United States Patent

[11] **3,570,784**

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[21] Appl. No. **676,933**
[22] Filed **Oct. 20, 1967**
[45] Patented **Mar. 16, 1971**
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[54] **LOW BALLISTIC COEFFICIENT RADIOISOTOPE HEAT SOURCE**
13 Claims, 4 Drawing Figs.

[52] **U.S. Cl.** **244/1,** 176/75, 176/84, 250/106
[51] **Int. Cl.** **B64g 1/30,** G21c 3/30
[50] **Field of Search** 176/73, 75, 84, 76; 244/1, 138; 250/106

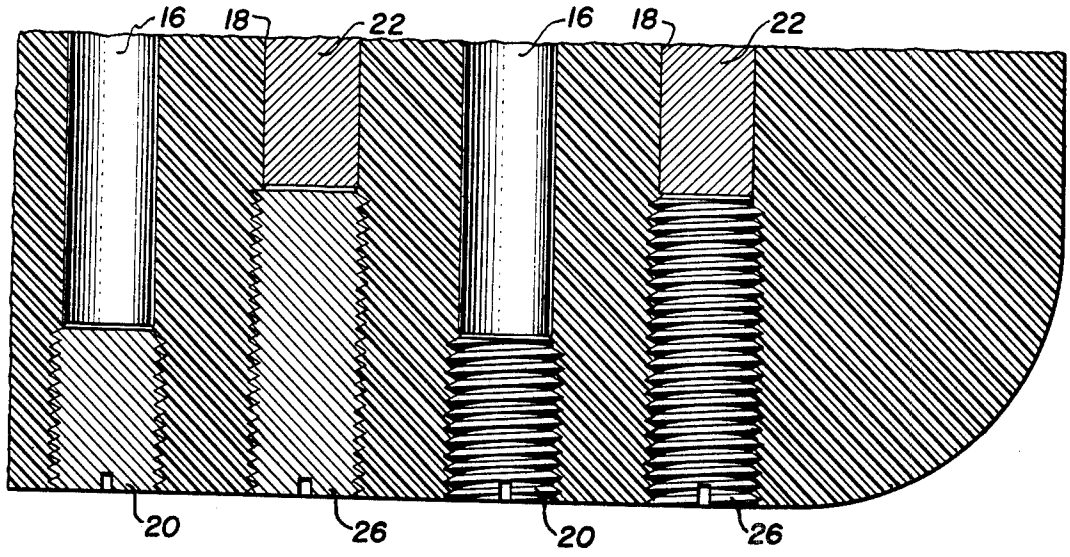
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ABSTRACT: A radioisotope containing heat source for use in artificial satellites and space vehicles. The source is configured so as to have a low ballistic coefficient allowing its random reentry into the earth's atmosphere at low terminal velocities. The low velocity prevents burnup and allows intact reentry of the radioisotopic fuel capsules. The heat source may typically be configured as a flat rectangular panel through which empty cylindrical channels are formed for lowering its weight and within which capsules containing the radioisotope are disposed. Graphite may be used to form the body of the panel.



