**UNIQUE FEATURES
OF DYNAMIC ISOTOPE POWER SYSTEMS
FOR HIGH-REQUIREMENTS SPACECRAFT**

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Abstract

The Dynamic Isotope Power System, designed for spacecraft requiring prime power in the 500-to-2000 watt range, has been successfully built and ground tested. A number of studies, summarized herein, have demonstrated the advantages of using such a power system instead of the conventional solar system, for a variety of earth-orbit missions. These advantages stem from the unique nature of the dynamic isotope system, different in kind from solar power systems. As a result, in many cases, the spacecraft design can be significantly simplified and more closely harmonized with mission requirements. This overall advantage can be crucial in missions which have stringent pointing, stability, viewing and/or positioning requirements.

Introduction

Since 1972, when the Atomic Energy Commission began to consider the development of radioisotope-fueled dynamic heat engines (i.e., closed-cycle turbine-alternator systems) for a number of candidate military and civilian space missions, Fairchild Space & Electronics Co. and others have studied spacecraft integration and design interfaces which make optimum use of the unique features of such power systems. Since 1975, the Department of Energy has been pursuing the development of the Organic Rankine Dynamic Isotope Power System (DIPS). As this development program has now resulted in the successful ground operation of the DIPS, it is useful to review the unique and unusual capabilities of this power system in the context of mission requirements and spacecraft designs.

In the past, space isotopic power systems have been available only in the form of Radioisotope Thermoelectric Generators (RTG's). Because of the relatively low conversion efficiencies of these devices (5-7%), RTG's have generally been limited to missions requiring less than 500 watts_e. The higher efficiency DIPS (15-to-20%), has been pegged initially for the power range of 500-to-2000 watts_e, although these are not the limits of its capability. Economy in the use of costly radioisotope fuel (Pu-238) is an obvious advantage of the DIPS. The prospect of recovery and reuse of the radioisotope fuel via the Space Shuttle could further improve the economic potential of this approach for Earth-orbit missions.

The advantages of DIPS-powered spacecraft designs tend to become more significant in missions which place the most stringent requirements (defined in more detail below), on conventional spacecraft.

The use of solar power usually brings with it requirements for mechanical or physical orientation; additional deployments; possible blockage of sensors, thrusters, and/or antenna beams; batteries (and their control systems); attendant thermal control requirements; vulnerability to trapped radiation (natural as well as enhanced) and to incident laser radiation.

While these requirements and limitations are normally accommodated at reasonable cost in most earth-orbiting spacecraft, they become more and more difficult and costly, and are sometimes wholly impractical to accommodate in spacecraft which must provide for high-level mission requirements as well.

Candidate Missions of Interest

The earliest of the previous studies dealt with the design of a multi-mission spacecraft for the economic benefits this would produce (fig 1)¹. These studies were updated in 1978-9 in a study which illustrated the advantages of using the thermally-integrated design approach² in a multi-mission spacecraft (fig 2)^{3,4}. Such a spacecraft provides all of the necessary housekeeping functions, including all prime power and payload thermal control. These last two functions are not normally (or readily) available in a solar-powered multi-mission spacecraft.

Fig. 1. DIPS-Powered Standard Spacecraft; Early Fairchild Version (1973)

Figure 2. Nuclear-Integrated Multi-Mission Spacecraft (NIMS) 1979 Version (References 3, 4)

Throughout these and other studies, certain key and unique features of the nuclear-powered designs have emerged, such as: the ability to fly in any orbit without modification, the ability to configure and to orient the spacecraft in much closer conformance with mission requirements, and the ability to eliminate flexible appendages and moving mechanical assemblies which are common to a large number of solar-powered spacecraft.

