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A Nuclear-Powered Communications Satellite for the 1980's

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Introduction of the Space Transportation System and the potential new requirements for communications satellites in the 1980's provide renewed incentives to examine carefully the role of nuclear power for geostationary repeaters. The results of mating developing isotope power supply technology to communications missions in the next decade are presented here. The Shuttle will accommodate larger, more powerful spacecraft, and commercial users will need the associated increases in capacity. A preliminary design of a nuclear powered, standardized satellite is offered. A system of five units is conceived, each has a capacity of 80,000 one-way voice channels. Ready for operation by the mid-1980's, this design will make optimum use of the Shuttle pricing scheme. Each satellite has its own staging propulsion and is envisioned as a cylinder of 4.1-m diameter and 3.4-m height with a liftoff mass of 6423 kg. On-station pointing accuracies of 0.09 deg in pitch and roll, and 0.4 deg in yaw are easily achieved with a bias momentum attitude control system. Initial power available is 2 kW$_e$ via a dynamic isotope system.

Nomenclature

h = value of nominal bias momentum of the attitude control system
I_{sp} = specific impulse
T_z = yaw disturbance torque
ΔV_{equiv} = annual velocity increment for dumping momentum
$\Delta V_{E/W}$ = annual velocity increment associated with East-West stationkeeping
$\Delta\lambda$ = change of longitude position of a geostationary satellite
ψ_{ss} = maximum allowable steady-state yaw offset of attitude pointing system
ω_0 = orbital rate of geostationary orbit, 7.29×10^{-5} rad/s

Introduction

IN 1980, the first orbital flight test of the Space Shuttle will lift off from Pad 39A at Kennedy Space Center. This will be a prelude to the opening of the Space Transportation Era, which will begin with the first operational flights in 1981. New customer requirements, advances in technology, and increased launch capacity force satellite designers to re-examine alternatives in areas such as power systems. A standardized, nuclear powered communications satellite design is offered here for the 1980's. Its mission is to provide global services into the 1990's.

The projected growth of traffic for the 1980's indicates larger and more powerful communications satellites for commercial users. The Shuttle-launched, standard bus presented here has been conceived to fulfill these needs. A system of five satellites, each capable of 80,000 one-way voice channels has been chosen to satisfy the projected demands of the mid-1980's. The proposed configuration is illustrated in Fig. 1. It incorporates a large Earth-pointing face for mounting a variety of high-gain antennas. This design has a 4.1-m diameter and a length of 3.4 m with antennas stowed. The combination of a cylindrical shape and a mass of 6423 kg takes optimum advantage of the launch pricing scheme.[1]

After insertion into a low parking orbit by the Shuttle, geostationary orbit is achieved via an integral Agena liquid rocket engine. The restart capability of the Agena will allow it to provide both the perigee and apogee velocity increments. During ascent, the spacecraft will be spin-stabilized. Final on-orbit attitude control is achieved with a double gimbaled momentum wheel providing pointing accuracies of 0.4 deg in yaw and 0.09 deg in pitch and roll.[2] The nominal momentum magnitude is 271 N-m-s with an operational spacecraft mass of 1497 kg. Nearly 2 kW$_e$ is provided by a dynamic isotope power system on a continuous basis at the beginning of mission (BOM). Satellite design life is 10 years. Figure 2 is a cutaway sketch showing internal components and subsystems.

The U.S. Department of Energy is currently developing the Kilowatt Isotope Power System (KIPS) for applications in this power range. Thus, this type of system is assumed for the satellite considered here. The intent here is to present a detailed preliminary design of a communications bus, and, thus, to identify and explain potential technical realities of an isotope-powered repeater for the upcoming decade of geostationary payloads. No attempt has been included to compare costs quantitatively.

An itemized mass breakdown of the spacecraft is provided in Table 1. These estimates are based on subsystem and component comparisons with satellites of similar masses and missions. Power system mass is based on extrapolation of KIPS design parameters.[3]

Primary Propulsion Subsystem

Each satellite is placed into a low parking orbit by the Shuttle, much like the procedure with Deltas and Atlas/Centaurs. However, beyond this, the spacecraft must provide propulsion or use expendable upper stages to achieve synchronous orbit. The Agena 8096 engine was chosen as the primary propulsion unit in this standardized bus, and it is usable for both the perigee and apogee burns. Restart capability and a demonstrated reliability of 0.998 makes the Agena a good candidate for this type of mission. This pump-fed engine uses storable hypergolic propellants (UDMH and RFNA) to produce a thrust of 71,170 N. A velocity increment of 2.45 km/s is required for the perigee burn and one of 1.84 km/s for the apogee burn, assuming an inclination change of 28.5 deg. Total burn time required is about 191 s, well under

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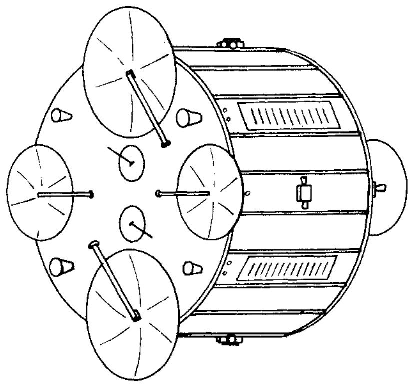

Fig. 1 Conceptual design of isotope-powered satellite.

