SUMMARY OF APPLICATION STUDIES FOR
NUCLEAR-DYNAMIC POWER SYSTEMS

Bernard Raab

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Since 1972, Fairchild has conducted a number of applications studies of nuclear-radioisotope dynamic power systems to Earth-orbiting missions. These studies have revealed various areas of performance and/or economic benefit of these power systems vis-a-vis the more common solar-array/battery power systems. Three of these studies are summarized herein. These are:

1. Global Positioning System (GPS)
2. Deep Space Surveillance System (DSSS)
3. Commercial Communications Satellites

Taken together, the studies illustrate, in three missions of totally differing requirements and constraints, that the comparisons of nuclear and solar power should not be drawn on the basis of cost and weight of the power systems themselves, but rather on the basis of overall mission performance and cost. Nuclear power allows the utilization of innovative design approaches which, in many cases, are not available to solar-powered designs, and which can result in substantial performance, or even economic benefits, to the mission as a whole.

Detailed reports of these studies are available upon request from the Department of Energy or Fairchild Space & Electronics Company, Germantown, Maryland.
1. GLOBAL POSITIONING SYSTEM

1.1 INTRODUCTION

The current phase 1 GPS navigation development satellites (p1 NDS) are intended as proof-of-principle units, which will enable experimenters to validate many of the assumptions of the GPS system. For the operational phase of the program, improvements are desired in a number of spacecraft performance areas.

Earlier work by Fairchild under ERDA contract had suggested that a nuclear-powered spacecraft, using a radioisotope-dynamic power system, could offer significantly more satellite-transmitted power in the critical navigation links. This would provide more anti-jam margin, and enough signal power to penetrate dense foliage or conditions of high atmospheric absorption. Moreover, because the GPS orbits are in the tail of the Van-Allen belts, solar array degradation can limit useful satellite lifetime to a few (e.g., 4-5) years. Nuclear power systems are, of course, not subject to degradation from this source. Because as many as 24 satellites are ultimately required in the operational phase of the system, extension of expected satellite lifetime by a small amount (e.g., 1-2 years) can have substantial economic repercussions on satellite replacement rates.

Nuclear-dynamic power systems use radioisotope heat sources. The heat sources decay to half power in 87 years. The power systems convert this heat to electricity in miniature turbine-generator sets which operate at fixed speed on gas-film bearings. Once started, power degradation is very slow; closely following heat source decay. Unlike solid-state thermoelectric or solar-cell converters, long-term material property changes cannot effect the output power of the generators. Consequently, long-term
fairly constant output power is a reasonable expectation for these systems. After 10 years, 92% of original thermal power is still being produced. (See recent announcement by NASA-Lewis for Brayton cycle generator, marking 40,000 hours of operation.)

Because of these and other considerations, Fairchild was asked to study an advanced GPS satellite design using nuclear-dynamic power. This study, jointly funded by SAMSO/YE and ERDA/NRA, was conducted from September 1976 through June 1977. The final report was issued in August 1977.
1.2 SPACECRAFT INTEGRATION CHARACTERISTICS

A study was conducted to determine the relative value of nuclear power for the Global Positioning System (GPS) space vehicle. The study was directed toward attempted 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: The first is related to the stability of the on-board frequency standard. The second is related to 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. Consequently, a nuclear-powered design was selected which would simplify the external characteristics of the space vehicle in shape, in surface properties, and in operating mode so as to improve the modelability of the spacecraft. Elimination of the solar array panels alone will result in improved modeling by reducing the projected area facing the sun. This reduces the absolute magnitude of the solar-pressure force, and hence the errors in estimating this force. 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. Figure 1.1 shows the nuclear spacecraft design which was evolved in the study.
A spacecraft of this type would have the power-to-user characteristics shown in Table 1.2 as compared to the current solar-powered GPS spacecraft. Table 1.2 also shows the expected improvements in mean-mission-duration which should be achievable by the nuclear power designs based on the reliability and lifetime figures now projected for these systems. Also shown are the projected improvements in the other critical parameters.