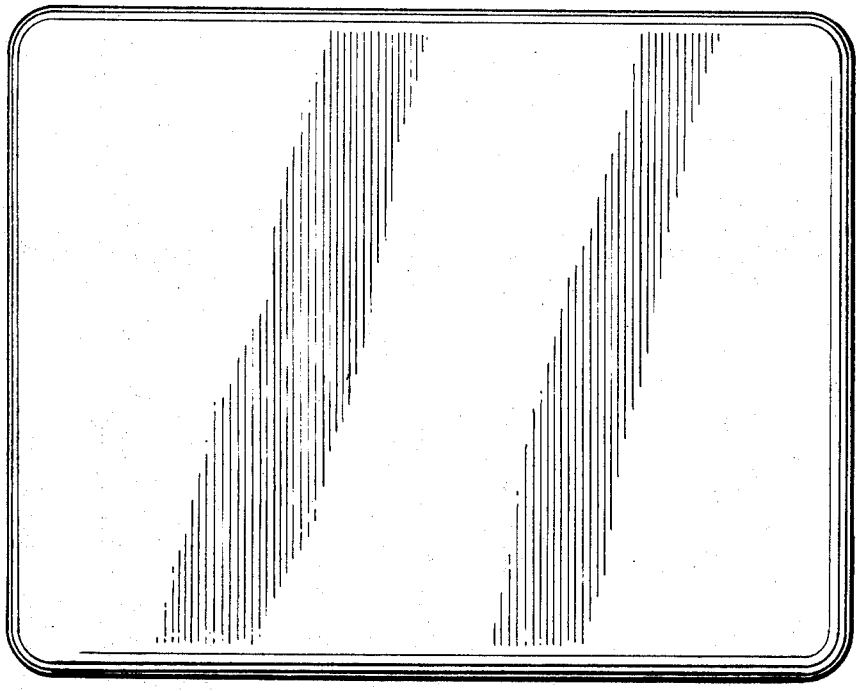


FIG. 1

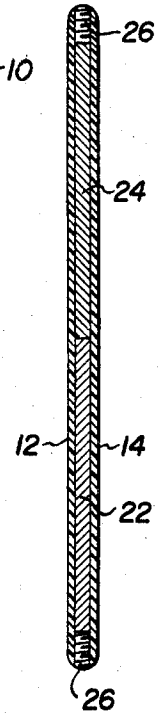


FIG. 3

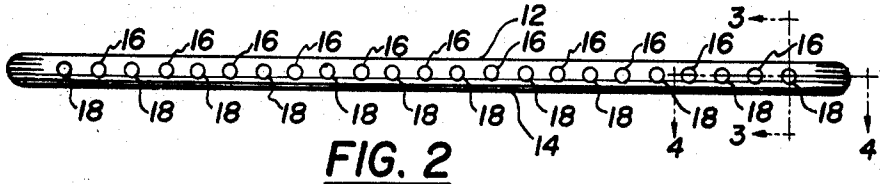


FIG. 2

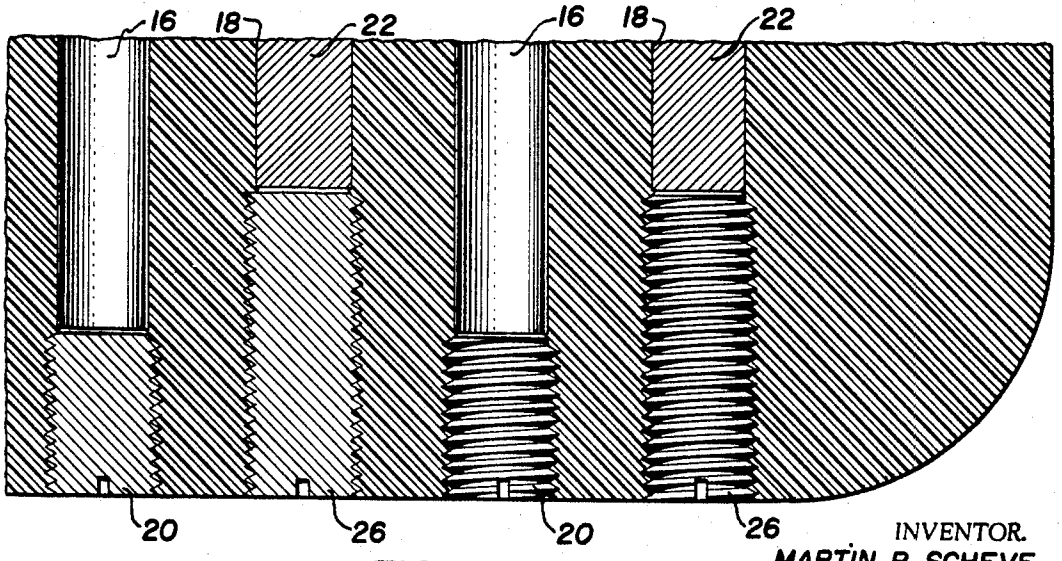


FIG. 4

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LOW BALLISTIC COEFFICIENT RADIOISOTOPE HEAT SOURCE

BACKGROUND OF THE INVENTION

1. Field of the Invention

This invention relates to power systems for artificial satellites, space traversing vehicles and the like and more particularly to a radioisotopic heat source for use with such systems which protects the fuel capsules during reentry.

2. Description of the Prior Art

The advent of highly sophisticated artificial satellites and space traversing vehicles has witnessed a catalysis within the scientific community of efforts for developing a broad spectrum of technical missions for space vehicles. This emphasis in exploiting the capabilities of orbiting vehicles and the like has been observed to range from astronomical and biological experimentation to systems of immediate practical utility, as evidenced in communications relay weather data collections and mapping missions. Evidencing further progress, space exploration programs have evolved a need for intermediately staged rendezvous satellites and the like for carrying on extensive and varied support functions.