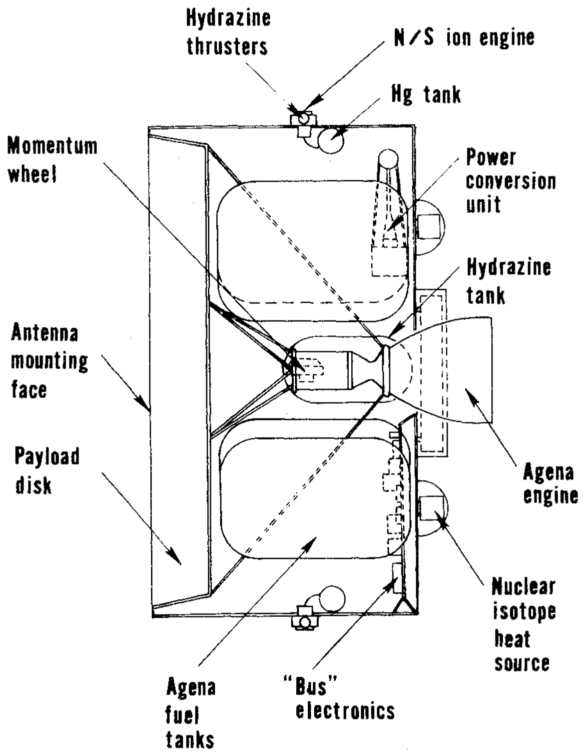

Fig. 2 Component locations.

the 240 s at which Agena is rated. The specific impulse is rated at 300 s. Total propellant mass requirement to place the 1497-kg spacecraft into synchronous orbit is 4926 kg. Hydrazine and mercury ion thrusters provide stationkeeping and will be discussed later.

Power Supply

Electrical power will be provided for this standardized bus by a dynamic isotope power system. A space nuclear power supply may, however, have one of several configurations and sizes, depending on the power requirements and type of mission.[4] Very low levels are available through radioisotope thermoelectric generators (RTG's). These are completely solid-state devices incorporating thermal-electric energy conversion. Power ranges for RTG's run from just a few watts to about 500 W_e. The next range of power, from about 500 W_e up to 3 kW_e, can be provided by dynamic systems. These devices use isotopes for heat generation, as do RTG's. However, this energy is used to run a turbogenerator. Power levels of tens to hundreds of kilowatts would require a nuclear space reactor power plant.

Two dynamic isotope systems had been under consideration until recently. Figure 3 illustrates the Brayton Isotope Power

Table 1 Estimated mass breakdown of standardized spacecraft

System or component	Mass (kg)
Agena engine (dry)	133
Nuclear power system	285
Payload[a]	408
Attitude control/stationkeeping	182
Agena propellant tanks	152
Structure	232
Telemetry, tracking, command	26
Momentum wheels, associated hardware, electronics	53
Thermal control	26
Total on-orbit	1497
Propellant	4926
Total launch weight	6423

[a] Includes power control and distribution hardware.

System (BIPS) and the Kilowatt Isotope Power System (KIPS). Ground demonstration units for each approach were developed, but only the KIPS will be pursued for flight applications. A flight experiment is planned for the early 1980's to qualify this device for further applications. The key objectives are to provide 1.3 kW_e at a specific power of 6.4 W_e/kg with an efficiency of 20-25%. A lifetime of 5-7 years is anticipated for this first flight unit. By the mid-1980's these units could provide power to the 3-kW_e level at improved efficiency, specific power, and lifetimes. A power system cost of \$3000-5000/W_e seems reasonable for this time frame.

Radiation emissions and thermal control requirements for Shuttle launches have been reviewed for isotope power supplies.[5] Since Shuttle payloads must be thermally independent, a thermal control system (TCS) is required for isotope power supplies for liftoff to separation. The TCS carries waste heat away and dumps it overboard via a water evaporation system. A direct fluid cooling method employs coils integrated into the heat sources. Thus, the radiator does not receive heat until payload deployment.

