncertain, and prediction errors of several months or sometimes even years are to be expected.

5. NEAR-EARTH ORBITS

The dominant perturbation influencing the lifetime of a satellite in a low altitude orbit is the atmospheric drag. The drag acceleration depends mainly on the three following parameters:

1) the ballistic coefficient,

2) the effective area to mass ratio,

3) the atmospheric density.

A large uncertainty is attached to these parameters, in particular

- the effective area depends on the satellite attitude, which is unknown for satellites no longer in operation;

- the mass depends on the amount of propellant left in the tank, which is often not exactly known;

- the atmospheric density depends on the solar activity and the geomagnetic index, which are poorly predictable.

As a consequence, the lifetimes of near-Earth satellites cannot be accurately estimated, and an uncertainty of at least 10 % is to be expected. This means that even three weeks before decay, the re-entry date is known only to ± 2 days. Even one week before re-entry, there is an uncertainty of several revolutions. Only during the last day does it become sensible to predict the final arc of

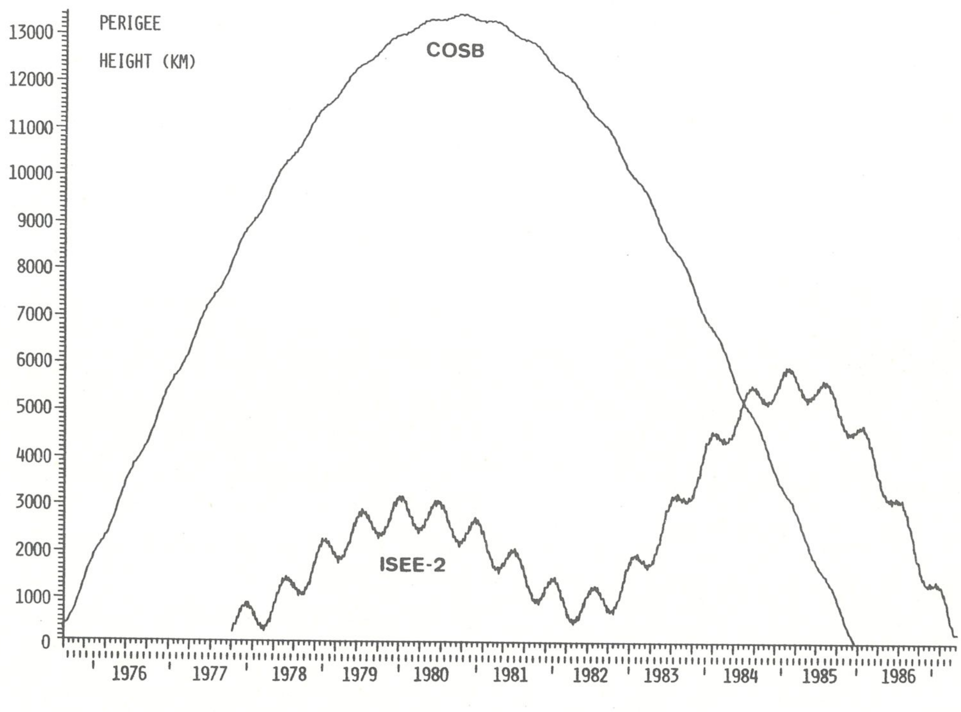

Figure 1. Perigee height history prediction for COSB and ISEE-2

decay.

Orbit generators for near-Earth circular orbits can rely on a large part on analytical theories. In Ref. 8, a method based on an averaging procedure including the Earth's zonal harmonics J_2, J_3 and J_4 and a refined treatment of the air drag perturbation is described. The air density model chosen in this method is the MSIS 77 atmosphere from Hedin et al. (Ref. 9). This model is also available as a homonyme module for numerical orbit computation.

This analytical orbit generator is used for EURECA mission analysis. It was also used for the ESOC re-entry prediction of COSMOS 1402A and C in January-February 1983.

Such an orbit generator is not adequate for simulating the last phase of re-entry, when the satellite begins to enter into the dense atmosphere.

A so-called *re-entry point* can be arbitrarily defined by the position of the satellite when it reaches a certain critical height above the surface of the Earth. At this point, usually chosen at an altitude of 130 km, the satellite experiences an increasing drag leading to a gradual reduction of the horizontal forward velocity.