As the technology associated with the space effort advances, there are generated new and correlative requirements for the launching of more complex, bulksome and heavy mechanisms. Additionally, as new satellite functions and their related systems become more complex and costly, practical economic considerations increasingly dictate that the orbiting missions be of relatively longer duration.

The design complexities encountered in accommodating all of the advancing technical desiderata have focused upon the general and practical requirements for highly reliable satellite instrumentational and related systems. Additionally, in order to gain maximum use of the payload capacities of existing launch vehicles, not only the instrumentation but also the support systems of the satellites must incorporate the highest of design efficiencies.

The effort of developing enhanced system efficiencies has, in particular, delved with concern into the need for improved artificial satellite powering systems. These systems have long introduced design burdens and restrictions resulting from their relatively bulksome size, weight and shape; their presentation of undesirably high profile drag areas, and their somewhat limited reliability and effective operational lifespans. It follows that the industry would be most receptive to the development of a power supply of long and reliable lifespan which, additionally, enjoys relatively low bulk and weight. Ideally, the power supply should be amenable to modular design approaches, thereby facilitating its incorporation within vehicles and launch systems now in design, production and use.

The limitations as above noted along with others have been seen to present themselves in one form or another in power systems now considered conventional. For instance, conventional batteries, while affording a relatively stable power output, impose an oppressively high weight penalty upon the launch vehicle. This weight factor necessarily must detract from the mass allowance allocated to the instrumentation payload. Further detracting from the use of batteries is their short operational lifespan.

Efforts to expand the operational lifespans of space devices have often devolved upon the use of solar cells. These devices, operating to photoelectrically convert light energy into electrical energy, are assembled within large planar banks to form panels. The panels are extended during orbit to collect solar radiation. While retaining some advantage of lower weight and holding the promise of longer lifespans, the panels have encountered operational restrictions due to their inherent delicacy and consequent tendency to degenerate during use. The presence of an extensive sail area is also considered undesirable by virtue of the panel's large profile area tending to promote orbital decay. Supplementary power systems are also required with the panels for earth orbit applications in order to accommodate movement within the earth shadow.

The power system considered to hold significant promise for meeting the rigorous demands imposed by space vehicle design criteria is that utilizing a radioisotopic heat source. In most applications these heat sources are associated with thermoelectric devices to evolve an electrical power output. Known generally as radioisotopic batteries or generators, the devices usually comprise a relatively small quantity of heat generating radioisotope which serves to heat one end of a number of interconnected thermoelectric elements. The thermoelectric elements, formed of certain semiconductive materials, are joined in pairs to form thermocouples, which when heated at a selected end serve to statically generate an electric current. An electrically interconnected array of thermocouples is generally referred to as a thermopile.

By virtue of the long half-lives which may be realized through appropriate selection of the radioisotope, a long term unattended source of heat and power is availed with this generation system. Additionally, the radioisotopic material itself along with the thermopiles utilized are of advantageous low bulk and weight. To the present, however, these distinct advantages have been considerably outweighed by the design criteria required of the supporting, containment and safety implements mandatorily incorporated with the system. The limitational criteria are reviewed in the following paragraphs.

In order to function efficiently, the thermocouples must be maintained within a certain ambient environ and must be heated in a manner maintaining a preselected differential of temperature across their individual lengths. The designs for radioisotopically heated thermoelectric units heretofore presented generally have assumed a somewhat cylindrical shape wherein a central radioactive heat producing core is surrounded on as many sides as possible by closely fitted clusters of thermocouples. By so clustering the thermocouple arrays, a degree of maximized consumption of the radioisotope heat energy is thought to be realized. In order to establish and maintain a requisite differential of temperature across the thusly arrayed thermocouples, it is necessary to introduce and interconnect heat conducting and disposing systems from the cold ends of the thermocouples to ambient surroundings. This disposal arrangement is usually provided by somewhat elaborate banks of radiative fins. To further inject a degree of heat distribution control, various forms of insulation are inserted about the thermocouple arrays and a protective inert atmosphere is introduced into portions of the generator housing.

Thusly deployed about the centralized heat source, the assemblage of thermoelements in most instances becomes structurally elaborate, close tolerances and difficulties of installation being the rule rather than the exception. To further add to their bulk and complexity, radiation shielding must also be incorporated within the device housings.