These advantages apply to many missions, but most strongly to a class of missions which have either one or a combination of the following stringent requirements:

1. Highly accurate knowledge of spacecraft position.
2. Very precise control of spacecraft orientation
3. Extremely stable platform for sensor mounting

The Global Positioning System (GPS Navstar) mission carries only the first of these requirements. On-board knowledge of spacecraft position to within a few meters is required at all times. A 1977 Fairchild study⁵ revealed the advantage of a nuclear-dynamic integrated design for this application, in which the unpredictable portion of spacecraft ephemeris drift is reduced by an order of magnitude or more, compared to the conventional solar-powered design.

The Deep Space Surveillance System (DSSS) is an example of a mission in which requirements number 2 and 3 appear simultaneously. A 1975-76 study⁶ conducted at Fairchild under ERDA sponsorship revealed how a nuclear-dynamic power system could enhance the performance of this mission with a spacecraft of inherently simpler and more practical design than the solar-powered alternative.

Most recently, missions which have all three of the listed requirements have become known. Some examples are the advanced Defense Meteorological Satellite-Block 6, Landsat-D, and the Stereosat/Mapsat, which combines an operational Landsat-type mission with a highly accurate three-dimensional earth-mapping mission. Some of the initial considerations of a nuclear-powered approach to the Landsat-type mission are described.

The ultimate selection as between a solar or a nuclear design approach for a particular mission must, of course, include such other factors as cost, reliability, lifetime, schedule and risk. The demonstration of reliability and lifetime within a reasonable cost and schedule capability is the major task remaining for DIPS. However, its unique technical capabilities make it a promising candidate for many missions of great future importance.

Global Positioning System (GPS)

The 1976-1977 Fairchild study was directed toward achieving improvements in a number of critical parameters of the existing solar-powered space vehicle. These parameters are: (1) increased power to user; (2) increased satellite lifetime; (3) improved system accuracy between satellite updates from the ground.

The last parameter is a function of two main factors: (4) the stability of the on-board frequency standard, and (5) the predictability of the satellite ephemeris into future time.

The predictability of the satellite ephemeris is, in turn, a function of the modelability of the satellite design with regard mainly to solar pressure effects.

To the extent that vehicle ephemeris drift can be modeled and predicted, the ground user will be able to compensate for this drift by the application of perturbation coefficients which will be transmitted by the satellite along with the Keplerian orbit elements. Such important disturbances as geodetic variations in the gravitational field of the earth, and the major solar-pressure effects, can be modeled and predicted with good accuracy. However, there are significant disturbances which cannot be modeled or which vary so much or so unpredictably from day to day that they will result in unpredictable changes in satellite ephemerides, and hence time-dependent degradation in system accuracy. Among the more significant of the unpredictable disturbances which apply to the existing solar-powered spacecraft are those shown in table 1 below.

TABLE 1.
UNPREDICTABLE DISTURBANCES TO SOLAR-POWERED SPACECRAFT EPHEMERIDES

NO.	OPERATING FORCE	NON-REPETITIVE CAUSAL FACTOR
1	Solar Pressure	Orientation of S/C surfaces with different α and ϵ to sun
2	Solar Pressure	Shadowing of S/C surfaces by protrusions, e.g. antennas, thrust cone
3	Solar Pressure	Variations in louver blade angles causing variable solar reflections
4	Thermal Radiation	Variations caused by independent operation of thermal control louvers
5	Impulse	Gas leakage or outgassing

The phenomena noted in items 1 and 2 above do, of course, repeat on an annual basis and only change slowly from day to day. However, the extent to which they repeat is governed by the accuracy of the control system, and the extent to which the satellite can be physically modeled.

It has been estimated by the GPS program office that these residual errors will ultimately result in an unpredicted ephemeris drift of the order of 1.25 meters/day. To this error must be added the equivalent error due to uncorrected drift of the on-board frequency standard. Three frequency standards based on different technologies are projected to be available for the GPS satellites. These are shown in table 2 with the order of magnitude of the expected drift rate of each type. Clearly, with the use of a rubidium standard, user-calculated position error would be clock-dominated, while a cesium standard would result in roughly comparable errors from both sources. Significant improvement in the stability of the cesium standard, or the introduction of the ultimate hydrogen-maser standard, however, would be incompatible with the continued use of the current solar-powered satellite.

