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COMMITTEE ON THE PEACEFUL USES OF OUTER SPACE
Scientific and Technical Sub-Committee
Working Group on Nuclear Power Sources

QUESTIONS RELATING TO THE USE OF NUCLEAR POWER SOURCES IN OUTER SPACE

Japan: working paper

Studies on technical aspects and safety measures of nuclear power sources in outer space

In accordance with paragraph 42 of the report of the Working Group on the Use of Nuclear Power Sources in Outer Space established by the Scientific and Technical Sub-Committee, the Government of Japan wishes to submit the following studies on technical aspects and safety measures of nuclear power sources (NPS) in outer space consisting of four papers.

These papers were written by four specialists listed below on the basis of discussions on the matter among them and other specialists of Government-related research laboratories, etc. and the future policy of the Government of Japan will not be restricted by these papers.

The Government of Japan hopes that these studies will contribute in some way to further examination of this question at the Scientific and Technical Sub-Committee and its Working Group on Nuclear Power Sources.

The four papers are:

1. List of Safety Problems Involved in the Use of NPS in Outer Space
by Dr. Yoshihiki Kaneko, Principal Engineer, Division of Reactor Engineering, Japan Atomic Energy Research Institute.

2. Implementation of ICRP Recommendations for Populations and the Environment in the Context of Space Vehicles Utilizing NPS
by Dr. Jiro Inaba, Senior Researcher, Division of Environmental Health, National Institute of Radiological Sciences.

3. A Note on Satellite's Orbit and Re-entry
   by Dr. Tatsuo Yamanaka, Senior Researcher, Space Research Group, National Aerospace Laboratory.

4. Format of notification
   by Mr. Toshiyasu Sasaki, Director, Space Activities Planning Division, Research Co-ordination Bureau, Science and Technology Agency.

The above papers are reproduced as annexes 1, 2, 3 and 4.

ANNEX 1.

LIST OF SAFETY PROBLEMS

1. Preface

A spacecraft carrying nuclear power sources (NPS) is required to have reliability for larger than that without NPS, in view of the radiological hazards. Therefore, safety measured should be taken in each phase starting with launching. In Section 2, technical problems connected with the safety of the NPS system are listed. Section 3 mainly lists technical problems of the burn-out of NPS in case of a re-entry accident following the failure in transfer of NPS to higher orbit or the failure in retrieval of NPS. Finally, in Section 4 is listed information on NPS useful for the evaluation of the radiological hazards in the re-entry accident. It is desirable that each item listed in this paper be examined in detail in the near future.

2. Technical problems connected with the safety of the NPS system

(1) Techniques for evaluating reliability of spacecraft carrying NPS

(For example, the fault tree method is applicable.)

(2) Techniques for ensuring safety of NPS

(i) Compilation and analysis of the information on accidents

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(It is useful for evaluation of the safety design of NPS to analyse the experiences in abnormal phenomena and accidents.)

(ii) Safety measures at the time of launching

(Techniques for retrieval of NPS in case of the failure of launching)

(iii) Safety measures in case abnormal phenomena occur in orbit

(Techniques for transferring NPS into higher orbit or retrieving it in case NPS runs abnormal or out of control from the ground station.)

(iv) Plans following the completion of mission

(Techniques for transferring NPS into higher orbit or retrieving it (after fairly long cooling time in orbit in case of a nuclear reactor))

(v) Safety design of NPS

(a) Safety and shut-down devices

(Techniques for having safety and shut-down devices of NPS (nuclear reactor) actuated directly by abnormal phenomena in the reactor core in the orbit phase and by the deacceleration force in the re-entry accident.)

(b) Integrity of ambient gas pressure in NPS

(Techniques for retaining the NPS ambient gas pressure within a suitable range with high reliability.)

(c) Nuclear fuel or radioisotope of NPS

(Use of such materials as will minimize radiological contamination of the environment in case of accidents.)

(d) Remote sensing technique for diagnosis of abnormal phenomena of NPS

(e) Behaviour of NPS in case abnormal phenomena such as loss of the coolant occur.

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(3) Techniques for predicting lifetime of spacecraft carrying NPS

Techniques for predicting the lifetime of spacecraft equipped with NPS to be used in determining the time to initiate the transfer of NPS into higher orbit or the retrieval of NPS before the occurrence of abnormal phenomena which are relatively more frequent at the last stage of the lifetime.

(4) Other engineering problems in designing NPS

(i) Efficiency of heat-to-electricity conversion system

(ii) Effect of the mechanical shock at the time of launching

3. Technical problems related to re-entry accidents such as burn-out of NPS

(1) Mechanical devices and structure to divide NPS (nuclear reactor) into small pieces on an accidental re-entry accident to enable the NPS to burn out, in case of a spacecraft launched on the premise that the NPS on board will be retrieved.

(2) Burn-out characteristics of NPS

Elements such as fuel, moderator, structural materials, reactor vessel and radioisotope heat generator

(3) Criticality in case of NPS falling into water

4. Information on NPS useful for the evaluation of radiological hazards in the re-entry accident

The following are NPS information items required to be announced before accidental NPS re-entry for the evaluation of radiological hazards.

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(1) Outline of the NPS structure

(i) Composition, size and weight of the fuel and reflector elements (nuclear reactor), heat generating elements (radioisotope generator) and structural elements

(ii) Reactor vessel and out-of-core components strongly irradiated

(iii) Information on neutron spectra; thermal or fast reactor

(2) History of NPS operation

(i) Thermal and electrical output and neutron flux density averaged over reactor core volume

(ii) Operating time from start-up to shut-down and cooling time before actual re-entry

(iii) Build-up of fission products for nuclear reactor; and, nuclide and radio-activity in curie units for radioisotope generator

(3) Information on impact time and area

(4) Size, weight and radiological activities of the components of spacecraft carrying NPS expected to fall down to the ground surviving re-entry

(5) Information useful for identifying falling materials as fragments of the spacecraft carrying NPS.

ANNEX 2

EFFECT OF RADIATION FROM NUCLEAR POWER SYSTEMS BUILT IN THE SPACECRAFT ON POPULATIONS AND THE STANDARDS OF ICRP

This paper is dealing with the human cost i.e., collective dose equivalent, of utilizing nuclear power sources (NPS) as an energy source in a spacecraft and with the question whether or not this cost is approved from a standpoint of the recent phylosophy of radiation protection.

This paper consists of three sections. In the first section, the biological effects of radiation and some features of exposure to radiation from NPS built in space-crafts are described very briefly. In the second section, approximate estimates of dose equivalent due to NPS are calculated for three assumed typical cases. In the third section, problems involved in the use of NPS in spacecrafts are pointed out based on the above-mentioned calculated dose and the standards of International Commission on Radiation Protection.

1. Biological effect of radiation and some features of exposure to NPS radiation

Radiation may cause various types of biological effects. The radiation effects can be divided into the acute effect and the late effect from the length of latent period between

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exposure and appearance of the effect. The effects can be classified into the somatic and the genetic, the former is expressed in the exposed individual and the latter expressed in his descendants. A convenient method of classification from the recent radiation protection principle is to devide them into stochastic effects and non-stochastic effects.

Stochastic effects are those for which the probability of an effect occurring, rather than its severity, is regarded as a function of dose without threshold. Genetic effect as well as radiation carcinogenesis are regarded as being stochastic. Non-stochastic effects are those for which the severity of the effect varies with the dose and for which a threshold occurrs. Cataract of the lens, non-malignant damage to the skin and cell depletion in the bone marrow are examples of non-stochastic effect of radiation. Thus, the facts that radiation exposure causes extremely serious diseases such as cancer and genetic defect and that the relationship between the probability of such effects and dose remains down to the lowest dose level, distinguish radiation from other factors, such as chemical pollutants, which cause deletarious effect and for which thresholds are usually assumed to occur. As to stochastic effect we do not have a corroborative evidence to show that there exists linear relationship between the risk and the dose even within a few rems or less, and then the above-mentioned principle of the linear relationship without threshold may well be called a conservative hypothesis for the radiation protection purpose.

When we use NPS as an energy source in a spacecraft, how are we exposed to radiation due to NPS? Two types of NPS which can be radiation sources have already been used. One is a radio-isotope (mainly ^{238}Pu) power generator and the other is a fission reactor generator. Exposure modes vary according to its radiation source. These NPS built in spacecrafts might cause very small exposure if the operation of the spacecraft could be controlled normally. However, the possibility of radiation exposure cannot be denied, because the spacecraft may be subject to an accident or emergency case which may take place during launch and after achieving spacecraft orbit. Re-entry of spacecrafts under uncontrolled condition is the most likely accident and the results may be classified fundamentally into three cases. The first is the case of complete burn-out of NPS in the atmosphere causing global contamination with radioactivity. SNAP-9A was the case. However, an estimation shows that complete burn-out of the reactor is unlikely. The second is the imcomplete burn-out of NPS and the radioactive materials come down to rest in local area, like in the example of Cosmos 954. The third is the intact NPS touching down on land and becoming an external radiation source. Of course, many other variations of these cases can be considered. As the result of these cases, the following three basic modes of exposure can be considered; (1) internal exposure by inhalation of airborne radioactive aerosols which are

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released and burned-out in atmosphere, (2) internal exposure by ingestion of foods and water contaminated with released radioactivity, and (3) external exposure to direct gamma radiation from the sources.

2. Population dose from NPS

Radiation dose received by the world population or each individual following the re-entry of NPS built in the spacecraft is calculated below. The readers should keep in mind that only typical cases are adopted and that many assumptions are introduced for simplifying the calculation. Therefore the results might be approximation.

Case 1 Population dose due to complete atmospheric burn-out of radioisotope (^{238}Pu) power generator

We assume that a spacecraft with a battery containing 10,000 Ci ^{238}Pu re-entered and burned out in the atmosphere. Vaporized ^{238}Pu is supposed to spread all over the world. The collective dose equivalent commitment due to this radiation source is calculated as follows. The principal dose calculation is based on the report of the United Nations Scientific Committee on the Effect of Atomic Radiation. UNSCEAR, reviewing radioactivity survey data published in various countries, calculated the dose factor between atmospheric inventory of ^{239}Pu from the past nuclear tests and the collective dose commitments to the lung and to bone lining cells to be an order of 10 person-rad per Ci. Assuming that this dose factor can be applied for ^{238}Pu, the population dose is

$$10{,}000 \text{ Ci} \times 10 \text{ person-rad Ci}^{-1} = 100{,}000 \text{ person-rad.}$$

Almost all of this dose commitment is caused by alpha radiation from the ^{238}Pu. Since the appropriate measure for risk estimation due to radiation exposure is dose equivalent rather than absorbed dose, rad is necessary to be converted into rem. Taking the effective value of the quality factor of alpha particle to be 20 from ICRP Publication 26, the population dose equivalent is

100,000 person-rad x 20 = 2000,000 person-rem.

The world population is roughly estimated to be 5×10^9.

If the population dose equivalent commitment can be divided meaningfully by the present world population, the average individual dose would be nearly 0.4 mrem. According to the ICRP Publication 26, the risk factors based upon the estimated likelihood of inducing fatal malignant disease for lung and bone are 2×10^{-5} rem^{-1} and 4×10^{-6} rem^{-1}, respectively. This means that the excess risk by lung cancer and bone cancer due to 0.4 mrem after atmospheric burn-out of a spacecraft with 10,000 Ci ^{238}Pu generator is 8×10^{-9} and 1.6×10^{-9}, respectively.

Case 2 Population dose due to atmospheric burn-out of fission reactor

Population dose due to atmospheric burn-out of the fission reactor built in a spacecraft at re-entry into atmosphere can also be calculated based on the dose factors which appear in the UNSCEAR report. Assuming that the thermal output of the reactor is 100 KWT and that the spacecraft re-enters immediately after 1 year operation, the

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inventory of fission products in the reactor is roughly 500,000 Ci. Major radioactivities which constitute major part of human exposure are as follows;

85 Kr	2×10^{1} Ci,
89 Sr	4×10^{3} Ci,
90 Sr	1×10^{2} Ci,
106 Ru	2×10^{2} Ci,
137 Cs	1×10^{2} Ci, and
144 Ce	3×10^{3} Ci.

Using these values and the dose factors cited in the UNSCEAR report for these radionuclides, the world population dose commitments due to this radiation source can be calculated to be almost 10,000 person-rad. Since the almost all of this dose is due to beta and gamma ray radiation for either of which the quality factor is unity, 10,000 person-rad means 10,000 person-rem and the average individual dose is roughly 0.002 mrem. In the Publication 26 the ICRP concludes that the mortality risk factor for radiation induced cancer is about 10^{-4} per rem, as an average for both sexes and all ages. This means that the excess risk for an individual is 2×10^{-10} after receiving 0.002 mrem from the atmospheric burn-out of a 100 KWT fission reactor generator.

Case 3 Exposure to direct radiation from a fission reactor generator

When a fission reactor generator built in a spacecraft re-enters into the atmosphere and reaches on the earth without burn-out or crash, people in the neighbour of the reactor

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receive direct gamma radiation from the fission products in the reactor. The following assumptions are made in the calculation. Thermal output of the reactor is 100 KWT. The reactor have been operated for one year and about 500,000 Ci of fission products are accumulated in the reactor immediately after the shutdown of the reactor. Uranium content of the reactor core is 50 Kg. All of the fission products are at the center of the core. Beryllium reflector is 15 cm thick. Gamma ray fraction of the radioactivity is one third. Average energy of the gamma ray from this source is 0.5 MeV and the exposure rate at 1 m apart from the source is 0.5 Rh^{-1} Ci^{-1}. Using these parameters, the exposure rates at 1 m, 10 m and 100 m apart from the center of the source which lies down on the earth can be calculated to be 315 Rh^{-1}, 3.15 Rh^{-1}, 31.5 m Rh^{-1}, respectively. Radioactivity of fission products in the reactor decreases rapidly. The decreasing rate is very high at earlier phase after the shutdown and becomes lower with the elapsed time, i.e., about 2 weeks after the shutdown the radioactivity becomes about one tenth of the initial value and it reaches less than one hundredth at one year after the shutdown. Taking account of the decrease of the radioactivity, it can be estimated that people should not live within 367 meters from the source in order to avoid radiation dose at the level of 500 mrem per year which corresponds to dose limit to a member of the public. In this connection the population

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density for Japan is about 300 person per Km^2 and that for Tokyo is about 5,300 person per Km^2. Owing to this rapid decrease of radioactivity, it can also be estimated that the exposure rates at 1, 10 and 100 meter apart from the source which has lied down on the ground for one year since the shutdown of the reactor are 3.15 R/h, 31.5 mR/h and 0.315 mR/h, respectively.