Assembled under the thusly described conventional design approach, the radioisotopic generators of relatively larger power capacities have been characterized as overly bulksome, heavy and intricate, requiring elaborate fin structures for heat dissemination as well as regulated safety procedures for avoiding radiation exposure.

A dominating hindrance to the widespread adaptation of radioisotopically powered electric generators to satellites and space vehicles has centered about considerations of aerospace nuclear safety. When adapted for attachment to an artificial satellite, the difficulties attendant with utilizing radioisotopic or other nuclear power devices become considerably involved. Three complexities will be immediately apparent to those skilled in the art, namely, the problem of shielding launch personnel from radiation hazard during and before launching; the protection of contiguous payload instrumentation from radiation damage or interference; and, of considerable importance and difficulty, the disposal of the radioactive products used within space vehicles before or during their reentry into the earth's atmosphere. The present invention is particularly addressed to the latter, most complex problem.

When injected into an adequately high orbit, for instance in the order of about 600 nautical miles, a satellite, without being manipulated otherwise, will remain orbited for an extended period of time. Contemporary computation allocates a multicentury orbital life to such altitudes before terrestrial reentry risks become high. Inasmuch as the half-life characteristics of the radioisotopic fuel will effect a gradual diminution of the intensity of radioactive emission, the risk of unacceptable earth contamination following a multicentury orbit is nominal.

The probabilities for inadequate injection into earth orbit, however, are of such a nature that disposal schemes must be programmed into radioisotopically powered satellites. During the recent past, two basic approaches to disposal have been prevalent within the industry. The initial approach has been to provide for destruction of the radioactive source during atmospheric reentry. Generally, the heat developed during reentry serves this function. Along with this reentry burnup, there is effected a broad dispersal over a portion of the earth of the contaminating radioactive product. Thusly dispersed over a significant oceanic or terrestrial area, it has been earlier considered that the radioactive fallout reaching earth will be of acceptably low levels or intensities. The latter consideration is presently the subject of reevaluation and as a consequence, such dispersal schemes are not received with favor.

The alternate approach to the problem of disposing of the radioisotopic power sources of satellites is an active one, as opposed to the passive arrangement described above. This technique contemplates a controlled return to earth of the vessel holding the radioactive source and is generally programmed either by rendezvous or controlled reentry schemes. Upon being returned to earth, the radioisotopic source is maintained in an intact state, as opposed to being dispersed. This intact status is now considered desirable. The immense costs associated with either of the active recovery techniques will be immediately apparent.

Where the use of a reentry vehicle is contemplated, there is involved separate retro power systems, logic control systems, heat shielding vehicular reentry orientation devices and terminal velocity controls for avoiding earth burial. Such complex reentry vehicles must also be fabricated so as to assure reliable operation following an orbit of an extensive period of years. The latter requirement follows from the above-discussed lengthy half-lives of the heat sources currently found acceptable for power generation purposes. For instance, after 20 years of orbit, a conventional fuel will still retain excessive levels of activity. Difficulties are further encountered in providing certain of the materials necessary for the reentry vehicles. Particularly, the ablative materials developed at present for reentry heat shielding are not immune from radiation damage where long term exposure is contemplated. As a result, more elaborate and weight contributing structures are necessitated for blocking heat shields and other sensitive equipment from damaging radiation. It follows that the necessarily complex reentry vehicles will most undesirably detract from any satellite instrumentational payload. Aerospace nuclear safety demands, however, have been seen to insist upon the imposition of this weight penalty in the absence of other and adequate solution.

SUMMARY OF THE INVENTION

From the above review of present day approaches to designs for artificial satellite or space vehicle power supplies, it will be apparent that each fails in one measure or another to provide all of the desired design attributes for a satellite power system. The present invention looks to the promising characteristics of the radioisotopic fueled power system while at the same time proffering solution to an otherwise highly complex requirement for aerospace nuclear safety.

The power system of the invention is characterized in providing a radioisotopic heat source which in and of itself serves as a vehicle having the capability of reentering the earth's atmosphere and descending to earth in an adequately intact physical state. Through the use of a heat source structure having appropriate aerodynamic properties, the invention provides for the reentry of radioisotopic fuels without necessitating supplementary and extraneous active reentry controls and implements.