Table 1.2. Summary of Improvements in Critical Parameters

<table>
<thead>
<tr>
<th>PARAMETER</th>
<th>N/GPS</th>
<th>Φ1 NDS</th>
<th>Δ</th>
</tr>
</thead>
<tbody>
<tr>
<td>1. POWER TO USER, dBw</td>
<td>L1: -154</td>
<td>-160</td>
<td>+6</td>
</tr>
<tr>
<td></td>
<td>L2: -163</td>
<td>-166</td>
<td>+3</td>
</tr>
<tr>
<td>2. MEAN MISSION DURATION, YRS.</td>
<td>5.5+</td>
<td>4.5</td>
<td>+1+</td>
</tr>
<tr>
<td>3. ORBIT PREDICTABILITY, M/DAY</td>
<td>0.03-0.3</td>
<td>2.5</td>
<td>+ x10+</td>
</tr>
<tr>
<td>4. MAXIMUM CLOCK TEMPERATURE VARIATION, °C/ORBIT</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>WITHOUT CONTROL</td>
<td>2</td>
<td>2</td>
<td>-</td>
</tr>
<tr>
<td>WITH HEATER CONTROL</td>
<td>0.2</td>
<td>-</td>
<td>+ x10</td>
</tr>
<tr>
<td>5. VULNERABILITY, JCS</td>
<td>1+</td>
<td>0.01</td>
<td>N/A</td>
</tr>
</tbody>
</table>

*POWER SYSTEM ONLY
Figure 1.3 shows the expected degradation of user position accuracy as a function of time from last update for both solar and nuclear satellites. The solar design is shown with both cesium and hydrogen maser frequency standards. The latter is considered to be the ultimate standard for the operational phase of the GPS system. Figure 1.3 illustrates that the full benefits of the more accurate hydrogen maser clock could not be realized with the existing solar powered spacecraft because of its ephemeris predictability limitation. When combined with a nuclear-powered spacecraft of the recommended design, however, the full benefits of both the stable frequency standard and the highly predictable orbital motion of the spacecraft can result in an overall system accuracy which degrades at an extremely slow rate for long periods after ground update.

Figure 1.3

NUCLEAR-POWERED SATELLITE IMPROVES GPS SYSTEM
ACCURACY AND SURVIVABILITY BY IMPROVING EPHEMERIS PREDICTABILITY

GPS-NAVSTAR

MINIMUM USER POSITION ERROR (METERS)

DAYS FROM LAST GROUND UPDATE
2. DEEP SPACE SURVEILLANCE SYSTEM

The Deep Space Surveillance Satellite System (DSSS) is an example of an Earth-surveillance satellite in a non-equatorial orbit with observation duties either Earth-centered or space-directed. The Earth-centered observation task is illustrated in Figure 2.1, which shows the normal scan mode of the spacecraft.

Figure 2.1
In order to achieve the required motions of the sensor, as well as to direct solar array panels toward the sun, a two-body satellite design of the type shown schematically in Figure 2.2 is required. The lower body rotates about an axis which is directed toward the Earth's center. The sensor is mounted on a gimbal platform within the lower body so as to allow changes in elevation angle to be achieved. The rotation of the entire platform provides the scan pattern for the sensor. The upper body is despun and contains the attitude control and other equipment. The independent rotation of the upper body together with the single axis rotation of the solar arrays allows the solar arrays to be oriented to the sun. In certain cases an additional rotation of the array is required if an extra pointing requirement is imposed on the upper body.

Figure 2.2
Despite three and possibly four rotating joints, the sensor platform is required to have high stability and low rates of change. In the long-wave infrared approach to this mission the sensor requires a cryogenic refrigerator to cool the detector elements. This refrigerator is shown mounted to the sensor in Figure 2.2 with flexible fluid lines connecting to the refrigerator radiator which is required for waste heat rejection. The refrigerator typically is operated by a heat engine which requires a substantial amount of thermal input and results in a somewhat greater waste heat output. In the solar-powered version the heat input is provided by electric heaters driven by the solar-array/battery system.

Two modifications of this design have been studied in a nuclear powered version. The first is illustrated in Figure 2.3, in which the electric heaters which derive their power from the deployed solar arrays are replaced by radioisotope heaters. This results in substantial savings in weight and size, as illustrated in Figure 2.3. The second modification involves replacement of the entire solar array/battery package with a nuclear power system, as illustrated in Figure 2.4. The nuclear power system is shown mounted above the payload section of the spacecraft; the latter contains all of the sensor and other payload equipment mounted within the waste heat radiator of the refrigerator.
Figure 2.3

