REPORT 808-3*

BROADCASTING-SATELLITE SERVICE

Space-segment technology

(Question 2/10 and 11, Study Programme 2K/10 and 11)

(1978-1982-1986-1990)

1. Introduction

1.1 *General*

The large coverage area possible from a satellite-borne radio transmitter, especially on a satellite in the geostationary orbit, and the supporting technology which is available at present, will make possible the establishment of a broadcasting service to the general public. An earth-station transmitter could direct programme material to the satellite which, in turn, would broadcast this material over a wide area to individual or community receivers.

The technology applicable to the space segment of the broadcasting-satellite service is similar in many respects to the technology applicable to other satellite services. In a few areas, however, the technology needed for the broadcasting-satellite service will differ from that required in other services, and will require specialized research and development. These specialized areas include the generation of high RF power, high efficiency radio frequency generators, effective methods of heat conduction and dissipation from these high power RF sources, and the design and development of spacecraft antennas having low side-lobe levels and asymmetrically shaped beams.

The following sections of this Report are confined to discussions of these aspects of satellite design technology, apart from the last three mentioned above.

2. Primary power (see also Report 673)

2.1 *Solar arrays*

As a result of the increase in power requirements, attention has been directed to the use of light-weight sun-oriented arrays. Most of the interest is centred around photo-voltaic cells mounted on a flexible substrate which is either folded, or rolled on a drum during launch and transfer orbit [Ray and Winicor, 1966]. Deployment methods take several forms, as indicated in Annex I.

A 1.5 kW roll-out array has been successfully flight tested. Present estimates suggest that a reliable 12 kW (decreasing to 10 kW at the end of five years) roll-out array could be designed. The performance characteristics which might be expected from new developments in light-weight, deployable solar array technology are summarized in Annex I.

The ratio of primary power capability to the mass of broadcasting satellites in the geostationary orbit is dependent to some extent upon the attitude stabilization utilized (spin stabilized or 3-axis stabilized). A spin stabilized spacecraft in general will require more mass than a 3-axis stabilized spacecraft to provide equal prime power. Figure 1 shows the beginning of life; prime power and in-orbit mass for several representative spacecraft. The ratio of "beginning-of-life" to "end-of-life" power for a lifetime of 7 years is of the order of 1.3 to 1. The figure also indicates the approximate date at which the particular design was "frozen". The higher ratio of prime power capability-to-mass for the CTS and BSE spacecraft compared with the other examples results from the fact that they do not carry multiple transponders and the associated filters.

A solar array does not provide power during passage in the shadow of the Earth or of the Moon. With a geostationary satellite there is one Earth solar eclipse each day, but only within the periods of approximately 27 February to 12 April and 1 September to 15 October. Near the centre of these periods, the eclipse lasts about seventy minutes about midnight at the satellite longitude; the duration is less towards the beginning and end of the periods (see Fig. 2). In the case of longer eclipses, sufficient warm-up time must be allowed after the end of the eclipse. In the past, about half an hour has been required.

* This Report should be brought to the attention of Study Group 4.