TABLE 2.
ON-BOARD CLOCK FREQUENCY STABILITY

FREQUENCY STANDARD	FREQUENCY STABILITY $\frac{\Delta f}{f}/\text{DAY}$	RANGE ERROR BUILDUP	
		M/HR	M/DAY
Rb	10^{-12}	1.1	26
Cs	10^{-13}	0.1	2.6
H ₂ - MASER	10^{-14}	0.01	0.26

It is obvious that by eliminating the solar arrays, a nuclear spacecraft would have a positive impact on ephemeris predictability simply by reducing the projected area and hence the absolute magnitude of the total solar force. Perhaps not as obviously, however, a nuclear spacecraft can have a substantial impact on all of the disturbances listed above. Elimination of the sun-pointing requirement of the solar arrays introduces a number of design options which can substantially improve or eliminate most of all of the listed disturbances. This requirement strongly influenced the design approaches which were explored and that which was ultimately selected.

The ideal spacecraft designed for ephemeris predictability would be spherical with uniform solar reflectivity. The unique feature of the DIPS is that it enables a spacecraft design which resembles this ideal in many essential features. It also enables the mission to be performed with a slow yaw rotation of the spacecraft, to further enhance this predictability factor.

Consequently, a nuclear-powered design was selected (figure 3) which would simplify the external

Fig. 3. Nuclear GPS Spacecraft

characteristics of the space vehicle in shape, in surface properties, and in operating mode so as to improve the modelability of the spacecraft. In addition to reducing the absolute magnitude of projected area, the introduction of a nuclear power system can allow the design of the spacecraft with simple external configuration, with minimum variations in surface material, and with the possibility for rotating the spacecraft in orbit. This latter factor permits further improvement of orbit predictability, as well as certain other benefits, because of the averaging effect of the rotation.

The desire for a highly temperature-controlled environment for the frequency standards led to the selection of a highly integrated design approach in this case. Here, the electronic payload equipment is mounted within the power system radiator and is cooled by heat-pipe-assisted conduction directly to the working fluid of the power system. The mechanical and thermal design approaches are illustrated in figure 3². Despite the high order of integration achieved thereby, the equipment module and power system module are readily separable for assembly and test prior to system final assembly.

The selection of a thermally integrated design in which the skin of the spacecraft acts as the heat radiator for both the power system and all of the spacecraft electronic equipment, enables the spacecraft to be designed with all surfaces uniformly coated and acting as portions of the radiator. This design approach also leads to a steady baseplate temperature for the frequency standards, and to a spacecraft which is naturally hardened against a laser threat³.

The yaw rotation of approximately one revolution per minute is enabled by the azimuth-independent nature of the GPS signal. This slow spacecraft rotation is used to counter the rotational momentum of the DIPS rotating group, and results in a substantial reduction in the importance of the solar pressure effects to the ephemeris prediction uncertainty.

The residual uncertainty of the nuclear GPS spacecraft was analyzed to be due to three main causes shown in table 3 below.

TABLE 3.
UNPREDICTABLE DISTURBANCES TO NUCLEAR-
INTEGRATED SPACECRAFT EPHEMERIDES

NO.	OPERATING FORCE	NON-REPETITIVE CAUSAL FACTOR
1	Thermal Radiation	Mis-estimation of surface temperatures
2	Solar Pressure	S/C misalignment in principal axis
3	Impulse	Misaligned or mismatched propulsion system jets

The Phase I GPS development satellites have demonstrated the usefulness and reliability of magnetic torquing for momentum wheel unloading in the GPS spacecraft orbits, thereby eliminating error source number 3 above from consideration.