3. Questions related to the use of NPS from the viewpoint of ICRP standards

According to the ICRP Publication 26, the aim of radiation protection should be to prevent detrimental non-stochastic effects and to limit the probability of stochastic effects to the acceptable levels. In order to achieve this aim, ICRP recommends a system of dose limitation, the main features of which are as follows: a) no practice shall be adopted unless its introduction produces a positive net benefit (Justification), b) all exposure shall be kept as low as reasonably achievable (ALARA)(Optimization) and c) the dose equivalent to individuals shall not exceed the limits recommended for the appropriate circumstances (Dose limits). At this statement we should recognize that the dose limit, of which numerical value such as 5 rems or 500 mrem per year is so famous, is merely one of the factors which constitute the system of dose limitation widely accepted as a fundamental principal of radiation protection.

I. should be additionally commented that ICRP recognized two exposure conditions, of which one has the measures to

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limit the human exposure and the other has not, while the system of dose limitation by ICRP is applicable only to the former case where the human exposure will be caused by controlled radiation sources. On the other hand, the exposure by NPSs in spacecrafts is not necessarily limited because of its accidental occurrence as described in the section 1. Hence, it could be considered that the system of dose limitation is not appropriately applied to the exposure to radiation from NPS in space.

When we dare to judge the use of NPS in spacecrafts from the phylosophy of ICRP mentioned above, there exist some problems and difficulties. Alternatives such as solar cells are available in practically all space missions except special missions such as Voyage mission. The most important problem involved in the use of NPS in spacecrafts exists in the fact that individual persons or the public of most countries other than the launching country have likely to bear undue risk without accepting benefit.

In the case of complete burn-out of either a plutonium generator or a fission reactor generator, the calculated individual dose equivalent is certainly far below the dose limit (500 mrem per year) recommended by ICRP. However, we should recognize that the world population collective dose is considerably high. When highly radioactive fragments pass through atmosphere and touch down on the earth, the occurrence of radiation exposure near dose limit can be

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supposed. In the Case 3 described in the previous section, some persons are likely to receive dose equivalent larger than the limit, moreover if reflectors on a spacecraft are detached, the dose will be rather higher. The evacuation of the people from a considerably large area may be needed, and this will be a very difficult problem in the populated area. From a standpoint of protective actions against accident, NPS built in a spacecraft is in very special situation where the technological as well as spacial assurance for safety is extremely difficult.

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ANNEX 3

A NOTE ON SATELLITE'S ORBIT AND RE-ENTRY

1. Introduction

Spacecrafts are susceptible to various kinds of forces in orbits. Due to the forces, a spacecraft decays the altitude and finally re-enters into the atmosphere, and there it may disintegrate and be dispersed minutely and/or may crash against the earth in one or more pieces. Accordingly, it is important to make an accurate prediction of the orbital lifetime and re-entry trajectory of the spacecraft in order to prepare for the accidental re-entry.

It is very difficult to predict accurately the spacecraft's decay date some months or years ahead. The reason comes from the difficulty in predicting accurately the primary forces, which change the orbital elements of the spacecraft. The forces, which cause to change the orbital elements of the spacecraft, are listed as follows:

(1) perturbative influence of the earth's asphericity,

(2) gravitational attractions of the sun and the moon,

(3) solar radiation pressure,

(4) aerodynamic forces, and

(5) artificial forces such as result from on-board thruster firing and/or out gassing.

The perturbative effects of the forces due to the items (1), (2) and (3) are well-known with certain accuracy. The effect of the item (5) can be well estimated if information

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on the spacecraft system is available. Aerodynamic forces, the item (4), are the most important for the accurate prediction of the orbital lifetime and re-entry trajectory.

Aerodynamic forces acting on a spacecraft depend strongly on the atmospheric density. The predition of the atmospheric density, therefore, is important for that of the orbital lifetime and re-entry trajectory. An orbital lifetime of a spacecraft with perigee height above 300km is measured by months or years, and it is needed to predict accurately an averaged air density profile through the long lifetime. The lifetime of near-circular orbit with perigee height below 300km is measured by hours or days. It becomes important to predict accurately the date and hour of the initiation of re-entry and the re-entry trajectory. In the latter case, it is needed to predict accurately the local variation of the atmospheric density and the motion of the upper atmosphere. Aerodynamics at these lower altitudes is in so called "transition regime". Aerodynamics in this regime is important for the prediction of re-entry trajectory. It is, however, one of the most difficult problems of today's space science and technology to estimate accurately the aerodynamic forces in the transition regime and to predict the future variation of the earth's atmosphere.

In this paper, the current state of the art of study of the atmosphere is described in the second section, and that

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of aerodynamics of free molecular and transition regime is in the third section. In the fourth section, some problems of improving the accuracy of orbit determination (OD) and orbit prediction (OP) by satellite tracking are discussed. The so-called "burn-up and dispersion in the atmosphere" of falling components by aerodynamic heating during re-entry is discussed in the fifth section, remarking reflectors of space nuclear reactors. Finally, an impact of attaching re-boost system or re-entry control system on spacecraft design is discussed.

2. Earth's atmosphere and its global motion

Until densities in the upper atmosphere began to be measured by sounding rockets, very little was known about the atmosphere above 100km, where orbital lifetime of spacecraft strongly depends on the densities. Since the first artificial satellite was launched in 1957, many atmospheric models have been presented by drag measurements from the orbital decays of the satellites. The early models have been static; i.e. they do not vary with time. It has been known since 1958 that the density of the upper atmosphere experiences large variations. Nonweiler[1)] et al. suggested that these variations were correlated with solar activity. Many efforts to correlate the density fluctuations with an index of solar flare activity have been done. In the course of this investigation, Paetzold[2)] et al. discovered

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the annual and semi-annual variations of atmospheric density. In the meanwhile, Bartels[3] et al. had predicted and Jacchia[4] had discovered a variation in density correlated with geomagnetic disturbance. With the development of a set of theoretical models by Nicolet[5] in 1961, the construction of model atmosphere took a new turn. Nicolet assumed that the density could be represented as a function of a single parameter, temperature. This assumption, together with the assumption of fixed boundary condition at 120km and diffusive equilibrium above 120km, enabled Nicolet to carry out the most complete theoretical calculation undertaken up to that time. Nicolet's methods strongly influenced the subsequent works.

The current knowledge of the earth's atmosphere is as follows: The absorption of solar ultraviolet radiation in the earth's upper atmosphere causes the neutral gas temperature to increase monotonically with altitudes above a temperature minimum near 80km. This upper region of the atmosphere is called the thermosphere, and as a result of the photoionization produced by solar XUV (extreme ultraviolet) absorption, a plasma, called the ionosphere, pervades the thermosphere. Because heat radiation losses are relatively small above 100km, and because the thermal conductivity of the atmosphere is relatively insensitive to gas density, the temperature gradient at a given height is nearly proportional to the energy absorbed above that height. Thus above

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approximately 20km, where the remaining atmosphere absorbs very little, the neutral particle temperature approaches an asymptotic value between 600K and 1500K depending on solar activity.

Up to 100km the atmosphere is quite well mixed. N_2 and O_2 make up the bulk of the mixture in a ratio to each other near that at ground level. Above 100km molecular diffusion begins to dominate the atmospheric mixing which is generated by internal gravity waves that propagate into this region from below. Molecular diffusion (the diffusion velocity increases inversely with the atmospheric density) tends to allow each species of gas to behave independently, and in particular to have its own scale height. Thus light gases like helium become relatively enriched with increasing altitude above 100km, whereas the opposite happens to heavy gases like argon and nitrogen.

Solar ultraviolet radiation in the thermosphere photodissociates molecular oxygen, and the resulting oxygen atoms can recombine efficiently to reform O_2 molecules only at the heigher pressures below 90km. Dissociation drives an internal convection pattern, with O_2 diffusing upward into the thermosphere where it is dissociated, and with atomic oxygen diffusing downward to where it can recombine. Considerable atomic oxygen resides in the upper atmosphere, and it is the dominant gas above 200km.

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If no vertical transport took place, nearly all the O_2 above 100km would dissociate in about a week. Similarly, without ultraviolet absorption and its associated photodissociation, most of the atomic oxygen would recombine in a few days. Yet curiously the winter polar region, which receives no ultraviolet radiation for months, has its full measure of atomic oxygen. A massive gloval horizontal transport in the upper atmosphere must move gas from the summer to the winter hemisphere. High altitude winds also flow from the dayside to the nightside. The seasonal transport causes other effects in the atmospheric composition. As previously mentioned, the atmosphere is relatively richer in helium at high altitudes, thus high altitude winds blowing from the summer to the winter hemisphere transfer an excess of helium there. The resultant "winter helium bulge" has helium concentrations up to an order of magnitude larger than over the summer pole. These same winds raise the ratio of atomic oxygen to molecular nitrogen in winter.

Hydrogen plays a rather specialized role in the upper atmosphere. The small amounts of water, hydrocarbons, and H_2 in the lower atmosphere are continuously transported upward where they are photochemically dissociated near 100km into atomic hydrogen, which then diffuses upward in the thermosphere at an average rate near 10^8 atoms/cm^{2}sec and ultimately escapes the earth's gravitational field at the

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same rate. The thermosphere stores only a few days' supply of hydrogen. Several transport mechanisms remove or add hydrogen locally at a rate comparable to its escape rate. Firstly, hydrogen moves laterally at great heights from hotter to cooler regions. Secondly, it changes to H^+ by charge exchange with O^+ on the dayside (due to an accidental resonance or equality in ionization potential for O and H), and accumulates in a huge reservoir above the normal ionosphere, then converts back to H at night by the reverse charge exchange process. Thirdly, the more energetic atoms in the thermal distribution can escape the earth's gravitational field directly, and lastly, the H^+ ions formed by charge exchange at high latitudes can flow out into the earth's magnetic tail and be subsequently lost, or perhaps be returned as auroral protons. The actual mix of these effects is poorly known.

Based on these knowledges, more refined atmospheric models are presented [6), 7)]. The prediction of orbital lifetime of spacecrafts, however, needs the future variation of air density. This needs consequently the future variation of solar activities and geomagnetic disturbances. These are by far the largest sources of error, and are very intractable.

Fig. 1 shows the variation of density with height by day and by night at a typical sunspot maximum and minimum. At altitudes near 600km the density varies by a factor of

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20 during the sunspot cycle, with corresponding changes in spacecraft decay rates. At present neither the intensity nor the date of a sunspot maximum can be predicted accurately, so lifetime estimates can only take account of solar-cycle variations in a very approximate way. If a spacecraft is expected to remain in orbit for several solar cycles, a mean density over a solar cycle is used. For a spacecraft which is not expected to remain in orbit for more than a fraction of a solar cycle, a mean density during the remaining lifetime may be used. This is extremely difficult to estimate, particularly at solar minimum, when it is not known how soon the solar activity will begin to increase. It should be emphasized, however, that the average density in the future may differ greatly, particularly if in the next 50 years there is a period of low solar activity akin to the "Maunder minimum" which occured between 1640 and 1710.

The day-to-night variation in air density is not so large as the solar-cycle variation, as Fig. 1 shows. But the maximum daytime density can exceed the minimum night-time density by a factor of up to 5 at 500km altitude, so that a lifetime calculated from the perigee density at the night-time minimum can be in error by a factor of 3 if the perigee later experiences the average density. The day-to-night variation is somewhat intractable, because even the current local time at perigee is not readily available and needs to be calculated; its future variation is not accurately predictable, because it depends on the rate of decay, which itself depends on the variation of local time.

In addition to the wide variations shown in Fig. 1, the air density undergoes a fairly regular semi-annual variation with maxima usually in April and October, and minima usually in January and July. Although the variations are smaller in amplitude than the solar-cycle and diurnal effects-the factor of variation is about 1.6 at altitudes near 200km increasing to about 3 at 500km—the semi-annual variation is more important for lifetime estimation (a) because it extends down to 150km altitude without much diminution, and (b) because the three-month interval between maximum and minimum introduces considerable errors into the most important lifetime estimates, those of between two and six months. The semi-annual effect varies from year to year in amplitude and phase, and cannot yet be predicted accurately. A standard variation is given in modern atmospheric models[6), 7)].

The thermosphere has tides which solar heating drives. Powerful ground-based backscatter radars have measured periodic daily changes in the gas temperature below 130km with very large amplitudes. The daily temperature variation at 115km above Puerto Rico matches annual temperature variation at ground level. Radar data from several locations have shown tidal winds at this same altitude in winter greater

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than 180m/sec, which considerably exceeds the highest surface wind velocities ever recorded. The "weather" at these and higher altitudes, superposed on the tides, takes the form of random-phase internal gravity waves with wind velocity and temperature changes of the same order as those due to the tides. Mass-spectrometers on Atmosphere Explore reveal a "noise level" of nearly ±10% in the concentrations of individual gases over scale sizes greater than a few hundred kilometers. The concentration changes, for heavy and light gases are usually out of phase.

The tidal motions of the upper atmosphere are superposed on an overall atmospheric rotation which appears somewhat faster than that of solid earth. The cause of this superrotation and its dependence on latitude and altitude remain poorly understood[9]. These irregular day-to-day effects must be regarded as largely unpredictable at present, although some of the likely variations may be partially predictable from the 27-day recurrence tendency in solar activity; forecasts of solar activity and geomagnetic disturbances up to three days ahead are made daily by the USAF Geophysical Warning Center. The skilled predictor will take advantage of any available information of this kind, which can be most useful for decay predictions at the very end of a spacecraft's life and for the prediction of re-entry trajectory.