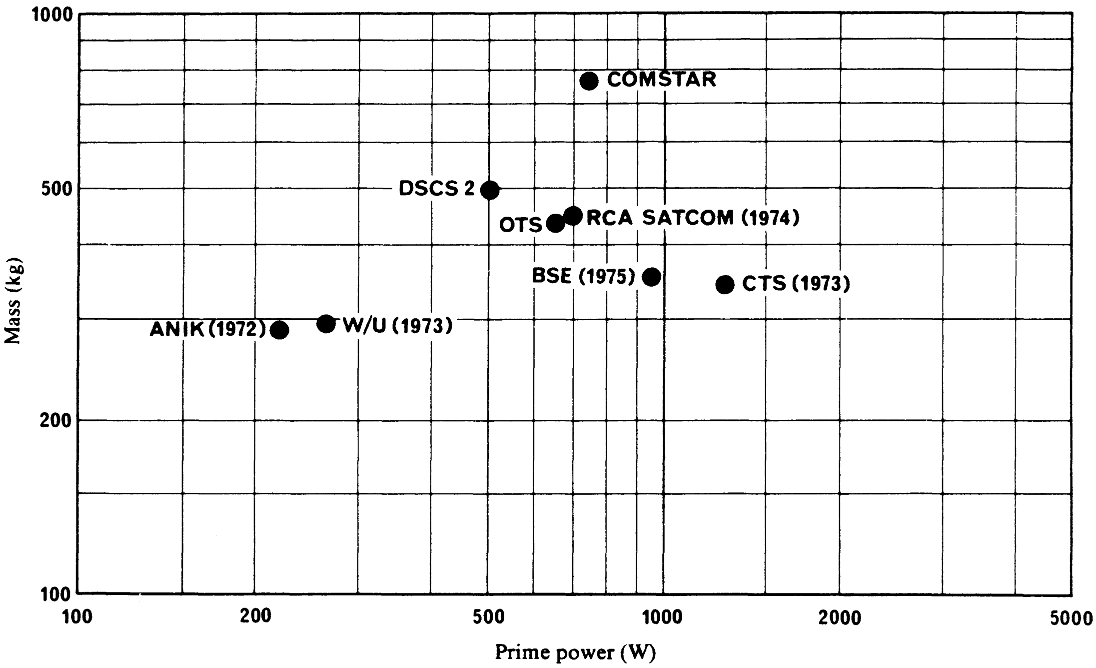

FIGURE 1 — *Relationships between in-orbit mass and prime power at beginning of life*

COMSTAR (COMSAT General + AT&T satellite)
CTS (Communications Technology Satellite)
BSE (Japanese experimental broadcasting satellite)
W/U (Western Union WESTAR)
ANIK (Canadian communications satellite)
DSCS (Defense Satellite Communications System)
OTS (Orbiting Test Satellite)
RCA SATCOM (RCA Americom satellite)

Note. – Dates in parentheses indicate approximate date at which the design was finalized.

Eclipses due to Moon shadow are not as regular in terms of times of occurrence, duration, and depth as Earth solar eclipses. The number of Moon solar eclipse occurrences per orbital location per year ranges from zero to four with an average of two per year; eclipses can occur twice within a twenty-four hour period. The duration of eclipses ranges from a few minutes to over two hours with an average duration of about forty minutes. Special problems in connection with battery recharging and spacecraft thermal reliability could arise when Moon solar eclipses of long duration and appreciable depth occur during the same period as Earth solar eclipses. It is possible to predict the characteristics of Moon shadow events with reasonable accuracy. Table I gives a prediction for the period 1981 to 1990 for orbital position 110° longitude. Because of the irregular nature of the Earth and Moon orbits, recurrence of similar Moon solar eclipses occurs at a minimum of one Saros cycle (approximately 18 years) and can be as long as three Saros cycles [Ehara, 1979; CCIR, 1978-82a; Siocos, 1981]. Table II gives the timetable of Moon solar eclipse occurrences for 19° W for the 18-year period, 1983-2000, while Fig. 3 provides information on their distribution of occurrence in terms of time of day and duration [CCIR, 1982-86a].