Either of these systems can be boosted to produce 2 kW_e of power by making certain internal changes to the 1.3-kW_e design. For example, KIPS output is increased by changing the jet condensor orifice. A cylindrical radiator is located around the outer shell of the spacecraft. This brings the power supply mass to 285 kg.

Attitude Control

The standardized bus is envisioned as using three-axis stabilization for on-orbit operations. The heart of the attitude control system is a double gimbaled momentum wheel with nominal momentum along the pitch axis. Roll and yaw control torques are generated by two gimbal actuators. Pitch control is achieved by changing the wheel speed. Antenna pointing accuracies of 0.4 deg in yaw and 0.09 deg in roll and pitch are specified as mission objectives. The value of nominal momentum h is determined by the maximum allowable yaw offset and is found from the relation[2]

$$\psi_{ss} = T_z / \omega_0 h$$

The maximum yaw disturbance anticipated is based on a 0.5-deg thrust misalignment of the inclination control ion thrusters. The torque of 7.98×10^{-5} N-m produced from such a misalignment requires that the bias momentum be greater than 157 N-m-s. A value of 271 N-m-s is chosen to insure that the pointing accuracy requirement is maintained. A backup wheel is also provided on the spacecraft for redundancy. Small hydrazine thrusters are located on the aft end of the spacecraft for reorientation and acquisition maneuvers. These

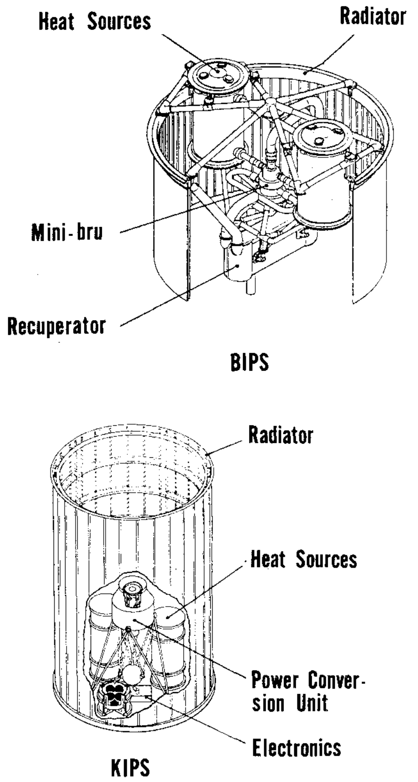

Fig. 3 Candidate dynamic isotope systems.

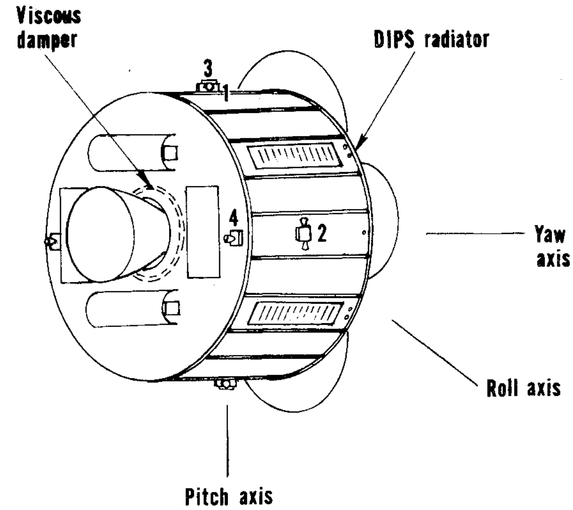

Fig. 4 Thruster locations.

thrusters also provide a means for momentum dumping when the wheel becomes saturated.

The rotating parts of the turbine-alternator unit create a momentum of 18.3 N-m-s when operating at nominal speed. This must be taken into account in the attitude dynamics of the spacecraft, especially during the spinning ascent, when its effects must be cancelled in some manner. Since the axis of rotation of these parts is aligned with the axis of rotation of the momentum wheel, momentum effects of the rotating KIPS hardware can be handled quite easily. During launch and ascent, the momentum wheel will be spun up in the opposite direction and to the same magnitude of momentum as the KIPS rotating hardware. The net pitch momentum would be zero while the spacecraft is spinning about its longitudinal axis. Once on-orbit the KIPS unit will provide some of the bias momentum for the spacecraft.

One area of concern associated with dynamic systems is the effect of torques produced as the turbine-alternator rotor deviates from its nominal speed. For example, the turbine-alternator nominally operates at a rotational speed of 33,700 rpm. A 0.05% deviation from this speed is expected due to on/off power control devices while a 1% day-to-day overall speed variation can be expected due to sink temperature variations and heat source variations. This 1% variation implies a momentum change of 0.18 N-m-s. The associated torques are countered by momentum wheel speed changes.