3. Aerodynamic coefficients which affect the accurate prediction of orbital decay and re-entry trajectory

Below altitudes about 100 km where a spaceraft begins to fall to the ground, the estimation of aerodynamic force becomes very important to predict accurately the initiation of falling and re-entry trajectory. When the centrifugal force loses a balance to the gravitational force due to drag force, the spacecraft begins to fall to the ground. Before that time and above that altitude, orbit-type calculations could be applied. After that time and below that altitude, the re-entry trajectory of the vehicle is affected by dominant aerodynamic forces and gravity force. Assuming a spherically symmetric atmosphere and a spherically symmetric earth, descent would occur in a meridian plane in the absence of lateral forces. This confines the problem to one of two dimensions for which polar co-ordinates (r, θ) are convenient. The velocity components of the vehicle are (v, u) respectively as shown in Fig. 2. The equations of motion of a spacecraft are as follows:

$$\left.\begin{aligned} -\frac{dv}{dt} &= g - \frac{u^2}{r} - \frac{L}{m}\cos\phi + \frac{D}{m}\sin\phi \\ \frac{du}{dt} - \frac{uv}{r} &= -\frac{D}{m}\left(\cos\phi + \frac{L}{D}\sin\phi\right) \end{aligned}\right\} \quad (3.1)$$

where

$$L = \frac{1}{2}\rho_\infty V_\infty^2 C_L S \quad ; \text{ lift}$$

$$D = \frac{1}{2}\rho_\infty V_\infty^2 C_D A \quad ; \text{ drag}$$

and

A : reference area of re-entry vehicle,

C_L : lift coefficient,

C_D : drag coefficient,

g : local value of gravitational acceleration,

m : mass of vehicle,

S : reference wetted area,

V_∞: resultant velocity; $\sqrt{U^2+V^2}$,

ρ_∞: density of free stream,

ϕ : flight-path angle relative to local horizontal direction.

The prediction of re-entry trajectory can be given by solving the equations (3.1) with initial conditions. The initial conditions are the time and the position at the initiation of falling. It is already explained in the previous section that the accurate prediction of the initiation can be hardly expected due to uncertainties of predicting future variation of air density. It is also explained in this section that the accurate estimation of aerodynamic coefficients, C_L and C_D, is difficult. Consequently, this causes a larger error in the prediction of re-entry trajectory.

Atmospheric mean free path for altitudes between 80 and 240 km, as given by the US Standard Atmosphere, 1962, is shown in Fig. 3. The estimation of aerodynamic coefficients is a problem of fluid mechanics. Comparing a characteristic

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body dimension with molecular mean free path, modern fluid mechanics is classified into four regimes. At very high altitudes, the atmosphere becomes so rarefied that it no longer behaves like a continuous fluid. The dimensionless parameter called the Knudsen number, K_n, has been introduced to serve as a criterion for determining the relative importance of these rarefaction effects.

$$K_n = \lambda / d \tag{3.2}$$

where λ is the molecular mean free path and d is a relevant characteristic dimension of the flow field. The Knudsen number is expressible in terms of the more familiar moduli of aerodynamics, the Mach and Reynolds numbers, as

$$K_n = 1.25\sqrt{\gamma}\,\frac{M}{Re} \tag{3.3}$$

where γ is the ratio of specific heats. The rarefied gas dynamics deals with flows in which K_n is not negligibly small.

In a flow field where molecular mean free path is smaller than the boundary layer thickness, δ, molecules collide many times with one another in the boundary layer. In this case, flow field in the boundary layer can be approximately dealt with as that of continuum by modifying the wall condition. This is called "slip flow". Here the relevant characteristic dimension of the flow field is the boundary layer thickness, δ , for large Reynolds numbers or the body dimension, d, itself for small Reynolds numbers. Since one has $\delta/d \sim 1/\sqrt{Re}$, at least two different Knudsen numbers, λ/δ , or λ/d , are appropriate for characterizing the slip flow regime, and

$$Kn_\delta \sim \frac{M}{\sqrt{Re}} \tag{3.4}$$

In a flow field where Kn is very large, the basic phenomena and theoretical approaches for this flow are significantly different from those for continuum flows. This is called "free molecular flow". The flow regime intermediate between slip and free malecular flow is known as the "transition flow regime". It corresponds to densities for which the mean free path has the same general order of magnitude as the characteristc dimension of the flow field, where collisions among molecules and interaction of incident molecules with a surface become samely dominant.

Tsien[10)] has classified approximately the flow regimes as follows:

continuum flow : $M/\sqrt{Re} < 0.01$

slip flow : $0.01 < M/\sqrt{Re} < 0.1$

transition flow : $0.1 < M/\sqrt{Re}$, $M/Re < 10$

free malecular flow : $10 < M/Re$

These flow regimes, together with the equivalent altitudes corresponding to a characteristic dimension of one meter are indicated in Fig. 4. The aerodynamics which governs the motion of a spacecraft at the very end of the orbital lifetime and that along the early trajectory of re-entry is in the flow regimes of free molecular flow and transition flow.

In free molecular flow the interaction of the molecular flux with a surface is formulated on the basis of coefficients. The coefficients relate the extent to which the properties of the incident flow are accomodated to the conditions of the surface, as well as the nature of the re-emission flow pattern. These coefficients are the surface reflection coefficients for tangential and normal momentum (σ, σ'), and the thermal accomodation coeffienct (α):

$$\sigma = \frac{\tau_i - \tau_r}{\tau_i} , \quad \sigma' = \frac{P_i - P_r}{P_i - P_w} , \quad \text{and} \quad \alpha = \frac{dE_i - dE_r}{dE_i - dE_w}$$

where τ is the tangential momentum, P is the normal momentum, dE is the energy flux. Subscripts i : incident on the surface, r: re-remitted from the surface, w: re-emitted if all incident molecules were re-emitted in Maxwellian equilibrium with the surface. When these coefficients are zero, there is no energy or tangential momentum exchange between the incident stream and the surface, and the reflection is said to be specular. When these coefficients have the value of unity ($\alpha = \sigma = \sigma' = 1.0$) the molecules are re-emitted randomly, and in complete thermodynamic equilibrium with the surface; where upon the reflection process is said to be completely diffuse. Most of the actual interaction is intermediate between them, which is said to be partially diffuse, as explained in Fig. 5. The coefficients depend on the direction and the intensity of incident flux, and the properties of a surface material. Aerodynamic coefficients, C_L, and C_D are closely related to these reflection coefficients. Unfortunately, the reflective properties of a spacecraft's surfaces vary in its long lifetime due to the influence of space environment. The uncertainty of C_L and C_D in free molecular flow primarily comes from that of the properties.

In transition flow regime, theoretical treatments are presently limited only near free molecular and near slip flows and do not cover the whole transition regime. Experimental results are rather limited in this flow regime. Even for the simplest shape such as sphere, an error of experimental and theoretical results is hardly better than ± 10% as shown in Fig. 6[11]. It can be easily understood that for a spacecraft more complicated in shape an accuracy of the estimation of aerodynamic coefficients is worse than that for a sphere in the transition flow regime. The accurate prediction of the re-entry trajectory is therefore very difficult.

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4. Evaluation of existing methods for orbit determination and prediction

It is already discussed in the previous sections that the accurate prediction of a spacecraft decay date and re-entry trajectory some months or years ahead is intractable because the prediction of future variation of air density and the accurate estimation of aerodynamic forces in transition flow regime are difficult. There is a possibility, however, for improvement in accuracies of orbit determination (OD) and orbit prediction (OP) for short periods by improving spacecraft tracking technology and also by estimating hourly the parameters such as ballistic coefficient, air density, etc, which have large uncertainties in themselves and are varying with time. This consequently needs a real time process of orbital data measured by spacecraft tracking and a larger coverage of tracking.

The OD process is to calculate state vectors (position $\mathbf{r}$, velocity $\dot{\mathbf{r}}$) and other parameters such as a ballistic coefficient, air density, etc, of a spacecraft at an epoch from observation data by tracking. The process of OP is to solve the equations of motion with an initial condition of the state vectors of the vehicle which are calculated by OD at the epoch, and to predict a future orbit. In the process of OP, equations of motion are required to be integrated numerically instead of processing observation data. A set of observables contains range, range-rate, azimuth, and elevation. Tracking consists of observing the spacecraft by a variety of possible means. Radar, radio-frequency for a co-operative target and laser are available, and a high precision optical camera tracking gives the most accurate information of angles within the accuracy of better than 2 arc seconds. Tracking is performed from the earth. On the earth both land-based and marine stations are used. In the near future, it becomes feasible to track spacecrafts in the lower orbits from another satellite in the higher orbit with a larger coverage. TDRSS of NASA and NAVSTAR/GPS of USAF may be used for that purpose.

A batch (least-squares) process and a sequential process are typical of OD methods. A batch process has been used at most of tracking stations to support the activities of tracking network by presenting an early prediction of spacecraft's orbit, because the algorithm is simple and because the process is stable and reliable for ordinary operations. But there are some shortcomings in the process, these are: (1) it needs an accurate mathematical model to describe orbital motion of a spacecraft, (2) it cannot deal with process noise, and (3) it is not applicable to requirement of real time process. Actually, there exist errors in our knowledge of natural forces. The best accuracy to be expected by this method therefore may be within ±30m, and that can be obtained only by processing the tracking data in a short arc of the orbit. If a larger span of tracking data for long time is processed by this method, the accuracy becomes degraded due to cumulative errors of model dynamics.

A sequential process improves the above mentioned shortcomings of a batch process. Fig. 7 shows the processes of OD/OP by both methods. Applications of a sequential process have mainly been made to on-board systems which require a real-time process. Some have been applied to a specialized purpose of target tracking such as launching and re-entry in which a real time processing is a stringent requirement. The problem on an actual application has been that of a divergence of filters, which has prevented practical applications to be made to spacecrafts. Many techniques to control the divergence have been studied. Factors to drive divergences of a filter are as follows; (1) errors in a model dynamics, (2) errors due to linearization and (3) errors in computations. The divergence due to the factor (2) can be improved by applying an extended Kalman filter. The control of divergence due to the factor (3) depends on capability of a computing machine. The process on land-based stations therefore has no problems. The process of on-board type depends on development of computing machine. The factor (1) is most important for the accuracy as in a batch process.

There have been developed two kinds of methods to compensate a divergence due to errors in a model dynamics, i.e. adaptive and nonadaptive methods. Adaptive methods compensate the divergence by estimating statistics of process noises along with the spacecraft's state. But the algorithms are rather complicated, while no discernible improvement in accuracies is expected compared with those of nonadaptive methods. Adaptive methods are well explained by Jazwinski[12]. Introducing nonadaptive methods with simpler algorithms of Tapley et al.[13] in this section, problems of an improvement in accuracies of OD/OP are discussed.

A equation of motion of a spacecraft is given as

$$\left.\begin{aligned} \ddot{\mathbb{r}} &= \mathbb{A}_{\text{model}} + (\mathbb{A} - \mathbb{A}_{\text{model}}) \\ \mathbb{W} &\equiv \mathbb{A} - \mathbb{A}_{\text{model}} \end{aligned}\right\} \quad (5.1)$$

where $\mathbb{A}$ is the real acceleration vector, $\mathbb{A}_{\text{model}}$ is the modeled acceleration vector, and $\mathbb{W}$ is the error vector of model dynamics. As stated previously, there exists an error of model dynamics $\mathbb{W}$ due to poor knowledge of natural forces and due to simplification of algorithms. In a batch process, $\mathbb{W}$ is neglected by assuming as $|\mathbb{A}_{\text{model}}| \gg |\mathbb{W}|$. SNC (State Noise

Compensation) method, one of nonadaptive methods, compensates an error due to model dynamics by dealing with W as Gaussian white noise, while DMC (Dynamic Model Compensation) method, another of nonadaptive methods, estimates the error itself as one of unknown parameters by approximating W as a first order Markov process (Ornstein-Uhlenbeck, see Fig. 8). Using these methods, a sequencial process presents a real time OD/OP with no worse accuracies compared with those of a batch process.

An OD/OP for re-entry trajectory of a spacecraft with NPS requires a real time mode and a higher accuracy. Dynamics of a spacecraft at the end of decay and during re-entry gives very complicated equations of motion as stated in the provious sections. Many parameters are contained such as aerodynamic coefficients of C_L and C_D, shape of the entering body, attitude angles, air density, Mach numbers, Reynolds numbers, etc., which vary also as functions of other parameters. A sequential process for OD/OP of this case is therefore very promising to apply. Some studies have been successfully tried to deal with drag parameter as process of Brownian motion, i.e. $\dot{\chi} = \nu$ ($\nu \equiv$ Gaussian white noise), where $\chi = C_D A/m$ is referred to as a reciprocal of the ballistic coefficient[14]. If experimental data of wind tunnel tests for C_D are taken into account for fitting or a DMC method is applied, there will be some possibilities of an improvement in OD/OP of re-entry trajectories.

5. On the so-called "burn-up" on re-entry

During entry into the earth's atmosphere, a spacecraft is susceptible of severe decelerations and intense aerodynamic heating. This is called "burn-up" on re-entry. A space nuclear reactor has a thick walled reflector covering the core of a reactor. Remarking the reflector, the "burn-up" is discussed in this section.

An approximate analytical solution to the motion equations by Chapman[14)] is applied to the present problem. Three assumptions made at the outset are:

i) Atmosphere and the earth are spherically symmetric.

ii) A locally exponential atmosphere.

iii) Peripheral velocity of the earth is negligible compared to the velocity of the entering vehicle.

In addition to these three physical assumptions, two mathematical approximations are made in order to effect major simplifications in the structure of the equations of motion as (a) $|dr/r| \ll |du/u|$, (b) $|(L/D)\tan\phi| \ll 1$ (see Fig.2). The limitations resulting from approximations (a) and (b) are examined by Chapman himself. It is shown that for vehicles entering from decaying spacecrafts orbits, with or without positive lift, the errors introduced are only the order of a few perccnt insofar as aerodynamic heating and peak decelerations are concerned. Surprisingly small errors result from approximation (b), even for very large L/D ratios, because, in orbital decay or in a smooth glide, the larger the L/D the smaller the angle ϕ at conditions near maximum heating and peak deceleration; this keeps the product $|(L/D)\tan\phi|$ small.

Descent in a spherically symmetric atmosphere about a spherically symmetric planet would occur in a meridian plane in the absence of lateral forces. Hence, two component equations of motion are given in the equations (3.1).