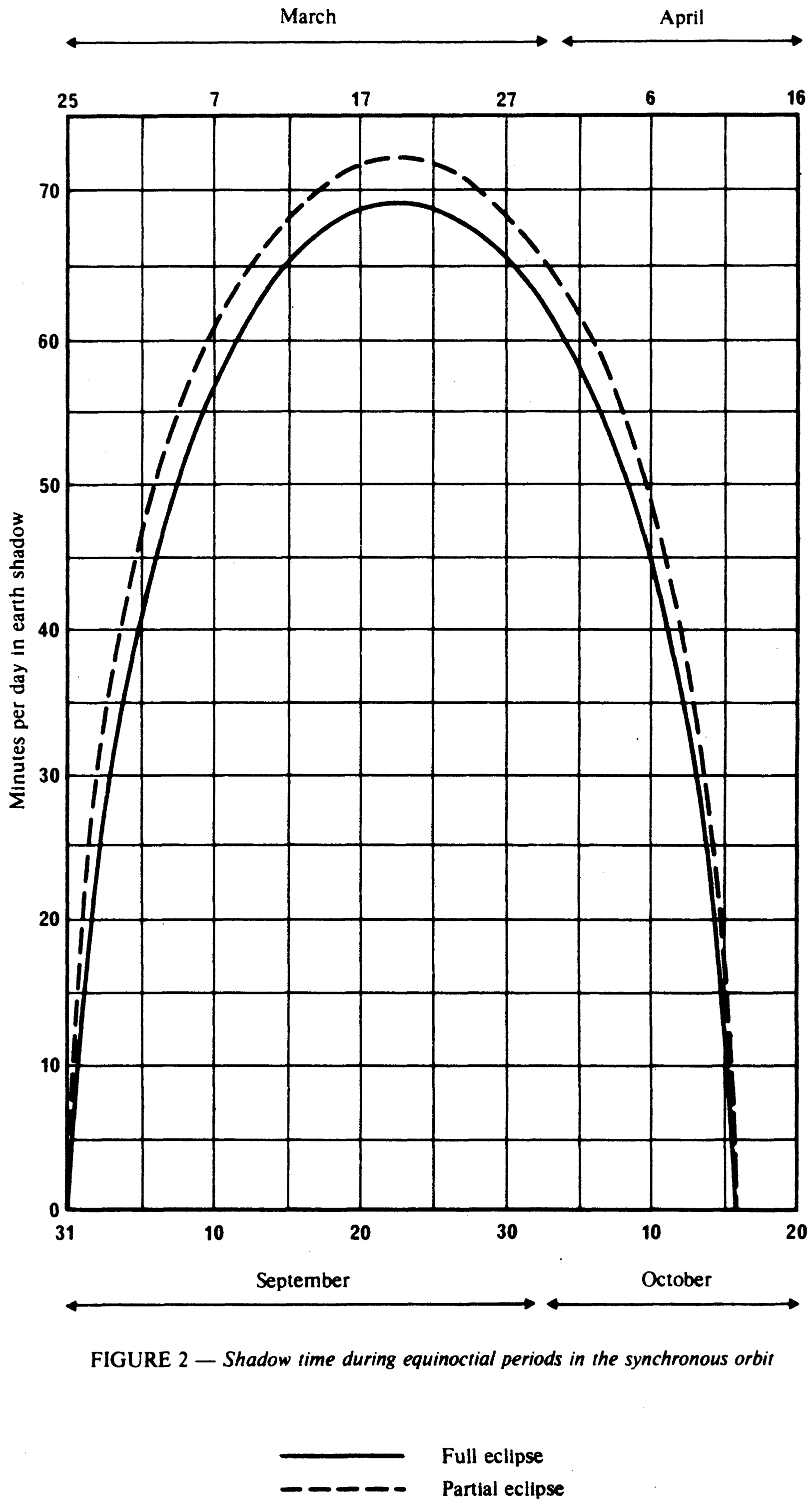

FIGURE 2 — *Shadow time during equinoctial periods in the synchronous orbit*

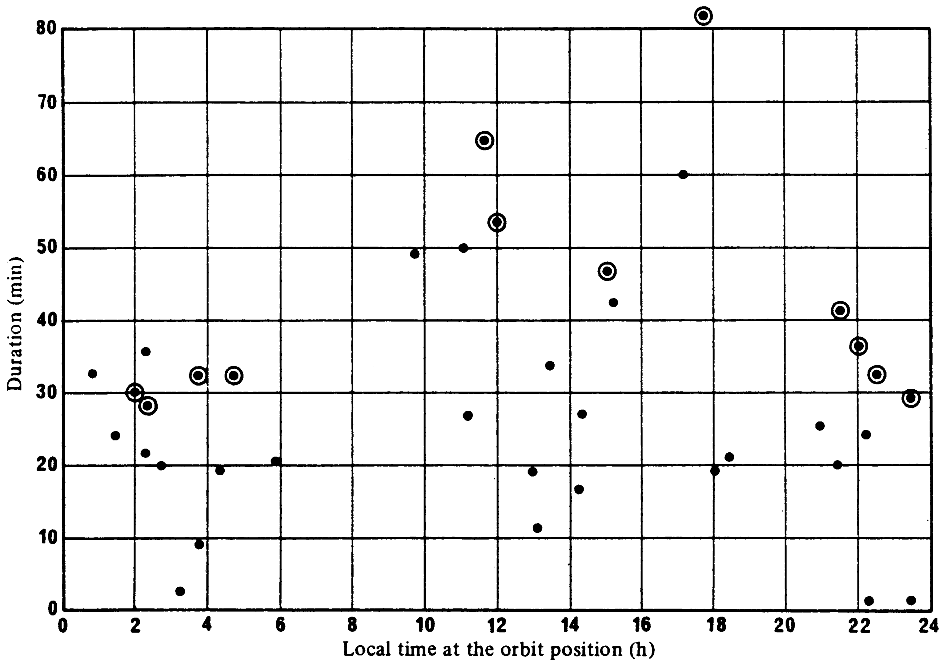

FIGURE 3 – *Distribution of Moon solar eclipses (19° W)*

• 0-50% shadowing
◉ 50-100% shadowing

TABLE I – *Predicted Moon solar eclipses at geostationary-satellite orbit.*
Position: 110° E longitude

Date	Entrance time (UT)	Duration (min)	Shadowing (%)
1981 January 6	0252	56	85
February 5	0832	102	86
July 30	1924	30	39
December 26	0134	66	96
1982 January 24	1926	32	35
1983 January 13	1926	22	11
January 14	1330	38	61
July 10	1524	14	9
November 5	0802	140	62
1984 May 1	0516	44	40
1985 April 20	0450	20	4
October 14	1250	38	69
1986 April 9	1354	34	43
1987 February 28	1054	66	44
March 29	1538	28	59
April 27	1856	26	19
September 23	0448	42	47
1988 August 12	1516	32	69
September 11	0404	44	56
1989 August 1	1622	10	3
August 31	0340	38	30
1990 July 21	1902	28	48

TABLE II – *Timetable of Moon solar eclipses for geostationary satellites*

Position: 19° W longitude

Date	Entrance time (UT)	Duration (min)	Shadowing (%)
1983 January 14	0208	22.5	26
July 10	0410	18.2	19
July 10	2138	19	17
1984 January 3	0210	28.2	68
June 29	0146	23.5	45
October 24	1410	26.4	20
1985 May 19	0845	19.2	8
1986 May 9	0049	32.6	32