Table 4 shows the calculated effects of the remaining error sources on the knowledge of spacecraft position in both radial and transverse directions relative to the earth's center. Because user position error calculations are less sensitive to transverse spacecraft position errors by factors of 5-to-10 relative to radial spacecraft position errors, the transverse errors are divided by 5 and added to the radial errors on the bottom line.

TABLE 4
MAXIMUM ORBIT EPHEMERIDES ERRORS DUE TO
UNPREDICTED SPACECRAFT CONDITIONS

ERROR SOURCE	ERROR RATES (ft/day)	
	RADIAL	TRANSVS.
THERMAL EFFECTS	0.015	0.15
• PREDICTION ERROR IN TEMPERATURE OF EARTH-VIEWING FACE = 4°C	0.010	0.10
• PREDICTION ERROR IN TEMPERATURE OF SPACE-VIEWING FACE = 2°C	0.005	0.05
• MEAN TEMPERATURE DIFFERENCE ON OPPOSITE SIDES OF CYLINDER DUE TO HELICAL COOLING COIL = 10°C	2.8×10^{-4}	2.8×10^{-3}
SPACECRAFT MISALIGNMENT (CONICAL ROTATION MODE)	0.085	0.13
• ALIGNMENT ERROR = 0.5°, UNIFORM REFLECTIVITY	0.025	0.05
• ALIGNMENT ERROR = 0.5° NON-UNIFORM REFLECTIVITY	0.025	0.05
• ALIGNMENT ERROR = 0.5° SELF-SHADOWING EFFECT	0.04	0.08
TOTAL, ALL EFFECTS (FT/DAY)	0.08	0.28
(M/DAY)	0.02	0.08
OVERALL (M/DAY)	0.04	

Thus, the result of transitioning from a solar three-axis spacecraft to a nuclear-integrated spacecraft is to reduce spacecraft ephemerides errors by more than an order of magnitude, making this error source potentially less significant than the drift of the ultimate hydrogen maser clock. The effect on the user position error calculations, assuming a complete 24-satellite constellation, is shown in figure 4.

Fig. 4. GPS System Graceful Degradation Rate Depends on Clock Drift and Ephemeris Predictability Factors

This is illustrative of the benefits which can be realized by any space system which requires highly accurate knowledge of spacecraft position only. Although the GPS satellites are intended to be updated daily while the master control station is operational, not every space system can afford the continuous tracking, modeling, and computational resources of the GPS program. Those systems which will rely on GPS receivers for this purpose will be vulnerable to failure of these receivers, while the GPS system itself will be vulnerable to failure of the master control station, resulting in user accuracy degradation along the lines described by figure 4.

Deep Space Surveillance Satellite System

Solar-Powered Design. The Deep Space Surveillance System (DSSS) is an example of a space-object surveillance satellite which requires a highly stable platform with very precise sensor pointing control requirements. A variety of scanning and staring motions may be required of the sensor, which may operate in either the visible or long-wave infrared (LWIR) region of the spectrum. A variety of orbits is under consideration.

This mission was reviewed in a 1977 IECEC paper⁶. In its solar-powered configuration, at least three and perhaps four rotating joints are required, in addition to a very large deployed solar array (total power requirement can be as large as 4 kWe). In schematic configuration the solar-powered design of this spacecraft will follow the format shown in figure 5.

Although the structural dynamics of this spacecraft design has not yet been studied in depth, a design with so many rotating joints and sources of structural excitation is inimical to the achievement of the required level of spacecraft stability (≤ 2 arcsec/sec) and fine pointing control ($\sim$ arcsecs). (See next section on Landsat D). A more recent study of