The behaviour of the satellite during this deceleration phase is difficult to predict. One problem is that up to now it has not been possible to make direct continuous measurements in the atmosphere between 80 km and 250 km, and atmospheric models for this range of altitudes are very uncertain.

By using a density function extrapolated toward low altitudes from a standard model and a simple shape for the satellite, a numerical integration of the equations of motion can be attempted to obtain a rough idea of the satellite trajectory.

Such an attempt is illustrated in Fig. 2 for the case of HEOS-A1.

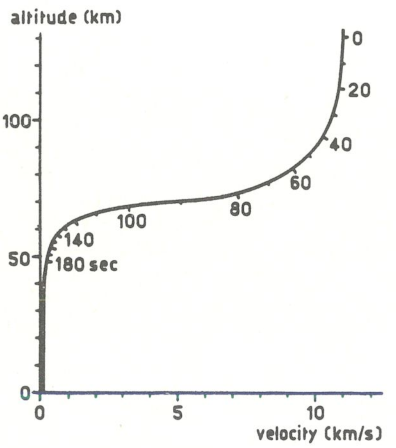

Figure 2. HEOS-A1 calculated velocity and altitude in terms of the time measured in seconds since re-entry point crossing (at 130 km height).

During the descent from a height of 100 km to 60 km, the satellite is subjected to a very high g-load and consequently also a high heat-load. It is therefore most likely that during this time the satellite disintegrates and burns up. If there are remaining parts, they follow individual paths according to their cross-section to mass ratio. Obviously, it is not possible to describe this phase in detail.

6. A LIST OF ESA OBJECTS AND THEIR DECAY PREDICTION

ESA (and formerly ESRO) has launched successfully 22 satellites up to now. Among them, 14 are still in orbit:

3 in a highly eccentric orbit (COSB, ISEE-2, EXOSAT);

9 in a 24-hour orbit (METEOSAT-1, IUE, OTS-2, GEOS-2, METEOSAT-2, MARECS-A, ECS-1, ECS-2, MARECS-B2);

1 in a 12-hour orbit (GEOS-1);

1 in a heliocentric orbit (GIOTTO).

Only the three scientific satellites in highly eccentric orbits are due to re-enter:

- COSB. On the day of submission of this paper, COSB, launched on 9 August 1975 in a 0.88 eccentricity orbit, is predicted to re-enter on 6 January 1986. This prediction is based on the orbital elements provided by the last ESOC orbit determination run of April 1982. The satellite has been switched off on 26 April 1982, after an operational lifetime of more than 6 years, and its tracking was ended. NORAD does not provide elements for this satellite.

- ISEE-2. This satellite, launched on 22 October 1977 in a 0.91 eccentricity orbit, is still operational, and is predicted to re-enter on 25 September 1987.

- EXOSAT. Launched on 26 May 1983 in a 0.90 eccentricity orbit, the actual prediction for EXOSAT's re-entry is for June 1986. However, sufficient on-board propellant is available so that perigee height increase manoeuvres can be performed allowing an orbital lifetime extension until 1987.

When a satellite is launched, it is accompanied by some debris such as rocket third-stage, adaptors, etc. A complete list of all objects launched by ESA/ESRO is shown in Table 1.

The first column of Table 1 gives the COSPAR international designation of the object.

The second column gives the name of the object or its type.

The NORAD number, if available, is shown in the third column.

The launcher is recalled in the fourth column. For an Ariane launch, the type of launcher and its number (the eight first promotional launches under ESA's supervision are numbered L01 to L08 and the subsequent launches under the responsibility of Arianespace are numbered V09, V10, ...) are also given.