Letting

$$Z = \frac{\rho_\infty}{2\left(\frac{m}{C_D A}\right)}\sqrt{\frac{r}{\beta}}\,\bar{u}, \quad \bar{u} = \frac{u}{u_c} \equiv \frac{u}{\sqrt{gr}}, \text{ and}$$

$$\rho_\infty = \bar{\rho}_0 e^{-\beta y}, \quad \sqrt{\beta r} \cong 30 \text{ for the earth's atmosphere,}$$

$$\left.\begin{aligned} &\frac{dZ}{d\bar{u}} - \frac{Z}{\bar{u}} = \sqrt{\beta r}\,\sin\phi \\ &\bar{u}\frac{d}{d\bar{u}}\left(\frac{dZ}{d\bar{u}} - \frac{Z}{\bar{u}}\right) - \frac{1-\bar{u}^2}{\bar{u}Z}\cos^4\phi + \sqrt{\beta r}\,\frac{L}{D}\cos^3\phi = 0 \end{aligned}\right\} \quad (5.1)$$

The free-stream Reynolds number per unit length can be expressed by Z as

$$\frac{Re_\infty}{l} \equiv \frac{V_\infty \rho_\infty}{\mu_\infty} = \frac{2\sqrt{g\beta}}{\mu_\infty \cos\phi}\left(\frac{m}{C_D A}\right) Z \qquad (5.2)$$

where y is the height and μ_∞ is the coefficient of viscosity of the free-stream. The Reynolds numbers involved during entry from a decaying orbit are relatively small. Near peak heating, for example, the value of Z ranges from 0.17 to 0.015, for L/D ratios between 0 and 1, so the corresponding Reynolds numbers are of the order of $(10^4 \sim 10^3)$B per meter, here $B = m/C_D A(Kg/cm^2)$; ballicstic coefficient.

These are sufficiently small for one to be optimistic about the practical possibilities of maintaining laminar flow for shallow entry from a spacecraft orbit.

Fairly simple expressions also can be obtained for the aerodynamic heating rate per unit area, q , and the total heat absorbed per unit area, Q/S. Following the analysis of Lees[16], the heating rate at any point on a body can be considered to be a certain fraction $K_1 = q/q_s$ of the heating rate q_s at a stagnation point of radius of curvature R. The heating rate in hypersonic flow at a stagnation point, can be expressed as

$$q_s = \frac{C}{\sqrt{R}}\left(\frac{\rho_\infty}{\rho_0}\right)^n\left(\frac{\bar{u}}{\cos\phi}\right)^m \text{ Kcal cm}^{-2}\text{ sec}^{-1}$$

where the constants C, n, and m depend on the type of boundary-layer flow, and $C \sim \sqrt{\rho_0 \mu_0}\ u_c^3\ P_r^{-\frac{2}{3}}\left[(1-\gamma)/\gamma\right]^{\frac{1}{4}}$ for hypersonic flow but C=25.5 Kcal $\text{cm}^{-\frac{3}{2}}\ \text{sec}^{-1}$ is adjusted here to match for air at velocities near peak heating ($\bar{u} \cong 0.8$). For laminar flow $n = \frac{1}{2}$ and $m = 3$ are given (with ρ_0 being the true sea-level density).

The laminar convective heat-transfer rate through the whole wetted surface, S, can be written in terms of the Z function as

$$Q(t) = 7.05\, k_2 S \sqrt{B/R_{eff}}\ \frac{\sqrt{\bar{u}^5}\, Z}{\cos\phi}\ \frac{\Delta H}{H_\infty}\ (\text{Kcal/sec}) \quad (5.3)$$

where $k_2 = \frac{1}{S}\int k_1\, dS = \frac{1}{S}\int \frac{q}{q_s}\, dS$ is the factor which takes into account the variations in heat flux over the whole surface S(cm^2) wetted by the boundary layer (for a hemisphere, for example, $k_2 \doteqdot 0.5$). R_{eff} (cm) is an effective radius of curvature at a stagnation point to that of a hemisphere. $\Delta H = H_\infty - H_w$, H_∞ and H_w are the total entalpies of the free stream and at the wall, respectively.

If the local wall temperature of the entry vehicle, T_w, is assumed to be much smaller than the local recovery temperature, T_r, of the outer stream, the total heat absorbed during entry is

$$Q = 2.9 \times 10^2 k_2 S \sqrt{B/R_{eff}} \quad \text{Kcal} \qquad (5.4)$$

The total heat Qm to burn up a body of mass weight W(Kg) into a melted state is

$$Q_m = W\,(C_b \Delta T_m + C_m) \quad \text{Kcal} \qquad (5.5)$$

where, Cb is the specific heat (Kcal/Kg°C), Cm is the latent heat of melting (Kcal/Kg), and ΔTm =(melting temperature of a material, Tm) -(initial temperature, Ti). A simple estimation of melting characteristics of an entry body can be given by the equations (5.4) and (5.5). Assuming that radiation cooling, Tw compared to Tr, variation of R_{eff} due to a partial melting, effects of mass diminution by melting on the trajectory are neglected, the mass of a re-entry body which burns up and is dispersed in the atmosphere during re-entry is given as

$$W \leq 2.9 \times 10^2 k_2 S \sqrt{B/R_{eff}} \,/\, [C_b \Delta T_m + C_m] \quad \text{Kg} \qquad (5.6)$$

Materials of lower melting temperatures can be well approximated by the equation (5.6), however, those of higher melting temperatures ($Tm > 1000°C$) must be taken into account an effect of radiation cooling. Effects of radiation cooling and mass diminution are taken into account as follows:

$$\tilde{T}w(t) = Tw(t_i) + \int_{t_i}^{t} \left[\dot{Q}(t) - \epsilon\sigma Tw^4 S_{rad} \right] dt \Big/ (W \cdot C_b) \quad (5.7)$$

where t_i is a starting time, is the total emissivity, σ is Stefan Boltzmann's constant, and S_{rad} is the radiating surface area. When $\tilde{T}w < Tm$, there occurs no melting on the entry body. When $\tilde{T}w > Tm$, the entry body begins to melt at the temperature Tm after the time of $\tilde{T}w(t) = Tm$. The rate of melting is given as

$$-\frac{dW}{dt} = \left[\dot{Q}(t) - \epsilon\sigma Tw^4 S_{rad} \right] \Big/ C_m \qquad (5.8)$$

The equations (5.3), (5.7), and (5.8) are numerically integrated with computing machines, considering the wall temperature condition.

Smaller entry bodies will burn up into a melted state and is dispersed in the atmosphere, while larger ones will not be melted and subsequently crash against the earth's surface. The "burn-up" of reflectors of space nuclear reactors are estimated by the above equations. Reflectors are usually made of Al, Al_2O_3, Be, BeO, graphite, etc. A typical configuration of reflectors is cylindrical and the wall is thick. The sizes of reflectors depend on core volumes of reflectors, i.e., on thermal outputs of reactors. Fig. 9 shows a tendency of core volumes of space nuclear reactors versus their thermal outputs. KIWI-NERVA series are nuclear rocket propulsion systems tested in 1960's for NASA-advanced programs and have been ceased for development since early 1970's. From this diagram, internal sizes of a reflector can be estimated. Thicknesses of a reflector will be determined by system design. Fig. 10 shows a relation between thicknesses of a reflector and mass of U^{235} at critical for a reactor of about mega watts thermal output. The thickness of a reflector is usually designed in a range of 5～50cm. Considering these design conditions of a space nuclear reactor, melting characteristics of Be, and Al reflectors are estimated by using the previous equations of aerodynamic heating. Beryllium is used for the reflectors of USA SNAP 10A, USSR Рома ⅢKд and Топаз[17]. The relevant physical constants of the materials are listed in Table 1.

A configuration and an attitude of the reflectors during re-entry are supposed as shown in Fig. 11, where ℓ and d are length and diameter of the core, respectively, t is thickness of the wall, and $\ell/d = 5/4$ is supposed for calculations. Fig. 12 shows the results. Critical thicknesses of "burn-up and dispersion" are shown for both of ℓ and Be reflectors with respect to the corresponding thermal outputs. The solid curves of $T_{core}/T_{ref} = 0$ assume that cores are heat shielded from reflectors and the dotted curves of $T_{core}/T_{ref} = 0.5$ assumes that aerodynamic heats are transferred to the core until to the corresponding temperatures. The reflectors of the upper areas do not "burn up" and is not dispersed but crash against the earth's surface. It can be supposed that the non-shielded cores are more practical than the shielded ones. It may be said therefore that almost all of the existing space nuclear reactors do not "burn up" and is not dispersed during re-entry if a special device is not attached to the space nuclear reactors in order to separate nuclear fuels from reflectors in the early phase of re-entry.

6. An impact of attaching reboost system or re-entry control system on spacecraft design

As it is discussed in the previous section, most of space nuclear reactors may not burn up and may not be dispersed in the atmosphere during re-entry if a special device is not attached to the reactors. Even if a special device is attached, reliability of such devices becomes important. That is to say, there is always a fear for human beings such that the launched space nuclear reactors would some day crash against the earth's surface. Space nuclear power systems therefore must be reboosted into a long lifetime orbit or let be fallen on a pre-determined area of the earth's surface for safe recovery after separation from or together with the spacecrafts. Mass increment required to attach such systems is discussed in this section.

For reboosting into a long lifetime orbit, an ascent trajectory via Hohmann transfer ellipse is the most efficient type. A short-powered phase is required at the apogee of the ellipse, to provide the required orbital velocity. The first injection, which is into a transfer ellipse, is followed by a long coasting period to the apogee, where the desired apogee injection conditions are obtained from a kick impulse. Control of the vehicle must be maintained during coast and the short-burning period at apogee. Fig. 13 explains the simplest maneuver of a coplanar reboosting via Hohmann transfer. Where ΔV_1 and ΔV_2 are velocity increments at perigee and at apogee, respectively, and r_i and r_t are radii of initial and terminal orbits, respectively, and μ is gravitational constant of the earth. Velocity increments are given as,

$$\left.\begin{aligned} \Delta V_1 &= \sqrt{\frac{\mu}{r_i}}\left(\sqrt{\frac{2r_t}{r_i + r_t}} - 1\right) \\ \Delta V_2 &= \sqrt{\frac{\mu}{r_t}}\left(1 - \sqrt{\frac{2r_i}{r_i + r_t}}\right) \end{aligned}\right\} \quad (6.1)$$

In order to estimate an effect of attaching a reboost propulsion system on spacecraft design, a simple case of re-startable single engine system is considered. Mass ratio is given for that case as,

$$\frac{m_e + m_p}{m_e} = \frac{\Lambda \exp\left(\frac{\Delta V_1 + \Delta V_2}{I_{sp}\, g_0}\right)}{1 - (1-\Lambda)\exp\left(\frac{\Delta V_1 + \Delta V_2}{I_{sp}\, g_0}\right)} \qquad (6.2)$$

where m_e is the mass of a spacecraft without reboosting system, m_p is the mass of the attached reboost propulsion system, Λ is the fuel looding factor, I_{sp} is the specific thrust, and g_0 is the gravitational acceleration on sea level.

Required mass increment for reboosting is shown in Fig. 14. An assurption is considered there such that spacecrafts of which missions are over at a 200km altitude circular orbit are reboosted into a 1000-year life orbit. Life of a satellite depends on the orbit and the mass/area ratio. According to the diagram of King-Hele[8], 1000-year life circular orbits together with mass/area ratios are given in Fig. 14. $\Lambda = 0.86$, $I_{sp} = 290$ secs, and $g_0 = 9.8$m/sec are used in the calculations. In Fig. 14, if a 200km altitude circular orbit spacecraft with mass/area ratio of 100kg/m^2 is reboosted into a 900km altitude circular orbit according to the arrow, its life will be about 1000 years and the required propulsion system is about 17% increase in mass to the spacecraft without reboosting system. The values of mass/area ratios of most usual satellites are between 50 and 200kg/m^2.

Another approach to dispose of NPS, of which space missions are over, is to recover safely on the earth. It needs a re-entry navigation, guidance, and control system, because non-controlled spacecrafts take random re-entry trajectories, and because no one predicts exactly the place where the spacecrafts or NPSs crash against. In order to estimate an increment in mass required to attach a re-entry control system, simple orbital maneuvers are considered. Fig. 15 shows two types of simple re-entry maneuvers, both are co-planar Hohmann transfers. Fig. 15(a) shows a re-entry via a parking orbit, and (b) shows a direct one.

In a parking orbit maneuver, a 200km altitude circular orbit is considered as that of parking and a final kick is calculated by Hohmann ellipse with perigee height of 90km. Then, we have

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$$\left.\begin{aligned}\Delta V_1 &= \sqrt{\frac{\mu}{r_i}}\left(1-\sqrt{\frac{2r_p}{r_i+r_p}}\right)\\ \Delta V_2 &= \sqrt{\frac{\mu}{r_p}}\left(\sqrt{\frac{2r_i}{r_i+r_p}}-1\right)\\ \Delta V_3 &= 0.39\ \mathrm{Km/sec.}\end{aligned}\right\}\quad (6.3)$$

If re-entries are discussed from a circular orbit with lower than 2000km altitude which has about 1000 (years m^2/kg) lifetimes from King-Hele; that is about 10^5 years life for a satellite with mass/area ratio of $100kg/m^2$, a difference in mass ratios between system mass of multi stages maneuver and that of re-startable one is very small. Then we have the same equation of (6.2) for the approximate estimation of mass ratio by using velocity increment of $\sum \Delta V_i$ instead.

A direct re-entry maneuver is estimated by taking perigee height of 90km via Hohmann elipse from a mission orbit. Then, we have

$$\Delta V = \sqrt{\frac{\mu}{r_i}}\left(1-\sqrt{\frac{2r_{g0}}{r_i+r_{g0}}}\right) \qquad (6.4)$$

The same values of Λ, I_{sp}, g_0 are used for estimation of re-entry control maneuvers as those of reboosting. Fig. 16 shows required system mass increments for re-entry control maneuvers. A navigation, guidance, attitude control system and a thermal protection system are required for safe recovery, in addition to a re-entry propulsion system. The required mass for proof of shock

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structure and thermal protection system is difficult to estimate generally, but, may be assumed here to be by under 20% increments. The required mass for a navigation, guidance, and attitude control system may be about 100kg by the current technology, which does not depend on sizes of spacecrafts and will be improved in weight by an advancement. It must be noted here that even if a re-entry navigation and guidance system is on a re-entry spacecraft, the dispersion of footprint by direct re-entry trajectory is generally worse than that of a parking orbit method. Controllability of navigation and guidance system must be carefully investigated for a direct re-entry.

7. Conclusions

It requires the accurate prediction of future variation of the earth's atmosphere density and the accurate estimation of aerodynamic forces to predict accurately a spacecraft's orbital lifetime and re-entry trajectory some months or years ahead by solving the equations of motion with a given initial condition. The earth's atmosphere has been investigated since the first artificial satellite was launched. The difficulty of orbital life prediction is explained by introducing the studies of the earth's atmosphere, which is relevant to orbital life. It is also explained that there exist difficulties for an accurate estimation of aerodynamic forces which govern the motion of a spacecraft at the end of lifetime and during re-entry.