The heat source of the invention is fashioned of shape and weight so as to retain a relatively low ballistic coefficient. By virtue of its characteristic low ballistic coefficient, reentry heat fluxes and temperature peaks are maintained within acceptable tolerances. The aforesaid low ballistic coefficient permits reentry of the heat source at adequately low terminal velocities. As a result of the relatively low terminal velocities at which reentry of the heat source is effected, forces encountered at earth impact are maintained at such low levels as to preclude the possibility of damage to the radioisotopic containment structure.

Another object of the invention is to provide a heat source reentry structure having earth impacting characteristics minimizing the probabilities of earth penetration and burial.

An additional object of the invention is to provide a reentering radioisotopic heat source capable of fabrication in a variety of shapes and which is readily adapted to use with efficient generator structures.

By virtue of its relatively low reentry velocities, the heat source arrangement of the invention is amenable to earth based tracking procedures thereby allowing for efficient disposal of its retained radioisotope following return to earth.

A further object of the invention is to provide a radioisotopic heat source for artificial satellites and the like which retains a capability for automatic ejection or separation from the satellite body during the period of reentry into the earth's atmosphere.

Another object of the invention is to provide a radioisotopic heat source which is of simple design and which may be incorporated within thermoelectric generator structures of advantageous high efficiencies, low weight and low bulk.

It is a further object of the invention to provide a method for recovering a radioisotopic material from earth orbit through the use of a heat source configuration characterized in having a low ballistic coefficient.

These and other objects and advantages of the invention will become apparent from the following detailed description and drawings.

BRIEF DESCRIPTION OF THE DRAWINGS

FIG. 1 is a pictoral representation of a heat source structure fabricated in accordance with the invention.

FIG. 2 is a front elevational view of the radioisotopic heat source of FIG. 1.

FIG. 3 is a side sectional view of the radioisotopic heat source of the invention taken along the plane of line 3–3 of FIG. 2

FIG. 4 is a sectional and partially broken away view of the radioisotopic heat source of the invention taken along the plane of line 4–4 in FIG. 2 and revealing internal structure.

DESCRIPTION OF THE PREFERRED EMBODIMENT

Looking to the drawings, the remarkable simplicity of the structure which may be developed from the concept of the invention is readily apparent. In the FIGS. a heat source structure shown generally at 10 is fashioned as a flat plate having opposed parallel planar surfaces 12 and 14. The structure 10 is formed in a generally rectangular shape and is of relatively small thickness. All edges and corners of the structure are shown as rounded for improved aerodynamic performance although this detail will be seen to be unnecessary for effective reentry execution.

Extending through the structure 10 in regularly spaced parallel fashion are a plurality of cylindrical bores or channels 16 and 18. Bores 16 are spaced in alternating fashion with bores 18 and, for descriptive purposes, are referred to as "lightening holes." Each of the lightening holes 16 is closed at either end by a blanking plug as depicted at 20 in FIG. 4. The

blanking plugs **20** are threadedly engaged with the internal surface of the outward portion of channels **16** and are outwardly notched to receive an appropriate insertion and tightening tool.

Disposed adjacent each of the lightening holes **16**, the alternate bores **18** are adapted to receive a pair of radioisotope capsules as at **22** and **24**. Capsules **22** and **24** retain any of a preselected isotopic fuel suited for the generation of an appropriate quantum of heat. Such fuels as SR–90, Pu–238, Po–210 or the like are typical of those selected for the instant purpose. As is indicated in the drawings, the capsules may be fashioned having a cylindrical shape and their structural configuration with follow the dictates of conventional fuel encapsulation techniques. For example, the fuel is generally contained within a cylindrical metallic liner having a relatively high strength at elevated temperatures. Disposed intermediate the fuel and liner there is conventionally deposited a metallic coating of similar protective surface selected for its chemical compatibility with both the fuel and liner. If required, an oxidation resistant outer clad is positioned over the liner.

The fuel capsules **24** are retained within the channels **18** by plugs as shown at **26**. Plugs **26** threadably engage the body of the heat source in similar fashion as plugs **20**. It will be noted, however, that the capsule retaining plugs **26** are comparatively extended in length. The added length of plugs **26** serves to position the outwardly dispose termini of capsules **22** and **24** a select distance inwardly from the outer edges of the heat source body structure.