October 3	2330	28.1	65
1987 March 29	1337	31.9	32
August 24	2129	41.1	72
October 22	0632	17.7	7
1988 April 16	0429	32.4	68
September 11	0202	35.7	36
1989 February 6	1744	83.2	88
1990 June 22	2339	1	1
August 20	1257	11.5	8
1991 February 14	0626	14	7
February 14	1050	64.6	70
December 6	0154	27.2	66
1992 February 3	1006	47.5	18
November 24	0313	1	6
1993 May 21	2234	31.2	70
December 13	1516	21.8	10
1994 June 9	1525	42.7	36
November 3	2202	35.3	63
1995 April 29	1119	25.4	13
September 24	0606	21	23
1996 March 19	0358	8.6	8
October 12	1254	18.1	11
October 12	2215	1	2
1997 April 7	1456	44.2	50
April 7	2057	26.3	35
1999 January 17	1203	51.4	100
February 16	0239	19.2	24
February 16	1716	60.1	37
August 11	0344	31.4	84
2000 January 6	1055	50.4	27

Arrays of silicon solar cells have served quite satisfactorily as the prime power source in satellites and are likely to be employed in this application for many years to come. The theoretical limit (approximately 25%) on the efficiency of silicon solar cells is much higher than the efficiencies now being realized. Therefore, several efforts are under way to improve the efficiency of the silicon solar cell [Lindmayer and Allison, 1973; Revesz, 1973; Arndt, 1974; Statler and Treble, 1974; Haynos *et al.*, 1974].