There is a possibility of improving accuracies in a short orbital life prediction by spacecraft tracking techniques and by real-time processing of measured data. A DMC (Dynamic Model Compensation) method, which is one of the modern sequential processes, is promising for the prediction of re-entry trajectories, because it executes real-time OD/OP processes by compensating the model dynamics errors caused by uncertainties of aerodynamic forces one after another. A great improvement for OD/OP by target tracking is achieved by world-wide cooperations in the following fields:

(1) Accurate calibration of positions of tracking stations through the world by a standard world geodetic system

(2) An international reference for world geodetic system for the above purpose

(3) International coordination of the study of satellite geodesy which has been carried out independently in each country

(4) An international cooperation through tracking stations of the world to make easy exchange of information on orbital observables of satellites possible.

Remarking reflectors of space nuclear reactors, the problems of so-called "burn-up and dispersion in the atmosphere" by aerodynamic heating during re-entry are discussed under an assumption. A calculation shows that almost all of the existing space nuclear reactors may not burn up and be dispersed in the atmosphere during re-entry except a reactor of small thermal output, unless a special device is attached to the reactors. To enable the fuels to burn up and to be dispersed in the atmosphere, space nuclear reactors must be designed at least to separate fuels from reflectors in the early phase of re-entry. To burn up and to be dispersed in the atmosphere must be assessed in another time by considering an effect of the accumulated radio active materials on living things of the earth. Reliability of a device for "burn-up" must also be discussed thoroughly.

It is not inevitable to disperse radio active materials in the atmosphere, or to let a space NPS fall randomly onto the earth's surface. It is one of the simplest tries to launch spacecrafts with NPSs into higher altitude orbits of which lifetimes are longer than those of the radio activities. In case it is inevitable that a NPS is used in a lower altitude orbit, there are two ways to dispose of the NPS after the mission is over. One is to reboost it into a higher altitude orbit of which lifetime is longer than that of the radio activities. The other is to recover it on the earth safely by re-entry control. The mass increase by attaching reboosting system is under 20%, while that of re-entry control system is between 20 and 80%. The latter way will be more desirable though the mass increase is larger than that of the former way, because an increase of numbers of satellites in orbits will cause another problem in future. In the future, re-entry control system will be attached to a spacecraft with NPS after launching by a new space transportation system such as US Space Shuttle Orbiter. There will be also feasibility to recover the launched spacecraft with

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NPS directly on-board by such a vehicle with Manipulator System or Teleoperator System.

In the near future, a new system such as US TDRSS or USAF NAVSTAR/GPS will be available to extend tracking network capability. An autonomous system on a spacecraft together with the new tracking systems, will contribute greatly to increasing accuracies of OD/OP by target tracking, as well as to reboosting, recovering and increasing reliability of disposal maneuvers by a new space transportation system such as US Shuttle Orbiter.

8. References

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Table 1. Physical constants of Be and Al

Material	density (kg/cm^3)	specific heat (kcal/kg°c)	latent heat of melting (kcal/kg)	melting temp. (°c)	total emissivity
Be	1.84×10^{-3}	0.426	50.0	1278	0.5
Al	2.7×10^{-3}	0.282	94.6	660	0.2

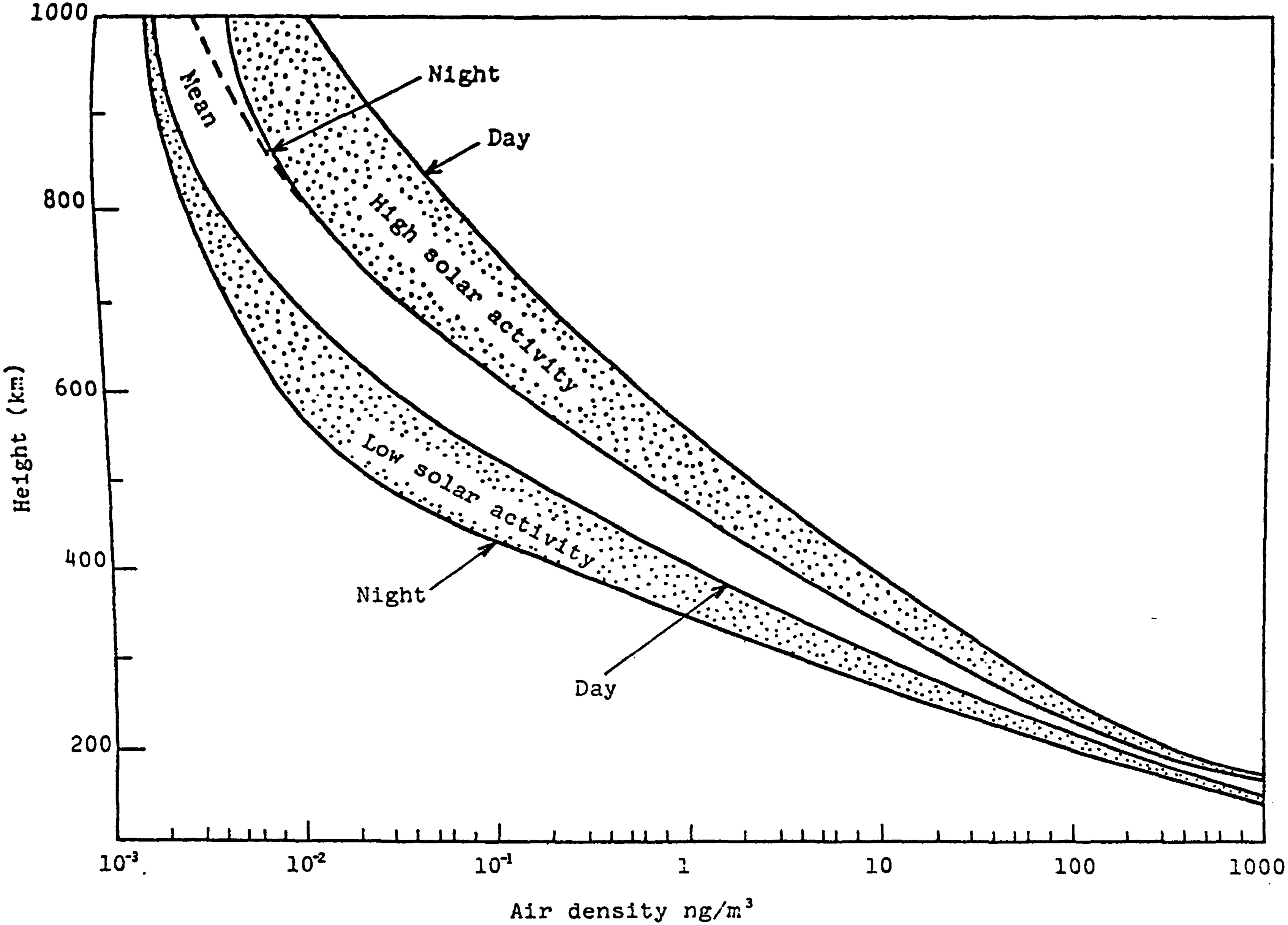


Fig.1 Variation of density with height for low and high solar activity, and the mean over a solar cycle. (Ref.8)