From the foregoing description it will be seen that the exemplary heat generator **10** is simply a relatively flat and somewhat rectangular panel. When appropriately fueled by the capsules **22** and **24**, the panel **10** will constitute a heat source. Thusly configured this heat source may be incorporated within a myriad of electrical power generating devices. The structure as described is particularly suited for positioning between two or more planar banks of thermopiles. Such sandwich or multilayered types of thermoelectric generators avail numerous design advantages while promising enhanced power generation efficiencies.

At the termination of its lifespan of operation within an earth orbiting satellite or space vehicle, the heat source **10** is separated from its vehicle and permitted to randomly descend into the atmospheric layer. Because of the low resultant ballistic coefficient deriving from its structure, the panel will descend through the atmosphere at a relatively low velocity. This low velocity will prevent burnup otherwise resulting from excessive heat flux and will also avoid such peak body temperatures as would otherwise damage and destroy the integrity of the fuel capsules **22** and **24**. The characteristics ballistic coefficient of the panel or source **10** will also derive terminal velocities of such values as to minimize the possibility of penetration into the earth and consequent burial at the time of impact. At the instant of impact the material, such as graphite selected for the body of the panel **10**, will absorb and dissipate through the fracture or the like the shock of impact, leaving the fuel capsules **22** and **24** intact for physical recovery. As is evident, the heat source assembly as a whole is used uniquely as the primary fuel container for reentry.

Separation of the panel **10** from the satellite and contiguous generator implements may also be uniquely simple. For instance, as the orbit of the space vehicle degenerates and insertion into the atmospheric layer commences, the material of the vehicle will heat up and, eventually, burn and disintegrate. The panel **10** readily may be retained within the generator structure of the space vehicle by conventional explosive bolts. These bolts may be selected so as to explode at a predetermined temperature, thereby releasing the panel to descend in isolation. Alternately, the retentive framing of the vehicle may be allowed to merely burn away from the panel, thereby releasing it to the atmosphere. Of course, the panel **10** may be precisely ejected from a satellite through the media of a command signal generated by radio transmission from earth. Because of its resultant low ballistic coefficient, the panel **10** will reenter at such speeds as to permit facile tracking by earth communications stations. This ballistic coefficient will be seen to represent that realized by a randomly rotating or spinning body. Thusly passively manipulated, the panel **10** will tend to develop higher temperatures about its edge or periphery. For this reason, the plugs **26** were observed to position the fuel capsules **22** and **24** a select distance inwardly from the panel edge to avoid high temperature damage. The empty channels **16** serve to reduce the weight of the panel **10** while not detracting from its surface configuration nor appreciably from its strength.

Turning now more specifically to the theory or basis of operation of the inventive reentering heat source, there exists a series of critical factors to be dealt with in order to assure intact reentry. These factors concern aerodynamic heating, terminal velocity, impact burial and the peak temperatures attained during descent. Aerodynamic heating must be controlled such that reentry heat flux will not burn up the panel. Temperature peaks, if excessive, would threaten the integrity of the fuel capsules. Impact burial is to be avoided inasmuch as burial will tend to concentrate or confine the thermal energy extant within the panel, thereby jeopardizing the integrity of the fuel capsules. The reentry velocity will be seen to affect the above factors as well as the degree of impact shock.

Basic to the success of the heat source panel configuration selected is the value of its ballistic coefficient. Defined as the relationship W/C_DA, the coefficient must be held to a low value. While it is desirable that the heat source body have a ballistic coefficient as low as possible, it has been determined that satisfactory results for the purpose of this invention can be realized utilizing bodies having ballistic coefficients less than 20. In the stated relationship, W represents the weight of the body, A is the area of profile presented to the direction of flight and C_D is the drag coefficient of the body. It will be apparent that for the present invention, the values of A and C_D are desirably as large as practical while the weight W should be low.

Returning to the earlier mentioned critical factors or parameters of performance to be contended with, the reentry heat flux dq/dt is generally represented by the following relationship:

$$dq/dt \alpha f\left(\frac{W}{C_D A}\right) \tag{1}$$

Similarly, the terminal velocity V of the reentering body may be represented by the relationship:

$$V \alpha f\left(\sqrt{\frac{W}{C_D A}}\right) \tag{2}$$

The extent of burial depth X of the body may be represented by the relation:

$$X \alpha f\left(\frac{W}{A}\right) \log \frac{W}{C_D A} \tag{3}$$

As may be evidenced from relations (1) through (3), a minimal weight, coupled with larger profile areas and values of drag coefficient will effect desirable restrictions upon the factors; reentry heat flux, terminal velocity and burial depth.