2.2 *Batteries*

Batteries can be employed to provide a limited operational capability during eclipse. However, to provide full operational capability for the high power requirement would greatly increase the weight of the satellite. The practical consequences of Earth solar eclipse outage can be minimized by having the service break occur after prime viewing time (normally taken after midnight) in the service area, by placing the satellite to the west of its service area. However, it will not be possible to overcome the effects of the Moon solar eclipse occurrences by such techniques, as this phenomenon is not as regular in times of occurrence, duration and depth as Earth solar eclipses (see § 2.1).

The elimination of eclipse protection constraints may significantly enhance planning flexibility and could increase the orbit-spectrum resources available for planning.

State-of-the-art battery technology makes it feasible to support eclipse operation of a significant number of channels for small service areas using the lowest capacity, low-cost spacecraft being considered for the broadcasting-satellite service. As an example, analysis has shown that STS/PAM-D class spacecraft* can support a communication payload capacity of 350 W RF power at the high power amplifier (HPA) output by battery operation through the full Earth solar eclipse period.

Table III illustrates the number of eclipse-operated channels for two typical levels of maximum e.i.r.p. for 7 to 10-year service lifetime. A TWT efficiency of 45% was assumed in the analysis [CCIR, 1982-86b].

TABLE III – *Eclipse operation capability for 7-10 year service lifetime using STS/PAM-D class spacecraft*

Beam diameter (degrees)	Maximum e.i.r.p. (1) 58 dBW		Maximum e.i.r.p. (1) 62 dBW	
	HPA output power per channel (W)	Maximum number of channels	HPA output power per channel (W)	Maximum number of channels
0.8	19	18	48	7
1.2	43	8	108	3
1.6	76	4	191	2
2.0	119	2	300	1
2.2	138	2	350	1

(1) Assumes 1.5 dB of TWT output circuit loss.

2.3 *Other power sources*

Nuclear reactors and fuel cells are possible sources of primary power, but additional development will be required before they will be competitive with solar arrays in terms of cost, mass and reliability.

Thermoelectric junctions and thermionic cells may also be considered as a means of converting heat from the Sun or from isotope sources into electrical energy, and offer the possibility of less total mass in the power unit for a given electrical output. Work is in progress on the development of such devices and their application to spacecraft [IEE, 1968].

3. **Radio-frequency power**

3.1 *Summary of radio-frequency power limits*

The final stage of the broadcasting transmitter is the major consumer of power on the satellite. Solid-state transmitter modules for a frequency of about 860 MHz and at power levels of about 100 W have been demonstrated by the USA on the Applications Technology Satellite-6 (ATS-6). Twenty watt solid state transmitters at 2.6 GHz have also been demonstrated on ATS-6. Appreciably higher powers, particularly at higher frequencies, will require vacuum tubes. For the frequency range 2 to 20 GHz, travelling-wave tubes or klystrons might provide maximum powers in the range of 1 to 7.5 kW, depending on the frequency. An efficiency of 35 to 65%, including any loss in power conditioning units, can be achieved with these systems. A 200 W travelling-wave tube with an efficiency of about 50% at 12 GHz was used in the US/Canadian Communications Technology Satellite Programme. For the 12 GHz band, Table IV provides pertinent parameters for existing travelling-wave tubes (TWTs) applicable for broadcasting satellites.

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* STS/PAM-D class spacecraft: spacecraft generally having a mass of 1250 kg in transfer orbit.

The total radio-frequency output power is limited by the solar array power, the losses in the power conditioning sub-system, and the transmitter efficiency. The output power of a single tube is limited by cathode loading and beam compression. The power in a waveguide component is limited by radio-frequency breakdown and heating. Other factors which impose practical constraints on spacecraft transmitter power include spacecraft weight, power flux-density limits applicable in particular frequency ranges, and the consequences of interference on the efficient use of the geostationary arc.