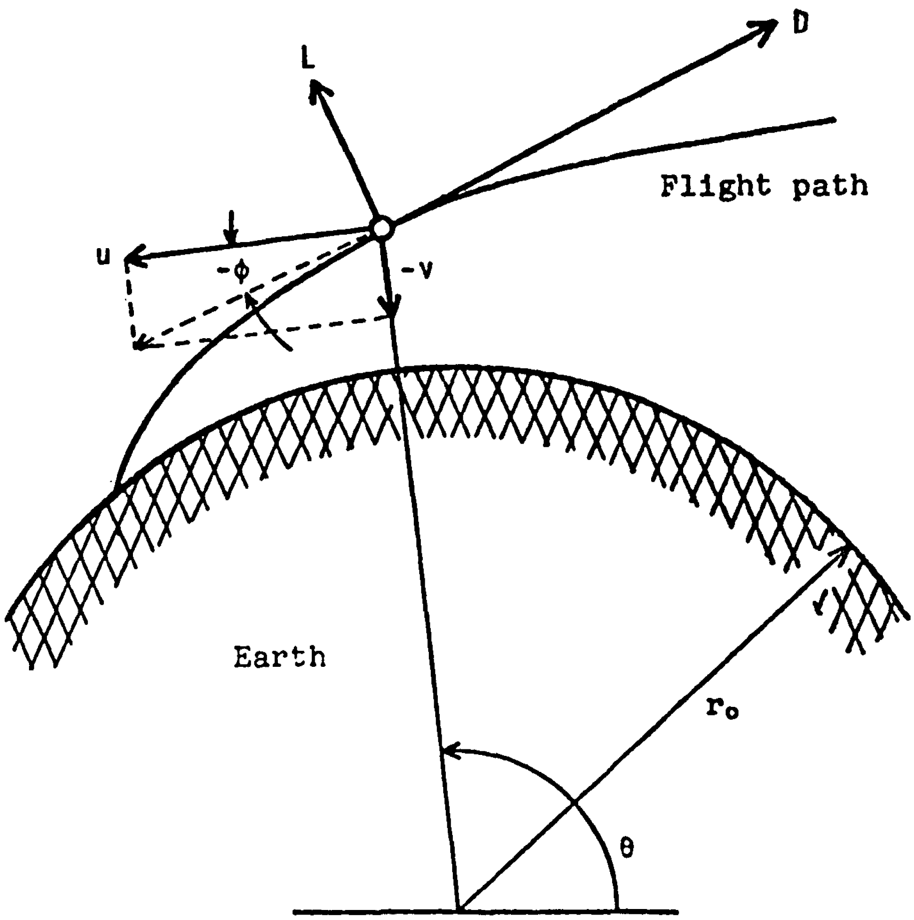


Fig. 2 Re-entry kinematics diagram

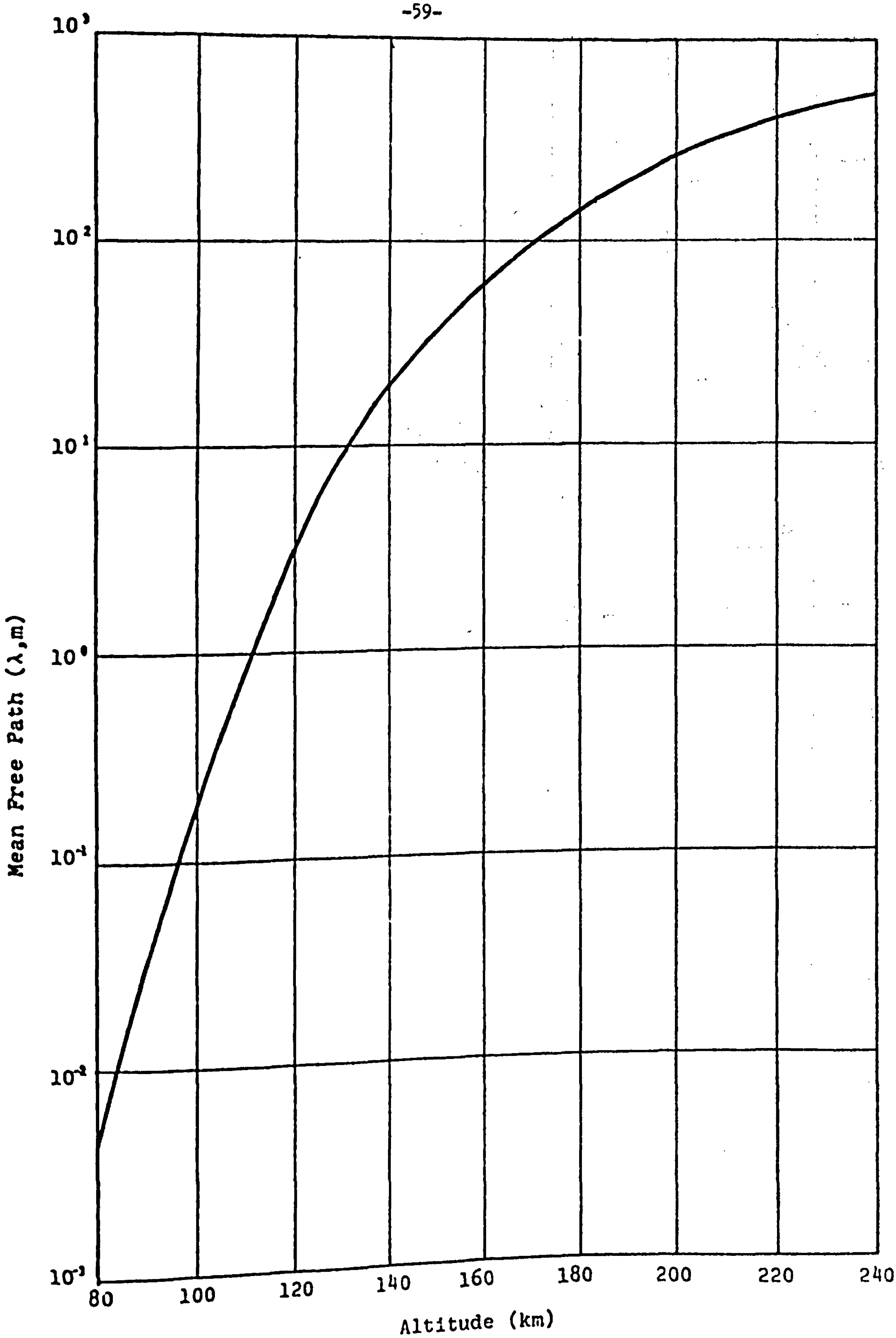


Fig.3 Atmospheric mean free path

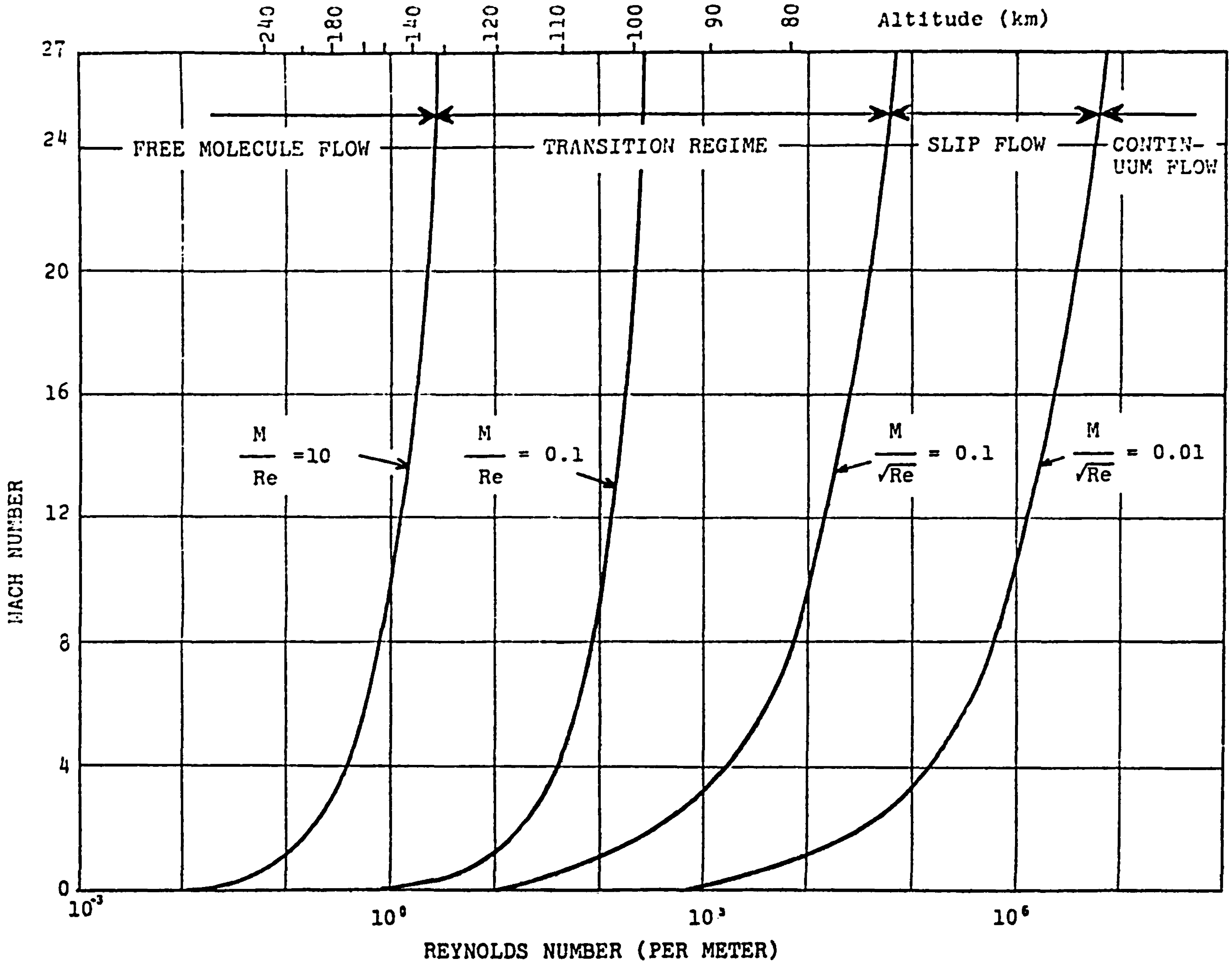
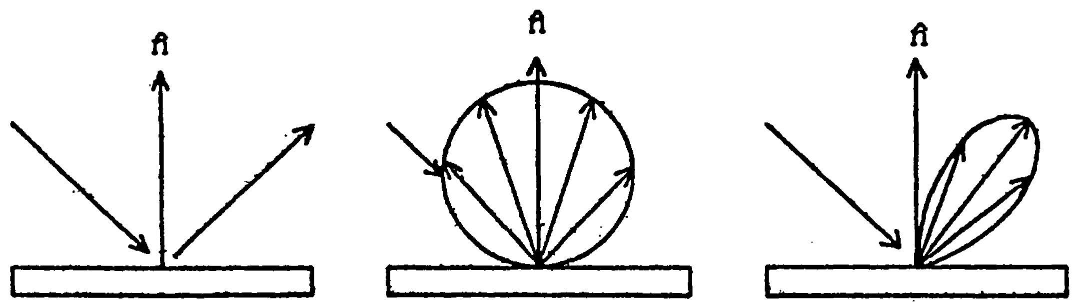


Fig. 4 Re-entry Fluid Mechanics Regimes



(a) Specular (b) Diffuse (c) Partially diffuse

Fig. 5 Reflection geometry

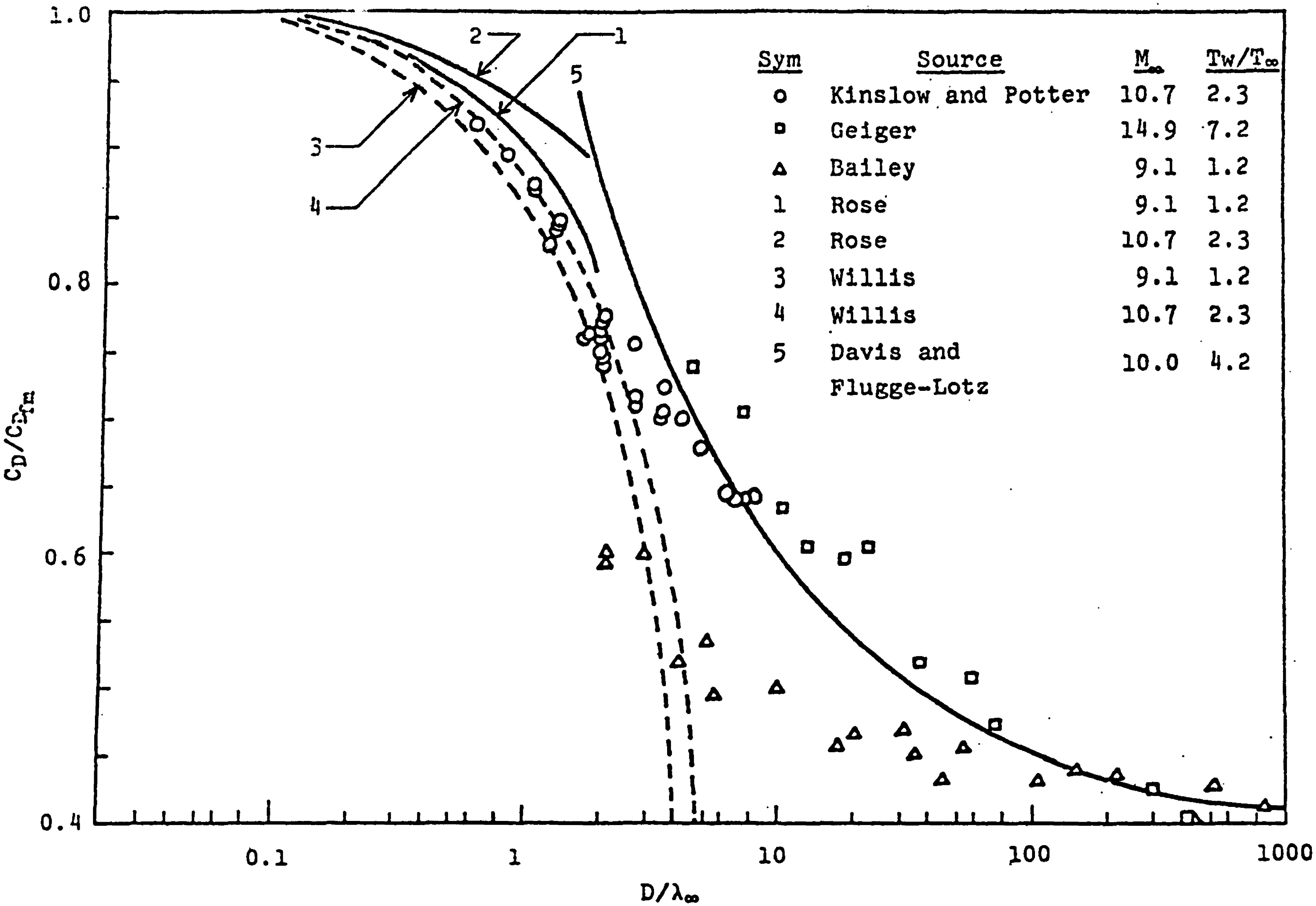


Fig.6 Comparison of Some Experimental and Theoretical Results for Spheres (Ref.11)

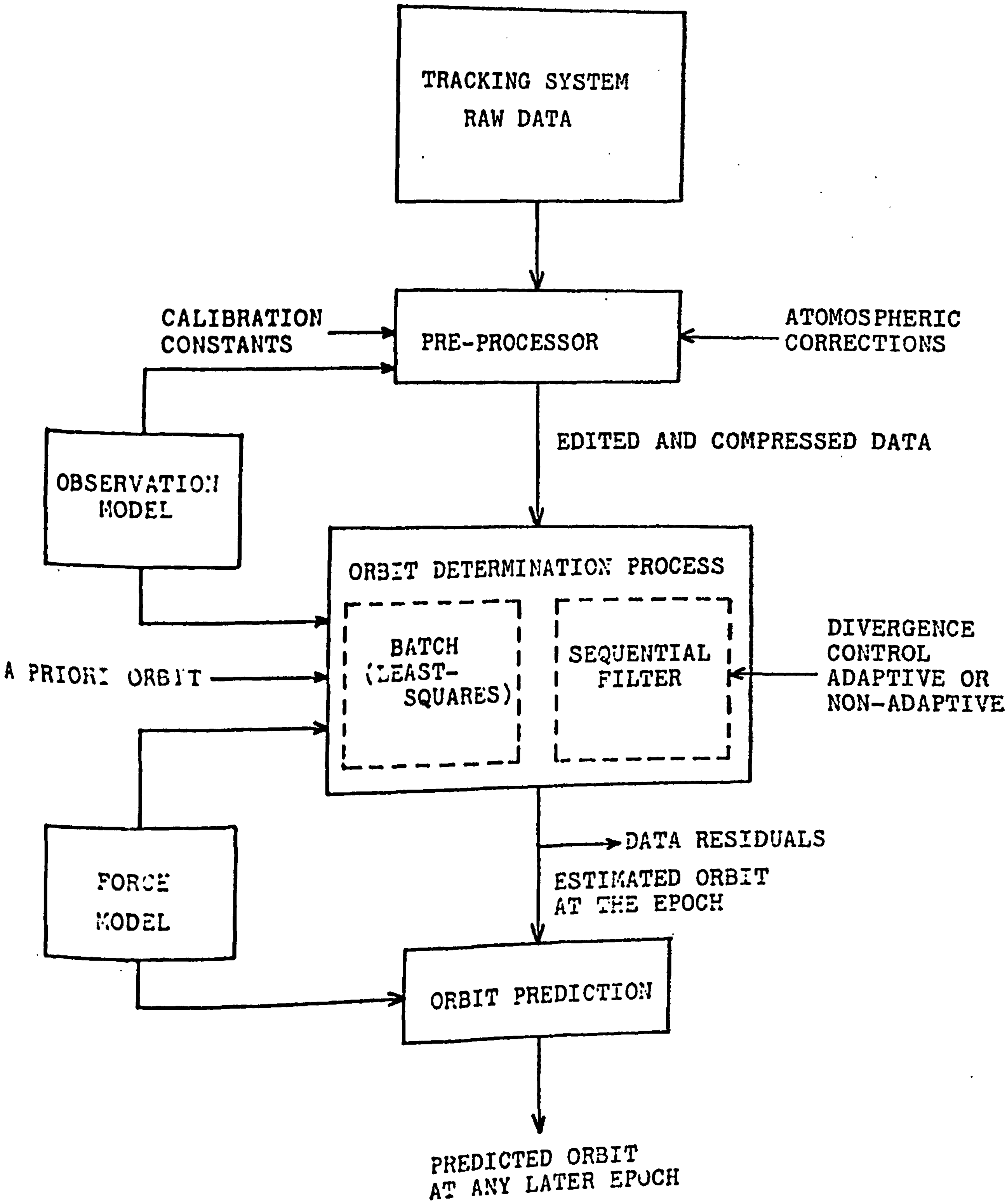


Fig. 7 Orbit Determination and Prediction Processes

R E A L S Y S T E M	S Y S T E M M O D E L
$\ddot{r} = \underbrace{A}_{\text{real acceleration, never known precisely}}$	$\ddot{r} = \underbrace{A_{model}}_{\text{modeled acceleration}} + \underbrace{(A - A_{model})}_{\substack{W \\ \text{unmodeled or mis-} \\ \text{modeled acceleration}}}$ $\ddot{r} = A_{model} + W$ $\dot{W} = -[\alpha]W + U_w$ $\dot{\alpha} = U_\alpha$ where $[\alpha]$ is a 3-by-3 diagonal matrix with $[\alpha]_{ij} = \alpha_i \delta_{ij}$ and U_w and U_α are white noise processes. Note: In the SNC mode, W is assumed to be white noise process.