COMPARISON OF POWER OPTIONS FOR VM REFRIGERATOR

Figure 2.4

NUCLEAR-POWERED HYSAT SPACECRAFT
In the nuclear-powered design both sections of the spacecraft are mounted together without intervening rotating joints. The sensor is similarly mounted in a fixed position within the spacecraft. 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 antennae are mounted onto this surface. Figure 2.5 shows how this entire spacecraft can fit within a typical Delta launch shroud with a six-foot dynamic envelope diameter. Table 2.6 shows the relative power requirements of both the solar base-line design as well as the nuclear power alternative. Table 2.7 shows the weight comparisons between the solar baseline design, the same design using more advanced solar arrays and batteries (1985 technology) as well as the nuclear-powered version. The advantages of the all-nuclear spacecraft design are summarized in Table 2.8, and pictorially illustrated in Figures 2.9.
Figure 2.5

ALL-NUCLEAR HYSAT SPACECRAFT IN DELTA BOOSTER

Table 2.6

POWER REQUIREMENTS -- 11 YSAT SPACECRAFT

<table>
<thead>
<tr>
<th>SUBSYSTEM</th>
<th>BASELINE</th>
<th>NUCLEAR-POWERED</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>SOLAR-POWERED</td>
<td></td>
</tr>
<tr>
<td></td>
<td>We</td>
<td>We</td>
</tr>
<tr>
<td>Payload</td>
<td>1735</td>
<td>735</td>
</tr>
<tr>
<td>TT&amp;C</td>
<td>181</td>
<td>181</td>
</tr>
<tr>
<td>POWER</td>
<td>96</td>
<td>100</td>
</tr>
<tr>
<td>ATTITUDE CONTROL</td>
<td>118</td>
<td>100</td>
</tr>
<tr>
<td>PROPULSION</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>THERMAL CONTROL</td>
<td>10</td>
<td>10</td>
</tr>
<tr>
<td><strong>TOTALS</strong></td>
<td><strong>2440</strong></td>
<td><strong>1126</strong></td>
</tr>
</tbody>
</table>

1126 W = 1000 We
Table 2.7

WEIGHT COMPARISONS--LBS

<table>
<thead>
<tr>
<th>SUBSYSTEM</th>
<th>BASELINE SOLAR</th>
<th>ADVANCED SOLAR</th>
<th>NUCLEAR</th>
<th>BASIS OF ESTIMATE</th>
</tr>
</thead>
<tbody>
<tr>
<td>PAYLOAD</td>
<td>380</td>
<td>580</td>
<td>430</td>
<td>GIANTAL + REACT. WHEEL, 150 LBS</td>
</tr>
<tr>
<td>TT &amp; C</td>
<td>95</td>
<td>95</td>
<td>95</td>
<td>NO CHANGES</td>
</tr>
<tr>
<td>PROPULSION</td>
<td>84</td>
<td>86</td>
<td>86</td>
<td>SEPARATE ACS DESIGN</td>
</tr>
<tr>
<td>THERMAL CONTROL</td>
<td>35</td>
<td>35</td>
<td>35</td>
<td></td>
</tr>
<tr>
<td>ATTITUDE CONTROL</td>
<td>227</td>
<td>227</td>
<td>143</td>
<td></td>
</tr>
<tr>
<td>POWER</td>
<td>464</td>
<td>464</td>
<td>417</td>
<td>NUCL. DYN. GEN. @ 35 W/LB + 50 LBS</td>
</tr>
<tr>
<td>SOLAR</td>
<td>285</td>
<td>285</td>
<td>-</td>
<td>FIXED FRACTION</td>
</tr>
<tr>
<td>NUCLEAR ELECT.</td>
<td>-</td>
<td>-</td>
<td>372</td>
<td></td>
</tr>
<tr>
<td>NUCLEAR HEAT</td>
<td>150</td>
<td>150</td>
<td>122</td>
<td></td>
</tr>
<tr>
<td>STRUCTURE</td>
<td>179 (11%)</td>
<td>179 (11%)</td>
<td>311 (19%)</td>
<td></td>
</tr>
</tbody>
</table>

*IF ONE FRUSA IS ELIMINATED

Table 2.8

ADVANTAGES OF ALL-NUCLEAR SPACECRAFT FOR HYSAT

- MONOLITHIC SPACECRAFT
- NO ROTATING JOINTS
- NO FLEXIBLE FLUID LINES
- NO FLEXIBLE SOLAR ARRAYS & ARRAY DEPLOYMENT
- NO C-G SHIFTS
- NO BATTERIES
- STATE-OF-THE-ART ATTITUDE CONTROL WITH ATTITUDE AND RATE KNOWLEDGE TO ONE-TENTH OF LIMITS
- WEIGHT REDUCTION > 15 - 20%
- LOWER COST BOOSTER
- NOT AFFECTED BY ORBITS CROSSING TRAPPED-RADIATION BELTS
- LOWER VISIBILITY TO OPTICAL OR RADAR SEARCH
- EASILY MANEUVERABLE
Figure 2.9