TABLE IV – *Travelling-wave tubes for broadcasting satellites*

	20-30 W	100-110 W	120-140 W	230 W	235 W	230-260 W	100 W	200 W	450 W
RF circuit	Helix	Helix	Helix	Helix	Helix	Helix	Coupled cavity	Coupled cavity	Coupled cavity
Number of collectors	1-3	4	4	4	5	5	3	10	5
Nominal efficiency (%)	35-40	50	50	52	50	49	50	48	50
Instantaneous bandwidth (MHz)	200	200	300	105-200	400	105-200	180	85	100-200
Design lifetime (years)	7-10	7-10	7-10	7-10	10	7-10	3	3	7-10
Collector cooling	Conduction	Radiation	Radiation	Radiation	Radiation	Radiation	Radiation	Radiation	Radiation
Collector/body dissipation (%)	–	70/30	70/30	70/30	70/30	70/30	40/60	40/60	70/30
Mass (kg)	1.0	3.0	3.2	2.5	4.5	3.5	6.8	11.9	6.8
Programme	OTS, ECS, TDRSS and others	BS-2	BS-3	TDF-1 & TDF-2	OLYMPUS	TV SAT TDF-1 TDF-2 Tele-X	BSE	CTS/HERMES	–
Status (1989)	Flown and qualified	Flown	To be flown 1990	Flown	Flown	Flown	Flown	Flown	Engineering model

3.2 *Equivalent isotropically radiated power and its stability*

For a given zone of coverage, say 2°, maximum values of e.i.r.p. of 75 dBW at 700 MHz and perhaps 70 dBW at 12 GHz can be expected with still higher powers likely to be technically feasible at a later date.

Based on measurements obtained from the BSE programme over a two-year period, the standard deviation of the variation of the transmitting power was determined to be less than 0.2 dB [CCIR, 1978-82b].

3.3 *Thermal control*

The major problems are associated with heat rejection from the power conditioning components and from the high-power stages of the transmitter. Solid-state components lend themselves to simple passive methods of control. However, the low operating temperatures (350 K to 390 K) require a significant amount of radiator area. Other devices, such as gridded tubes and microwave tubes, have high heat dissipation densities and high temperatures. The higher operating temperatures (470 K to 500 K) minimize the radiator area requirements.

The development of heat pipes provides a promising method of heat transfer from the source to the radiator. Heat pipes have been used for thermal control on spacecraft [Anand, 1968] and in heat rejection from high power tubes on the ground.

4. Station keeping and attitude control

This section treats the operational requirements for, and the status of technology relating to, station-keeping and attitude control of geostationary satellites.

The possibility of physical interference between spacecraft in the geostationary orbit, and the blockage of the emissions of one satellite by another, the countermeasures that could be employed, and the estimates of the increase in fuel mass that could be required, are discussed in Report 1004.

4.1 *Station keeping*

The slight inequalities in the gravitational field of the Earth, together with the gravitational forces due to the Sun and Moon have perturbing effects on satellites which otherwise would remain stationary, but these can be encountered by orbit correction or "station-keeping" techniques.

A geostationary satellite will experience extremely slight eastward or westward forces which change the longitudinal drift of the satellite.

Other perturbing forces tend to change the inclination of the orbital plane by approximately 0.8° per year, thereby causing the satellite to undergo corresponding daily variations in latitude.

Present station-keeping techniques develop corrective thrust to overcome the gravitational forces by the use of small propulsion jets on the satellite, operated by propellants stored on board. The extent to which correction is required depends upon the allowable displacement of the satellite.

East-west (longitudinal) station-keeping is usually essential, because the uncorrected drift may be relatively large and rapid. Fortunately, the required rate of propellant consumption is very low. North-south (latitudinal) station-keeping, to keep the orbit close to the plane of the equator, will become more important as satellites achieve longer life. Latitude station-keeping requires about ten times the amount of propellant as does longitude station-keeping.

For frequencies up to 1 GHz, where the required beamwidth of the receiving antenna is not expected to be less than 5°, a station-keeping accuracy of 1° will be sufficient to ensure that the satellite remains in the beam of receiving antennas. Above 1 GHz, accuracies of the order of 0.25° may be required. The longitudinal drift for satellites at present in the geostationary orbit can be held to 0.1° during a satellite lifetime of at least five years. Satellites now under construction will be capable of controlling the daily variation in latitude to the same accuracy. Station-keeping techniques for achieving the orbital accuracy required for a geostationary broadcasting satellite are, therefore, technically feasible.