Fig. 8 Real and Model Systems in the Dynamic Model Compensation Method

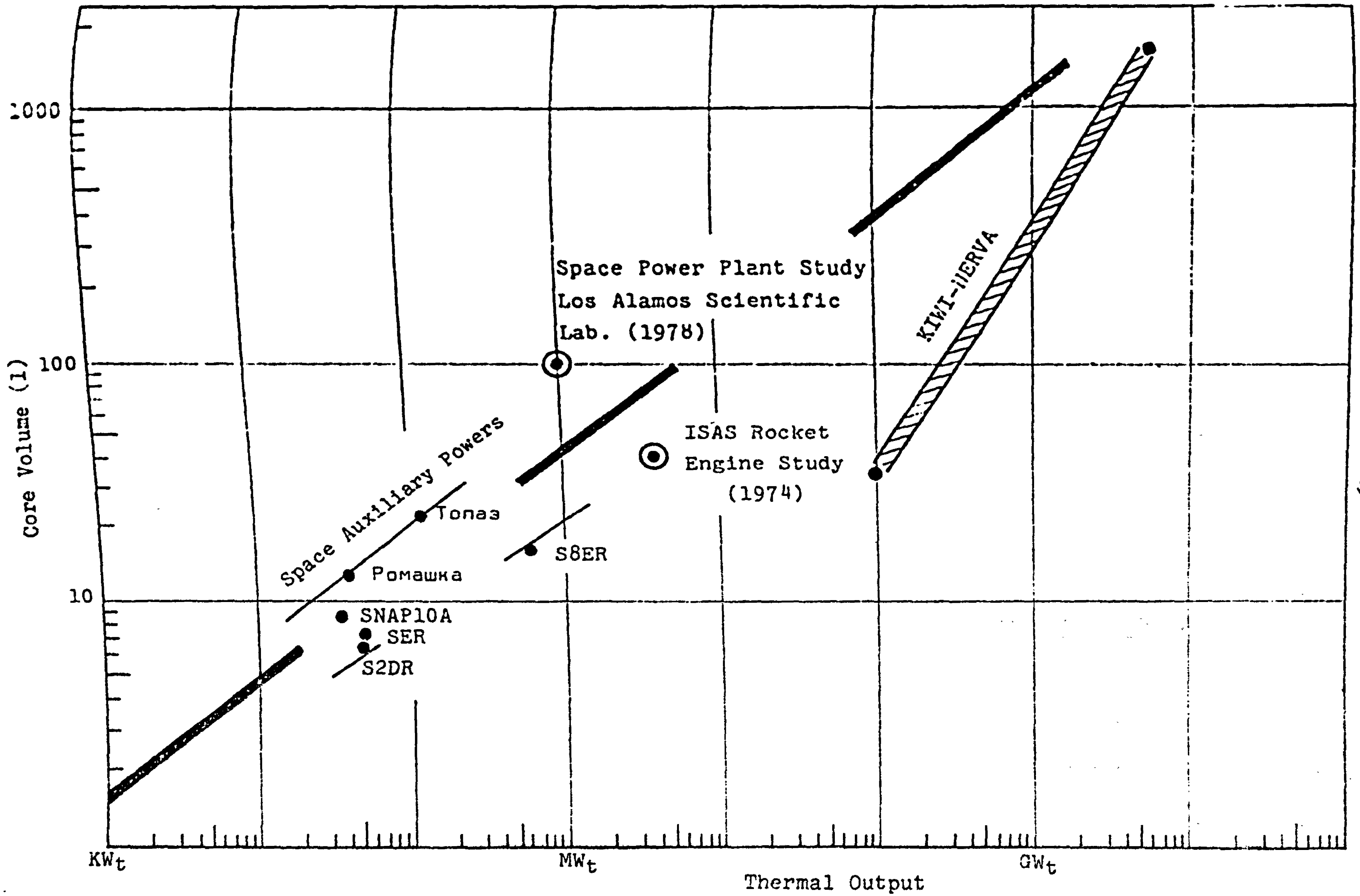


Fig. 9 Nuclear Reactors Thermal Output versus Core Volume

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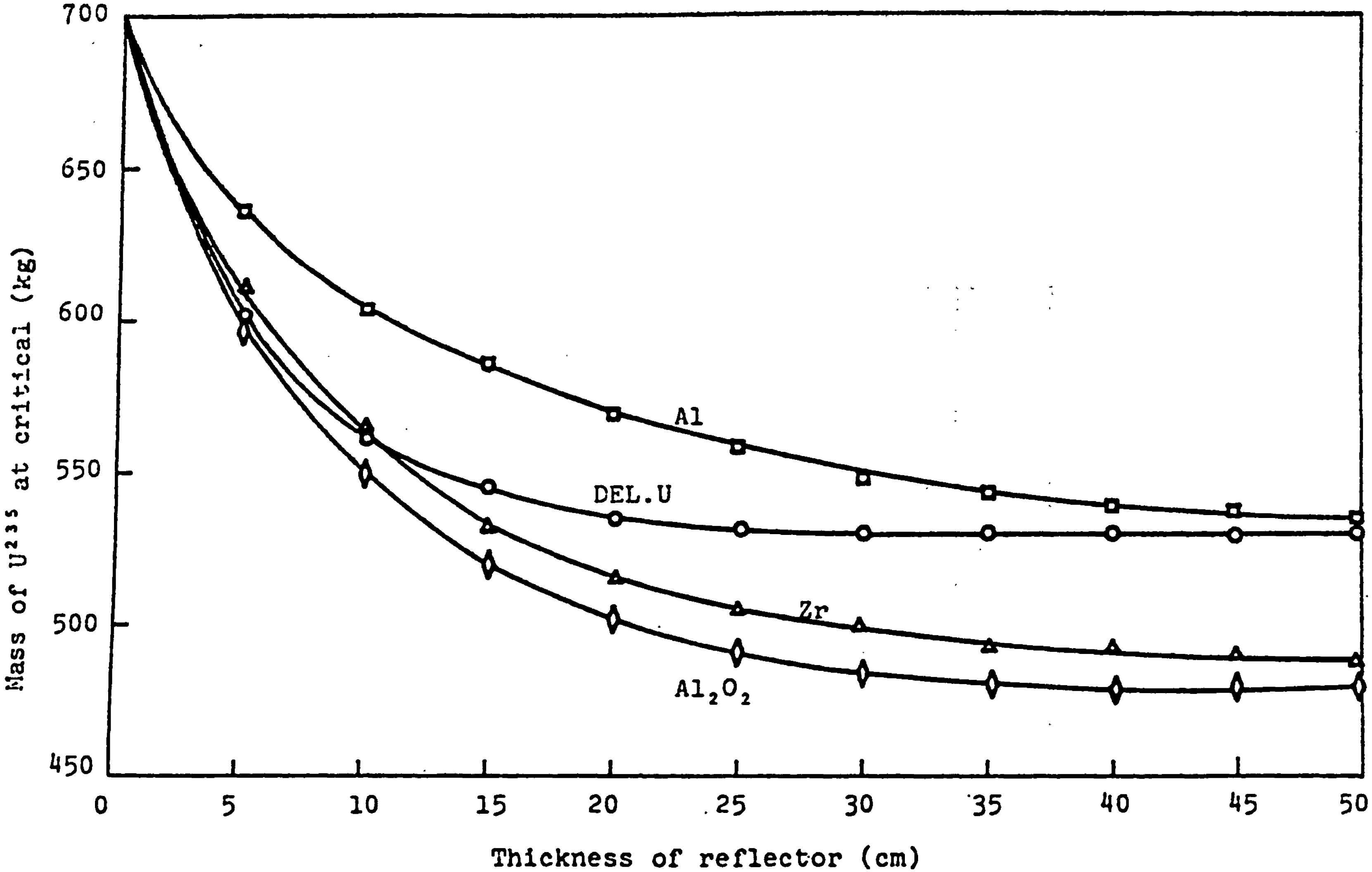


Fig.10 Mass of U^{235} at critical versus thickness of reflector (Ref.17)

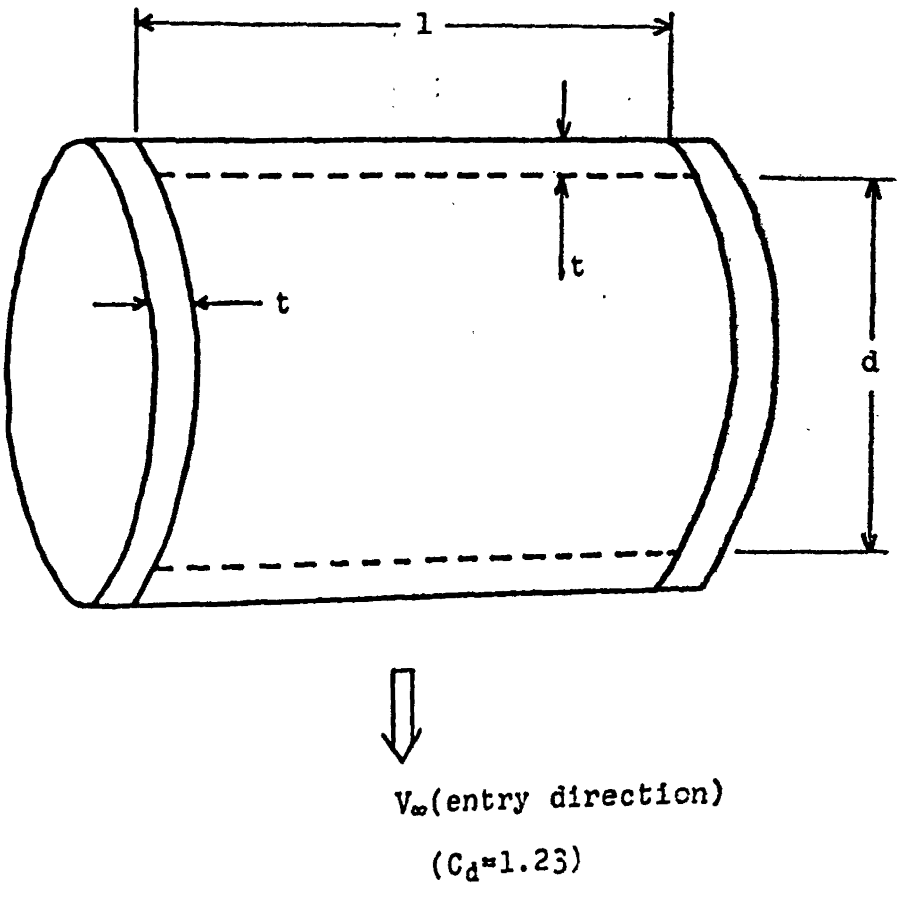


Fig. 11 Reflector on re-entry

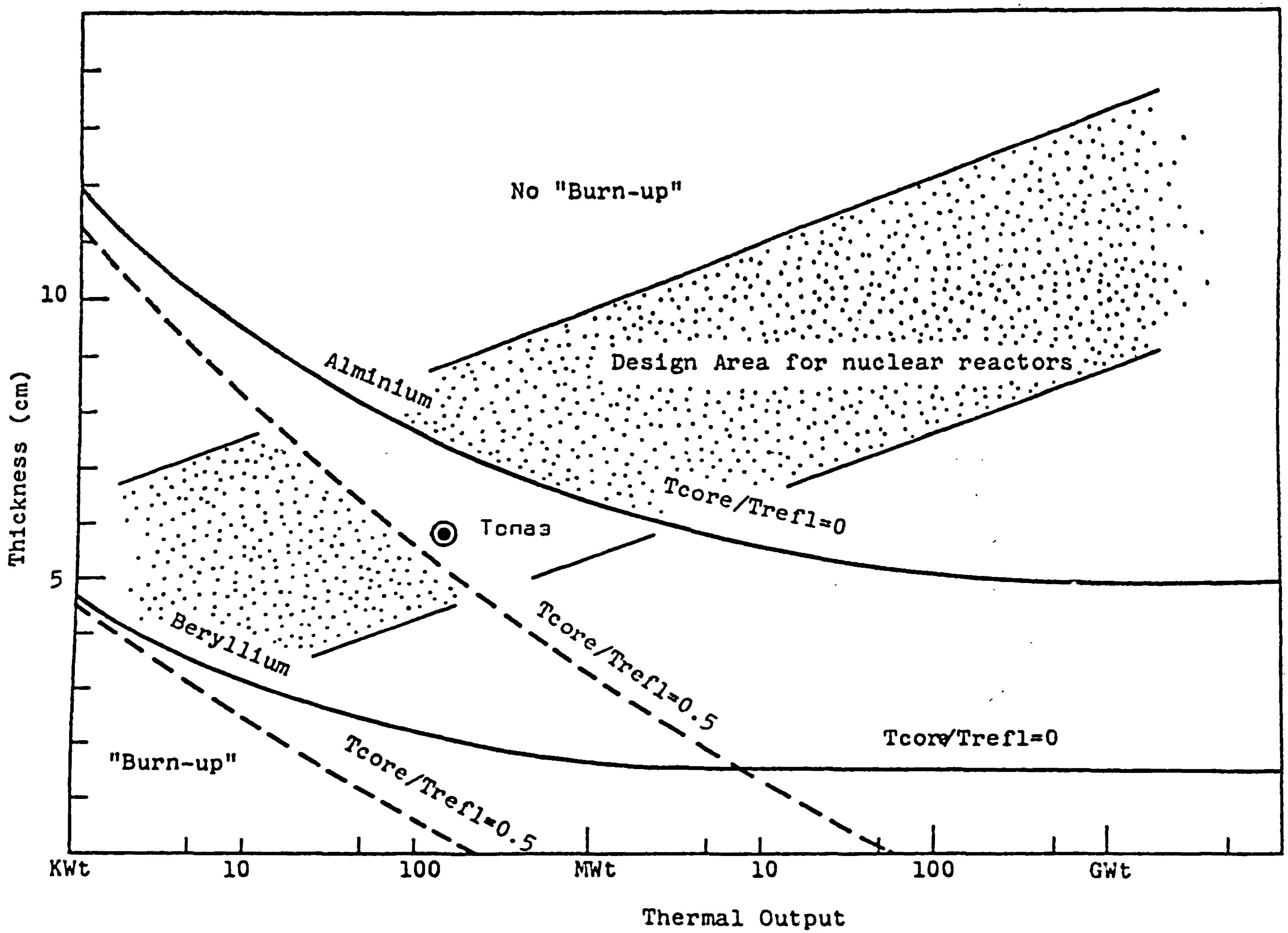


Fig.12 Melting of reflector by aerodynamic heating

(calculated on several assumptions)