HOW NUCLEAR POWER CAN SIMPLIFY SPACECRAFT DESIGN

NUCLEAR-POWERED DESIGN:
- Eliminates rotating joints & moving mechanical assemblies
- Reduces size & weight
- Compact design is readily maneuverable

INFRARED SURVEILLANCE MISSION

SOLAR-POWERED DESIGN

NUCLEAR DYNAMIC POWER SYSTEM
3. COMMERCIAL COMMUNICATIONS SATELLITES

A study was conducted of the possible applicability of radioisotope power to commercial communications satellites. These satellites, confined to synchronous equatorial orbit, have tended to be mainly cost- and weight-sensitive spacecraft designed to fit within the most efficient boosters. From the mid-1960's through the mid-1970's these have generally been based on the dual-spin concept, and were usually designed to fit on one of the Thor/Delta family of boosters. Figure 3.1 is a pictorial history of such satellites through the 1971-launched Intelsat 4. All models prior to Intelsat 4 were launched on one of the Delta boosters. The latter was designed for launch on the Atlas-Centaur with an approximate doubling in size, weight, and cost of both spacecraft and booster. Most commercial (i.e., non-Intelsat, non-AT&T) operators, however, prefer to remain within the lower cost Delta family and chose instead to proceed with a new design approach; i.e., three-axis-stabilized spacecraft.

Figure 3.1

SPIN STABILIZED COMMUNICATIONS SATELLITES

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Primary Power W</td>
<td>40</td>
<td>75</td>
<td>120</td>
<td>280</td>
<td>600</td>
</tr>
<tr>
<td>Simplex Circuits</td>
<td>480</td>
<td>480</td>
<td>2400</td>
<td>12,000</td>
<td>12,000</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>40,000</td>
</tr>
</tbody>
</table>
Although the weight capability of the Delta family has been increased to handle more payload than the Anik/Westar-type satellite, this increased payload capability could not be fully utilized by a spin-stabilized design because of the volume limitations inherent in the Delta booster. These volume limitations provide a maximum practical size for the body-mounted solar array of the spinner design, limiting the power output of such designs to the 300-400 watt(e) range.

This basic power limitation of the Delta booster could be obviated by the use of a radioisotope dynamic power system, while still retaining the low cost and well-proven advantages of the spinner design approach. This is also illustrated in Figure 3.1, where it is seen that a nuclear-powered spinner can be packaged in a volume small enough to allow incorporation of both C- and Ku-band dual-polarized antennas while providing approximately 700 watts of electrical power. This substantial increase in both bandwidth and payload capability within the low-cost Delta-class booster can result in an ultimately lower cost per satellite circuit despite the higher cost of the nuclear power system itself. Figure 3.2 shows the nuclear spinner design in somewhat greater detail. Figure 3.3 compares the payload capability of the two competing solar power approaches with the reference-design nuclear spinner, but also the increased payload capability introduced by a nuclear spinner with a power system of 3.5 watts(e)/lb.

Both of the solar system curves assume 1985 technology, while the advanced nuclear technology is that projected for the 1981 time period. Thus, although a nuclear system of 3.5 watts(e)/lb. is compared to a solar-array system of some 8 watts(e)/lb., the overall weight advantage of the nuclear spinner results in a greater payload weight-power capability. If
the maximum number of satellite circuits is designed into each of these reference cases based on the capabilities projected in Figure 3.3, cost per satellite circuit is shown in Figure 3.4. This illustrates that the achievement of low weight nuclear systems projected for 1985 and beyond (4-6 watts(e)/lb.) can result in substantial cost savings in a commercial communication satellite when compared to advanced array/battery technology.
Figure 3.3

COMMUNICATIONS SATELLITES PAYLOAD* CAPABILITIES FOR FIXED WEIGHT & VOLUME LIMITS (DELTA CLASS)

PAYLOAD MASS (LBS)

PAYLOAD POWER, WATTS

Figure 3.4

CASE 2. COST-BENEFIT COMPARISONS (FUEL COST = 600 $/Wt)
Although it is recognized that future commercial communication satellites may no longer be launched on expendable boosters such as the Thor/Delta, weight and volume limitations will still serve to constrain satellite designs in order to minimize launch costs. Under these circumstances, innovative design approaches can result in purely economic advantages for a nuclear-powered spacecraft despite the higher cost per watt(e) or weight per watt(e) of the nuclear systems.