Fig. 5. Solar-Powered Spacecraft; Basic Configuration (Schematic)

the thermal control requirements of this spacecraft in its LWIR version⁶, illustrated and verified the required solar-powered spacecraft design approach (fig 6). Figure 6a follows the schematic layout of figure 5 precisely. Figure 6b illustrates the contractor's conclusion that in a low altitude orbit the (deployable) refrigerator radiator must be mounted on the despun section of the spacecraft, necessitating the development of a fluid swivel joint, to duct refrigerator coolant across the rotating joint. In the high altitude version, a flexible coolant line is required, as depicted in figure 5.

a. HIGH ALTITUDE EQUATORIAL

b. LOW ALTITUDE EQUATORIAL

Fig. 6. DSSS Designs; Vought Corp. (Reference 7)

Nuclear-Powered Design. In the nuclear-powered design approach, both sections of the spacecraft are mounted together without intervening rotating joints. The sensor is similarly mounted in a fixed position within the spacecraft (figure 7). This is the so-called "monolithic" design approach in which all rotating joints are eliminated. In this case the entire spacecraft performs the required motions, controlled by a small attitude control package mounted to a lower platform. This platform is oriented at all times

perpendicular to the earth's nadir and is hinged along one edge to accomplish this orientation. The hinge angle is varied, however, only when the elevation angle is to be changed and is then locked into its new position. Both attitude control equipment and earth communications antenna are mounted onto this surface. Alternatively, this mission can be performed with Control-Moment Gyros for attitude control.

Fig. 7. Nuclear-Powered Hysat Spacecraft

In reference 6, this design approach was shown to be capable of reducing spacecraft electric power requirements by some 50%, and reducing spacecraft weight by some 20%. Its superior structural dynamics and stability characteristics compared to the multi-jointed solar design are evident qualitatively although neither design has been analyzed to date. Table 5 summarizes the advantages of the nuclear DSSS design.

TABLE 5
ADVANTAGES OF NUCLEAR SPACECRAFT FOR DSSS

- FLIES IN ANY ORBIT WITHOUT MODIFICATION
- PERFORMS CONTINUOUS SCAN, STARE, OR OTHER MOTIONS
- INSENSITIVE TO NORMAL OR ENHANCED VAN ALLEN RADIATION
- NUCLEAR & LASER HARD
- REDUCED VISIBILITY TO OPTICAL OR RADAR SEARCH
- MANEUVERABLE WITH MINIMAL WARNING TIME
- MONOLITHIC SPACECRAFT DESIGN
 - ELIMINATE ROTATING ASSEMBLIES
 - ELIMINATES SENSOR GIMBALLING
 - ELIMINATES FLEX FLUID LINES, FLUID SWIVEL JOINTS
 - NO SOLAR ARRAY DEPLOYMENT
 - NO RADIATOR DEPLOYMENT
 - NO CG SHIFTS RESULTING FROM SENSOR MOTION
- NO BATTERIES REQUIRED FOR ECLIPSE OPERATION
- NO THERMAL STORAGE REQUIRED FOR REFRIGERATOR OPERATION DURING ECLIPSE

DMSP-Block 6/LANDSAT-D/MAPSAT

These three missions combine the three stringent requirements for highly accurate position knowledge, attitude control, and sensor stability which are likely to become typical of future optical-observation missions. Of these, only the Landsat-D is a currently approved mission, now scheduled for launch in late 1982. Mapsat, now in the conceptual design study phase, will be similar to Landsat-D in most spacecraft requirements. The Landsat-D, therefore may be a paradigm for future missions of this type (figure 8).

Fig. 8. LANDSAT-D Spacecraft

In order to achieve the required optical performance, the Landsat spacecraft error budget in the three parameters of interest has been established as shown in table 6:

TABLE 6
LANDSAT-D RESIDUAL ERROR BUDGET

SOURCE OF ERROR	ERROR (1 σ)	
	m	sec
EPHEMERIS	2.8	-
ATTITUDE	2.8	0.8
SPACECRAFT VIBRATION	1.0	0.3
OTHERS	5.0	-

The ephemeris and attitude requirements are not expected to be achieved by spacecraft equipment alone, despite the use of a GPS receiver on-board the spacecraft for the former. Fortunately, these two requirements can be finessed in the Landsat case by a posteriori analysis of the transmitted image data and by matching this data to known ground features. In addition, these requirements may ultimately be satisfied autonomously by only modest improvements in the on-board inertial control system and by long-range improvements in the GPS accuracy, along the lines previously indicated.