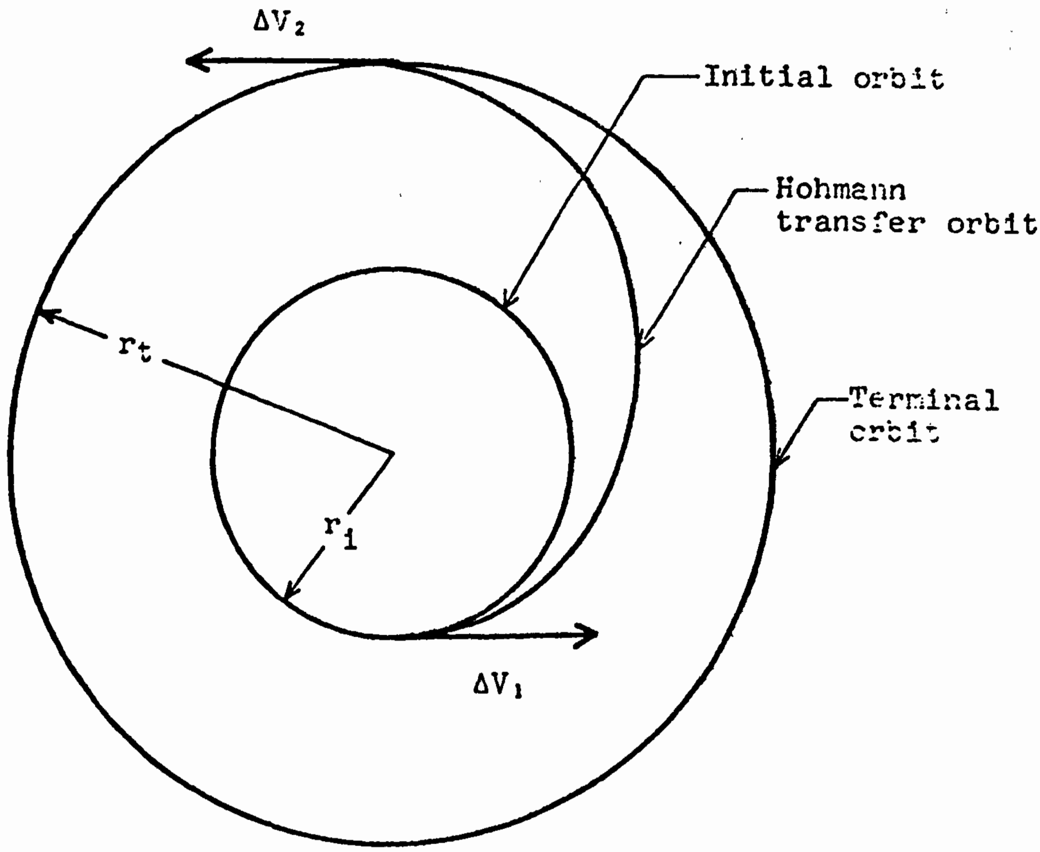


Fig.13 Reboosting via Hohmann transfer

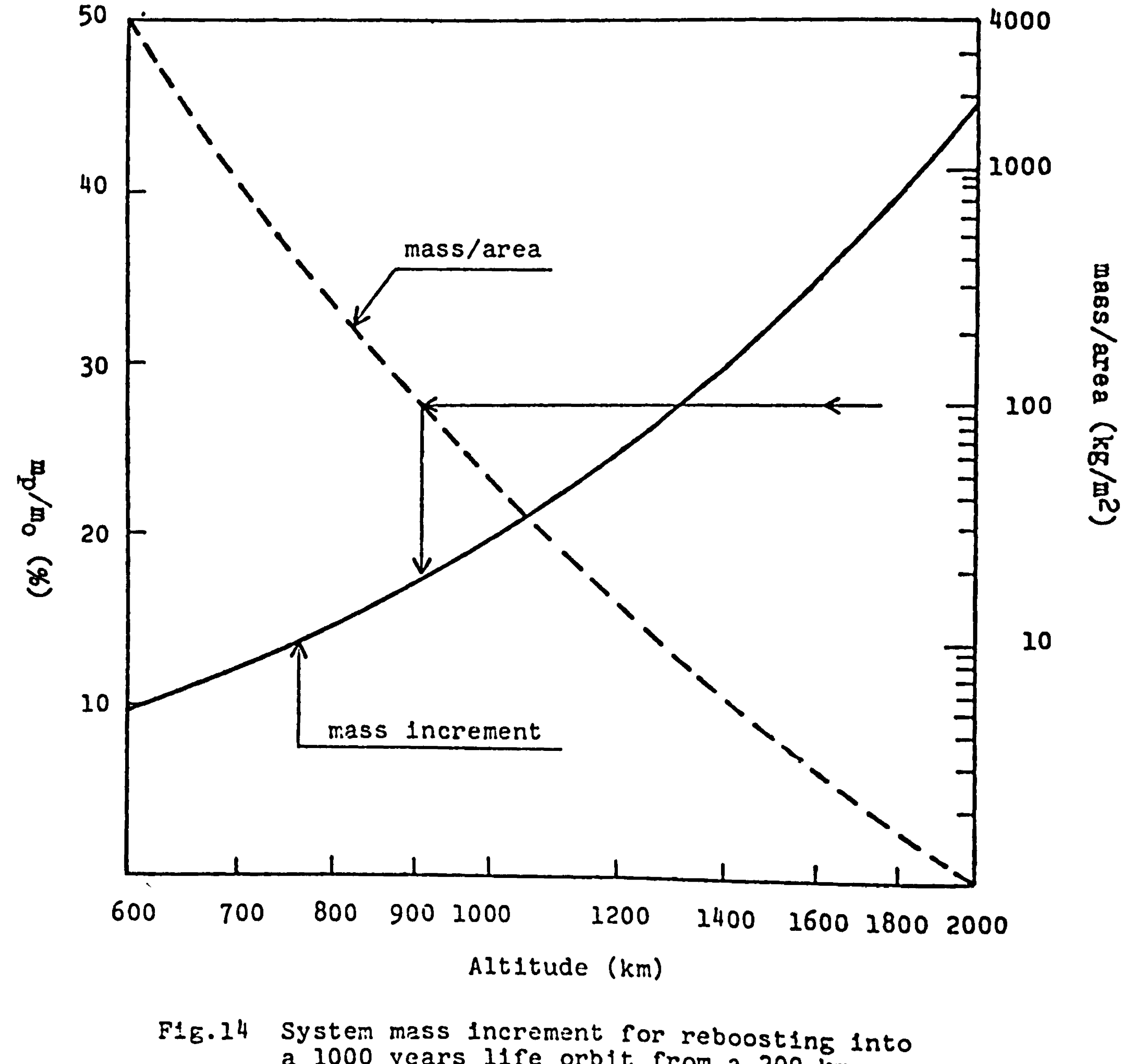
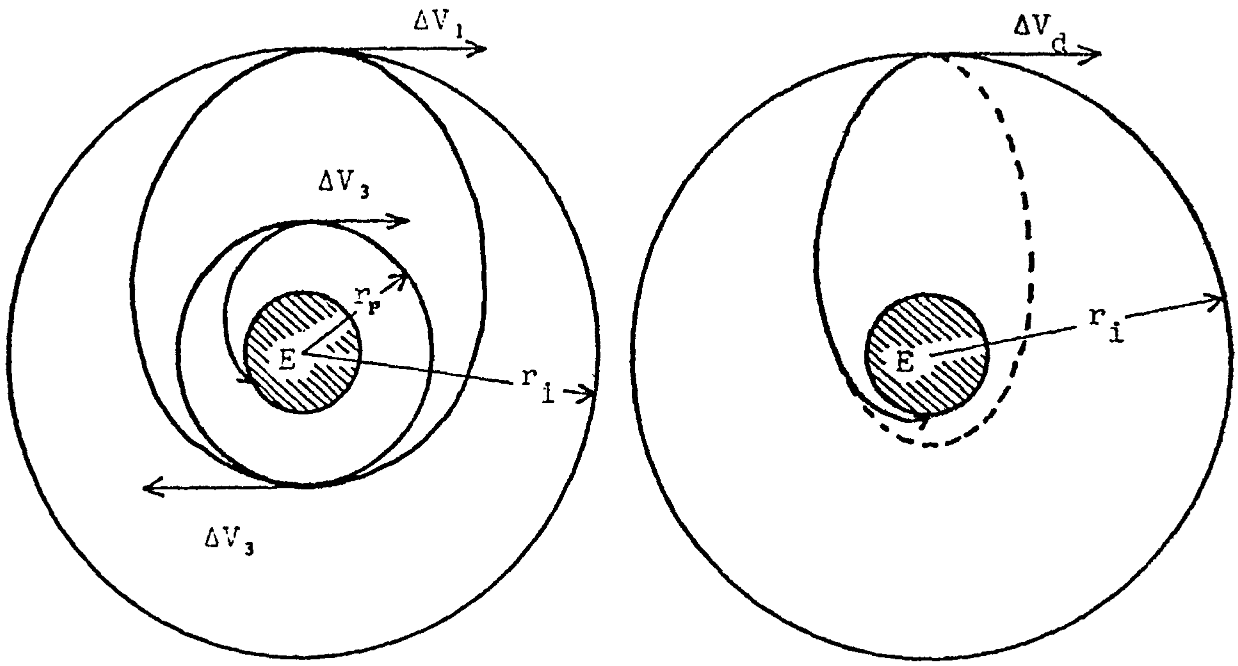


Fig.14 System mass increment for reboosting into a 1000 years life orbit from a 200 km altitude circular orbit



(a) Entry via parking orbit (b) Direct entry

Fig. 15 Two types of entry trajectories

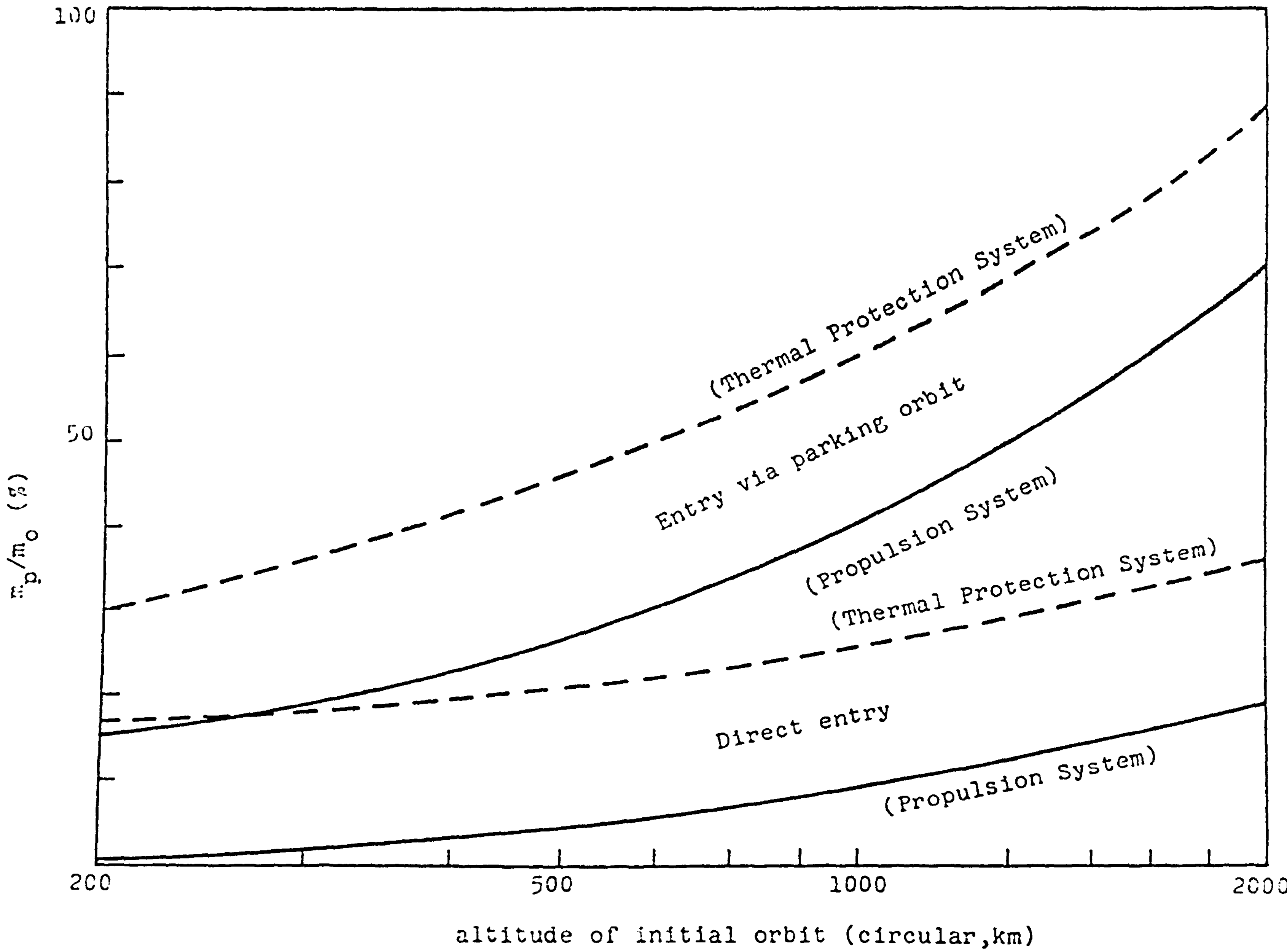


Fig.16 System mass increment for re-entry control

ANNEX 4

FORMAT OF NOTIFICATION

Format of notification for the spacecraft carrying nuclear power sources (NPS) should include the following items.

1. Prior to launch of the spacecraft

    (1) Name of launching State or States

    (2) Date and territory or location of launch

    (3) Basic orbital parameters

        (i) Nodal period

        (ii) Inclination

        (iii) Apogee

        (iv) Perigee

    (4) General function of the spacecraft

    (5) Information on NPS

        (i) Type

        (ii) Thermal and electrical output

        (iii) Structure and materials of parts

        (iv) Weight of each material

        (v) Composition size and weight of fuel or radioisotope

        (vi) Planned duration of NPS operation

    (6) Information on spacecraft

        (i) Weight

        (ii) Structure or profile

    (7) Information on safety measures

        (i) Planned Safety measures for used NPS
        (time and procedures for such measures as retrieval and transfer into higher orbit)

        (ii) Planned back-up measures for safety

2. On launch of the spacecraft

(1) Name of launching State or States

(2) An appropriate designator of the spacecraft or its registration number

(3) Date and territory or location of launch

(4) Basic orbital parameters

(i) Nodal period

(ii) Inclination

(iii) Apogee

(iv) Perigee

(5) General function of the spacecraft

3. Safety measures taken with regard to the spacecraft in orbit

(1) Retrieval or transfer to higher orbit

(i) Time and procedures of prior notification

(ii) Results of the safety measures taken

(2) Other safety measures if necessary

4. Before and after re-entry of the spacecraft

(1) When the spacecraft enters into orbit in which the spacecraft has a fair possibility of re-entry

(i) Forecasted orbit map

(ii) Latest orbital elements and drag parameters

a) Semimajor axis
b) Eccentricity
c) Right ascension of ascending node
d) Inclination
e) Augument of perigee
f) Mean anomaly
g) Drag parameters

(iii) Technical description of radio beacon or transponder if available for spacecraft tracking

(iv) Latest attitude of spacecraft

(v) Latest situation of NSP

a) History of NSP operation

b) Amounts in curie of radio-active materials like fission products

(2) Prior to re-entry

(i) Forecasted impact region

(ii) Precautionary measures to be taken by a State or States which may possibly be affected

(3) After re-entry

(i) Date and time of re-entry

(ii) Damage and radiological pollution caused by the re-entry

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