One important difference between expendable launches and the Shuttle is that the payload will not be injected into apogee transfer by the booster. The payload "gets off" in parking orbit. Thus, the ascent is initiated by the spacecraft. One possible sequence of events is offered here. After orbit injection the payload bay doors are opened. Orbiter orientation is established to satisfy spacecraft axis alignment for spinup. Ejection is accomplished with some angular momentum imparted by the release mechanism for initial gyroscopic stiffness. After a safe drift distance is achieved, spinup rockets raise the momentum to the required level for ascent. Attitude is accurately determined and corrections commanded from the Orbiter. The perigee injection burn is performed as the equator is passed. During ascent to apogee, a reorientation maneuver is carried out to prepare for the apogee burn. A ground commanded signal begins the second Agena firing to inject into geostationary orbit. This is followed by attitude acquisition and station adjustment.

The attitude control system includes Earth and Sun sensors for ascent attitude determination and maneuvering. Telemetry output of these devices is used on the ground or in the Orbiter to aim the spacecraft accurately for perigee injection. During transfer to apogee these data are processed on the round and commands are transmitted up to carry out the reorientation maneuver for apogee injection. Once on orbit, attitude control is automatic through the use of an Earth sensor and gimbal logic. The cylindrical shapes and propulsion system selection insure spin stability from Shuttle release to final orbit. A viscous damper for dissipating nutational energy is included on the spacecraft.

Thermal Control

The introduction of dynamic isotope power systems brings with it a unique capacity to control the spacecraft thermally. Properly designed, the temperature control system can cool or heat specific equipment items. The working fluid of the power system can be used to carry heat to or from internal components. If heat is to be rejected, this fluid absorbs heat inside the spacecraft and carries it to the radiators. Thus, many of the constraints now imposed upon the solar powered satellites can be eliminated.[6]

The communications satellite proposed here uses a combination of active and passive thermal control techniques. On the exterior surfaces silver teflon foil will minimize the thermal changes during the transition periods to and from an eclipse. Internally, multilayer insulation will be used throughout where necessary. Constant-conductance heat pipes with bimetallic louver-modulated radiators will also be used. Although this system may be heavier than one using passive control, it offers greater flexibility to accommodate changes in payload thermal dissipation. Heat pipe connections will be available for joining to the payload disk so thermal control of this equipment can be maintained.

Stationkeeping

Stationkeeping tasks will be divided between hydrazine and mercury ion thruster systems. The hydrazine system is composed of eight 4.4-N and two 8.9-N engines positioned as shown in Fig. 4. The function and size of each are listed in Table 2. This monopropellant system will be responsible for spinup and spin-down, East-West stationkeeping, and momentum wheel unloading. It will be capable of limited

Table 2 Thruster sizes and functions

Thruster no. (see Fig. 4)	Functions	Size
1	East-West, spinup	4.4 N
2	Spinup, North-South backup	4.4 N
3	North-South (primary)	4.4 mN
4	Momentum dump, ascent maneuvers, additional velocity corrections	8.9 N

velocity addition and backup the primary North-South ion thrusters. This hydrazine system was sized using an average I_{sp} of 235 s. The following requirements were assumed:

1) 30 rpm spinup and spindown (assuming $I_{yaw} = 3180$ N-m-s^2).

2) 155.6-deg reorientation for apogee firing using two 8.9-N thrusters firing over a 1-rad arc during each rotation of the satellite on its axis.

3) 90-deg longitude to orbit normal for the acquisition maneuver.

4) 180-deg longitude repositioning (assuming 30-day maneuver time) (Fig. 5 illustrates the dependence of propellant on maneuver time for longitudinal changes of $\Delta\lambda = 120$ deg and 180 deg using hydrazine).

5) $\Delta V_{E/W} = 3$ m/s/yr for 10-yr life for East-West positioning.

6) $\Delta V_{equiv} = 2.4$ m/s/yr for momentum dumping for 10-yr life.

7) A 10% fuel margin which could be applied to limited velocity addition or backup stationkeeping.

The North-South thrusters are two 4.4-mN mercury ion engines which are 8 cm in diameter and have a design I_{sp} of 3000 s. The engines which are mounted along the pitch axis, have been sized to provide 1 deg/yr inclination change.[7] They are fired alternately over an arc containing the node of the orbit as seen in Fig. 6. The power requirement is 125 W_e for approximately 9 h per day, with a cycle of 4.5 h per engine per day.