Continuing to the critical factor of limiting the temperature obtained, the temperature T of the panel at any time during reentry may be represented by the relationship:

$$T \alpha f\left(\frac{P}{A}\right) \tag{4}$$

where A is the profile area and P is a thermal body factor. Conventionally the value of T has been held at lower levels by the use of ablative heat shields disposed over profile areas A which, by design necessity, were of lower extent than desirable. In the present invention, the value of A is relatively large, while the value of P is maintained at adequately lower levels through selection of materials having high emissivity and low absorbtivity. The thermal body factor P may, in practice, be found as the expression Q/ϵin another equation deriving the temperature T_R of a radiating reentry body:

$$T_R = \sqrt[4]{\frac{g}{\epsilon \kappa A} - T_s^4} \quad (5)$$

where: g is the amount of heat dumped into the environ (KW); ϵ is the emissivity of the reentry body; κ is Boltzman's constant and T_s is the temperature of the sink area. For practical purposes, within a space environ the value of T_s goes to zero and may be disregarded. From equation (5) it may be seen that the value of P is minimized by selecting materials of high emissivity and low absorbtivity.

In view of the above discussion of theory, it will be seen that a broad variety of practical shapes may be selected for the heat source of the invention. The panels **10** may be curved rather than flat while still retaining adequate values of ballistic coefficient and peak temperature. The value of the empty channels **16** within panel **10** becomes apparent when considering the desirability for lowering the weight factor W. Materials selection may be varied to suit requisite weights, peak temperatures and design velocities.

A heat source designed as the panel **10** having a $36 \times 36 \times 1$ inch size and formed of graphite will have a ballistic coefficient of about 11.0. Its large surface area will reradiate much of the reentry heat back into space resulting in a capsule temperature of less than 2000° F. except for 50 seconds during reentry when the temperature may rise to a maximum of 2500° F. The terminal velocity of such a device is only 125 feet per second, a value negating the possibility of direct burial, even in very soft earths.

Typical of other materials which may be selected for the heat source is beryllium. Where very low ballistic coefficients are availed, the heat source structure may be fashioned from a Haynes alloy or columbium honeycomb configuration.

It will be apparent to those skilled in the radioisotopic generator and artificial satellite design arts that many variations may be made in the detailed disclosure set out herein for illustrative purposes.

I claim:

1. A radioisotopic heat source for use in supplying power to space vehicles and the like, said heat source being adapted for transport from outside the earth's atmosphere to the surface of the earth and comprising:

a. a streamlined, planar body structure adapted to supply thermal energy to a space vehicle power system;

b. said structure being shaped so as to develop a relatively high average profile area during free fall through the earth's atmosphere;

c. said structure being formed having a weight with respect to said area permitting it to retain a substantially low ballistic coefficient during free fall through the earth's atmosphere; and

d. containment means disposed within said body structure for retaining a preselected quantity of radioisotopic fuel remote from the edge thereof.

2. The heat source of claim **1** wherein the said body structure is formed of a material having low density, high melting point, high emissivity and low absorbtivity.

3. The heat source of claim **1** wherein said body structure material is graphite.

4. The heat source of claim **1** wherein said body structure material is beryllium.

5. The heat source of claim **1** wherein said body structure is configured having an average ballistic coefficient value less than **20**.

6. The heat source of claim **1** including channel means disposed within said body structure for reducing the weight of said structure.

7. The heat source of claim **1** wherein said containment means comprise a plurality of spaced channels each adapted to retain an encapsulated radioisotopic material in intimate, thermal exchange relationship.

8. The heat source of claim **7** wherein said containment means and said channel means are alternately disposed within said body structure in parallel spaced relationship.

9. The heat source of claim **1** in which said body structure is fashioned substantially as a thin parallelepiped.

10. The heat source of claim **9** wherein said containment means comprise cylindrical channels disposed in spaced parallel relationship across said body; and including plug means situate at the termini of said channels for retaining said fuel within said channels.

11. The heat source of claim **10** including channel means disposed within said body structure in alternating parallel spaced relationship with said containment means channels.

12. The heat source of claim **10** wherein said radioisotopic fuel is retained within said channels a select distance from the periphery of said body structure so as to remain isolated from the heat energy developed by the aerodynamic heating of said periphery during free fall through the atmosphere.

13. The heat source of claim **12** wherein said body structure is formed from graphite.