COSPAR designation	name	NORAD no.	launcher		launch date	apogee (km)	perigee (km)	incl. (deg)	period (hour)	re-entry date
1968-041A	ESRO-2	3233	scout		68/05/17	1086	326	97.20	1.648	71/05/08
1968-084A	ESRO-1A	3459	scout		68/10/03	1538	258	93.76	1.717	70/06/26
1968-109A	HEOS-A1	3595	TD		68/12/05	223440	418	28.28	112.493	75/10/28
1969-083A	ESRO-1B	4114	scout		69/10/01	389	291	85.11	1.523	69/11/23
1972-005A	HEOS-A2	5814	TD		72/01/31	245380	396	89.81	128.480	74/08/02
1972-014A	TD-1A	5879	TD		72/03/12	551	524	97.55	1.590	80/01/09
1972-092A	ESRO-4	6285	scout		72/11/22	1173	245	91.11	1.650	74/04/15
1974-070A	ANS	7427	scout		74/08/30	1173	258	98.03	1.652	77/06/14
1975-072A	COSB	8062	TD		75/08/09	99873	342	90.13	37.112	86/01/06?
1977-029A	GEOS-1	9931	TD		77/04/20	38357	2110	26.25	12.001	indef.
1977-102B	ISEE-2	10423	TD		77/10/22	132815	5363	31.26	57.315	87/09/24?
1977-108A	MET-1	10489	TD		77/11/23	--------	geostationary		--------	indef.
1978-012A	IUE	10637	TD		78/01/26	45888	25669	28.63	23.928	indef.
1978-044A	OTS-2	10855	TD		78/05/11	--------	geostationary		--------	indef.
1978-071A	GEOS-2	10981	TD		78/07/14	--------	geostationary		--------	indef.
1979-104A	CAT-1	11645	AR-1	L01	79/12/24	22968	187	17.8	6.655	91/11?
- B	3rd st.	11659	-	-	-	36000	202	17.6	10.558	82/11/14
1981-057A	MET-2	12544	AR-1	L03	81/06/19	--------	geostationary		--------	indef.
- C	3rd st.	12546	-	-	-	33227	253	10.87	9.854	>1995
- D	CAT-3	12562	-	-	-	24932	231	10.49	7.214	>1995
- E	adapt.	14125	-	-	-	28112	244	10.28	8.142	>1995
1981-122A	MARECS-A	13010	AR-1	L04	81/12/20	--------	geostationary		--------	indef.
- B	CAT-4	13011	-	-	-	34325	238	9.90	10.065	>1995
- C	3rd st.	13025	-	-	-	19654	229	10.35	5.755	88/04?
1983-051A	EXOSAT	14095	TD		83/05/26	190054	2191	72.8	90.714	86/06?
1983-058A	ECS-1	14128	AR-1	L06	83/06/16	--------	geostationary		--------	indef.
- C	3rd st.	14130	-	-	-	32002	232	8.44	9.329	>1995
- D	sylda	14151	-	-	-	15658	196	8.54	4.733	86/07?
1983-105B	3rd st.	14423	AR-1	L07	83/10/19	15151	141	8.30	4.591	86/01?
1984-023B	3rd st.	14787	AR-1	L08	84/03/05	31628	295	10.95	9.234	>1995
1984-081A	ECS-2	15158	AR-3	V10	84/08/04	--------	geostationary		--------	indef.
- C	Sylda	15165	-	-	-	31847	241	6.66	9.284	86/08?
- D	3rd st.	15166	-	-	-	33707	662	6.63	10.003	>1995
1984-114B	MARECS-B2	15386	AR-3	V11	84/11/10	--------	geostationary		--------	indef.
- C	3rd st.	15388	-	-	-	35912	346	7.13	10.604	>1995
- D	Sylda	15389	-	-	-	24331	175	6.95	7.032	86/01?
1985-056A	GIOTTO		AR-1	V14	85/07/02	--------	heliocentric		---------	indef.
- B	3rd st.	15876	-	-	-	35393	206	7.02	10.394	>1995

Table 1. List of ESRO/ESA catalogued objects. Status on December 1, 1985.

After the launch date indicated in the fifth column, the apogee, perigee, inclination and period of the satellite, as currently available, are indicated. Should the object be already decayed, the initial elements after injection into the operational orbit are recalled.

The last column indicates the actual decay date. Should the object still be in orbit at the time of submission of this paper for publication, the expected decay date is indicated with an interrogation mark.

In order to limit the computer time usage, lifetimes are estimated only for a period of 10 years.

The input elements for the lifetime calculation are taken from recent ESOC orbit determination results or from the so-called NASA 2-line elements distributed by the Goddard Space Flight Center.

Table 1 indicates the situation on 1 December 1985. Such a table is updated at ESOC on a regular basis.

One should note that satellites no longer in operation in geostationary orbits, such as METEOSAT-1 and GEOS-2, are subject to a periodic longitude drift and a change of the orbital inclination. They are therefore no longer strictly geostationary. This is particularly true for GEOS-2 which, as already mentioned in Chapter 1, has been de-orbited into a higher orbit.

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