The problem of spacecraft vibration is more insidious, however, and more difficult to solve within the context of a solar-powered design. Although the problem is rarely, if ever, identified with the use of solar power, the solar arrays are implicated both directly and indirectly.

The cause of spacecraft vibration is the dynamic interaction of on-board moving elements, of which the solar array drive is one, with a flexible-body spacecraft. The major sources of spacecraft flexibility are the solar arrays and the TDRS antenna boom.

The latter in particular has been identified as a major source of dynamic influence on the spacecraft, caused in large part by locating a relatively heavy component (6-foot-diameter antenna plus electronics) at the end of a 12½-foot-long boom. However, the

length of the boom was determined by the need to prevent blockage of the antenna beam (5-degree clear field-of-view) by the solar array. The geometric situation is seen more clearly in figure 9, with the elevation gimbal angle of the antenna shown (Azimuth angle = 400 degrees). Despite the extension of the antenna some blockage of the beam about the midnight array position is experienced. Such extreme gimbal angles are only required during northern hemisphere winter periods, so that the overall data loss to the mission is considered to be acceptable at this point. Clearly, however, a shorter boom would result in unacceptable daily blockage of the TDRS antenna beam, which is the main communications link of the spacecraft.

Fig. 9. LANDSAT-D Geometry

Thus, the solar array contributes to spacecraft dynamic flexibility both directly and indirectly, by its own presence and by dictating the need for a long antenna boom. Although a nuclear-powered version of Landsat has not been designed, an approach similar to the monolithic design of the DSSS shown previously would result in a more rigid spacecraft, less subject to potentially damaging and image-degrading internal vibration. The potential advantages of a DIPS-powered Landsat, presumably applying as well to Mapsat, are summarized in table 7:

TABLE 7
ANTICIPATED ADVANTAGES
OF NUCLEAR-POWERED LANDSAT/MAPSAT

- Reduces Spacecraft Vibration
- Eliminates TDRS Antenna Blockage by Solar Arrays
- Eliminates Thruster Misalignment Due To Deployments
- Eliminates Safe-hold Mode Reorientation Requirements
- Allows Near-total Freedom in Spacecraft Orientation for Thrusting

Summary

The use of nuclear power results in the elimination of the sun-tracking or orientation requirement, and the elimination or reduction of major deployment requirements. This leads to the unique features of nuclear power which have profound consequences for high-requirements missions, by enabling spacecraft design to focus more directly on mission requirements without the need for also satisfying the difficult solar array requirements. These features and consequences are summarized in table 8:

TABLE 8
KEY FEATURES OF DIPS-POWERED SPACECRAFT

MISSION	KEY FEATURES	KEY CONSEQUENCES
Multi-Mission Spacecraft	<ul style="list-style-type: none"> • Independence of Sun Angle • Pumped coolant loop 	<ul style="list-style-type: none"> • Universal orbit/altitude capability • Power plus thermal control for payload are standard functions
NAVSTAR/GPS	<ul style="list-style-type: none"> • Pumped coolant loop • Pseudo-spherical design 	<ul style="list-style-type: none"> • Improved ephemeris predictability • Excellent thermal control
DSSS	<ul style="list-style-type: none"> • Elimination of sun tracking 	<ul style="list-style-type: none"> • Monolithic spacecraft • Improved stability • Simplified design • Lower weight & power
Landsat/Mapsat	<ul style="list-style-type: none"> • No deployed array • Short TDRS antenna boom 	<ul style="list-style-type: none"> • Improved vibration response • No TDRS antenna blockage

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