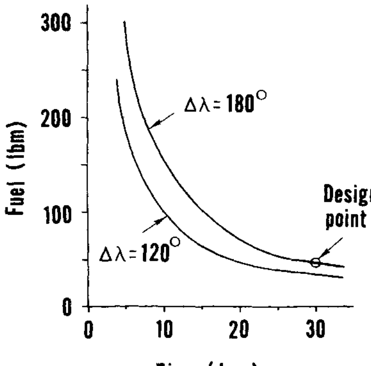

Fig. 5 Longitudinal repositioning performance.

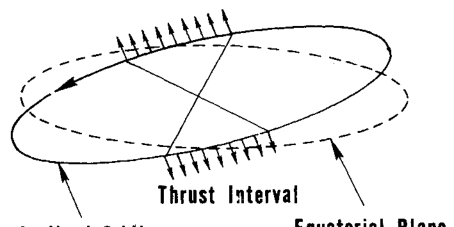

Fig. 6 Thrusting geometry for North-South stationkeeping.

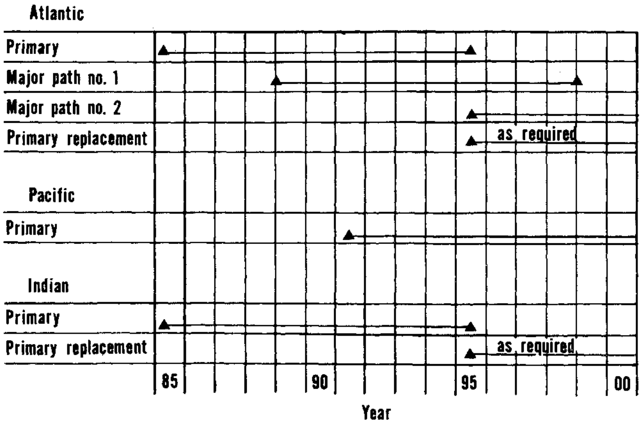

Fig. 7 Satellite on-station plan.

Table 3 Projected INTELSAT traffic[8]

Year	Atlantic (Region)	Pacific (Region)	Indian (Region)	Total
1977	15,720	3872	5964	25,286
1979	22,307[a]	5211[b]	8026[a]	35,544
1981	31,060	7134	11,176	49,370
1983	43,248	9765	15,561	68,574
1985	58,195[c]	13,368[b]	20,939[a]	92,502
1987	78,307	18,299	28,175	124,781
1989	105,369	25,049	37,913	168,331
1991	141,785	34,290	51,015	227,090
1993	190,786	46,939	68,646	306,371
1995	256,721	64,255	92,371	413,347
	(one-way voice channels)			

[a] Extrapolated at 18%.
[b] Extrapolated at 17%.
[c] Extrapolated at 16%.

Communications Payload

The proposed satellite is conceived to fulfill the requirements of the telecommunications market as projected by Van Trees,[8] and presented in Table 3, for the period 1985-1995. A system of five satellites, each capable of 80,000 one-way voice channels, is envisioned for this period. Figure 7 illustrates the on-station plan. Each satellite would continue to utilize the 4/6 GHz and 11/14 GHz frequency bands as planned for INTELSAT-V. Some increase in bandwidth is expected to result from the 1979 World Administration Radio Conference (WARC) due to anticipated needs. However, frequency reuse by polarization and multiple spot beams is expected to provide the bulk of the increased capability. The needs of high traffic areas will require additional techniques. The 4/6 GHz band is expected to provide 25,000 one-way channels, with the remaining 55,000 channels utilizing the 11/14 GHz. It is expected that the communications payload will require approximately 1650 W of power at full capacity.

The communications system will use circular polarization and generate several narrow spot beams simultaneously. Multiple spot beam generation could be accommodated by use of the transverse electromagnetic mode (TEM) lens or by use of flexible offset-feed reflectors. Both systems are compatible with the standardized bus concept proposed here.

Summary

The concept of a dynamic isotope powered communications satellite has been introduced with the intent of identifying potential advantages and its availability in the 1980's. This system offers to satisfy traffic requirements well into the 1990's, make efficient use of Space Shuttle launch constraints, and utilize long-overdue technologies such as ion thrusters and dynamic isotopes. Unique advantages of such a satellite include independence of sun angle, no power system deployment or slip ring power transfers, and an integrated thermal and power system.

The spacecraft utilizes a standardized bus design in order to minimize recurring costs. The configuration selected makes efficient use of the payload bay volume while providing a spin-stabilized body during ascent. A system of five of these satellites, each capable of 80,000 one-way voice circuits, will satisfy projected communications demands through 1994. This represents only one of many new applications of nuclear power to spacecraft designs.

Acknowledgment

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