Station-keeping of the order of 0.1° is desirable to maximize the efficiency of utilization of the geostationary satellite orbit spectrum space. As a general rule, satellite drift should be limited to 5% or less of the spacing between adjacent satellites.

4.2 *Attitude control*

The pointing accuracy of the satellite antenna beams is very important in satellite broadcasting in order to obtain the best utilization of the antenna directivity. On the other hand, solar pressure and thermal gradient are the causes of depointing of the satellite antenna beams. To maintain the pointing of the antenna, it is therefore necessary to control the attitude of the satellite with as high an accuracy as possible. This accuracy depends mainly on the type of sensor and on the system chosen for the attitude control.

Attitude control for all three axes of a geostationary satellite is generally required to maintain coverage of the desired service area and to minimize spillover. In some rare applications, it may be sufficient to provide control of only the pitch and roll axes [CCIR, 1978-82c]. However, this would require an axially symmetrical (i.e. circular) circularly polarized antenna beam pointed at the sub-satellite point (an unlikely situation that will not be treated further since the satellite is normally positioned west of the service area to delay the onset of solar eclipse) or pointed with radio-frequency sensors toward a beacon located at the centre of the service area. Generally, even this latter case would benefit by controlling all three satellite axes when solar array mispointing and propellant budget factors are considered.

The pointing accuracy which can be achieved depends on the types and quality of attitude sensors employed for each axis.

Allowing for satellite attitude angle errors, the boresight error circle of the transmitting antenna should be capable of being maintained within 0.1° radius. With the introduction of improved systems (e.g. radio-frequency sensing; see Report 546) this error could be considerable reduced. Measurements made on the TDF-1 satellite have shown that a boresight error circle of 0.01° radius can be achieved.

Present attitude control systems used on most geostationary communications satellites can control yaw so that the error is in the range of ± 0.3° to ± 0.8°, depending on various factors. The lower values (of the order of ± 0.3°) can be achieved by using two separate attitude references sufficiently far apart; for example, use of an RF sensor and an IR sensor (when the coverage area is sufficiently far away from the sub-satellite point) or use of two RF sensors (when the coverage area is large enough). Yaw stabilization to within ± 0.1° has already been demonstrated in orbit with the ATS-6 satellite by using star sensors [Redisch, 1975] but such sensors represent a significant increase in the mass and complexity of the satellite.

The BSE experimental satellite of Japan limited the dynamic errors of its attitude control by zero-momentum three-axis stabilization, within ± 0.03° for pitch and roll by the use of the earth sensor, and ± 0.3° for yaw by the use of a combination of earth sensor and radio-frequency monopulse sensor, almost throughout the day [Shimizu *et al.*, 1980]. (An angle of approximately 7° was subtended between the earth station beacon transmitter and the sub-satellite point as viewed from the satellite.)

The relationships between attitude errors and movement of the antenna footprint on the Earth's surface are described in Report 546. A range of values is given, reflecting future developments in recognition of the critical effect of pointing error on planning.

5. Transmitting antennas

The maximum gain of an antenna and the way in which the gain decreases as a function of angle is important in interference calculations. Therefore, guidelines are required as to the probable performance of transmitting antennas for satellite broadcasting and of receiving antennas on the ground. A detailed examination of antenna patterns and technology is given in Report 810.

6. Coverage

The area of the Earth covered by a satellite antenna beam, and the shape of that area, depend on the satellite antenna pattern *per se*, and also on the pointing offset of the beam from the satellite nadir (the sub-satellite point). Since the satellite is not at the origin of the terrestrial co-ordinate system, the antenna pattern co-ordinates are not linearly related to the terrestrial co-ordinates. Methods for calculating Earth "footprints" of satellite antennas are available in the literature [for example, Jacobs and Stacey, 1971; Adamy, 1974].

7. Lifetime

Current system planning assumes a mean satellite life of about seven years. So far, studies and the performance of satellite systems encourage the view that a life expectancy of up to ten years can be achieved by careful design and provision of certain reserve equipments. In particular, the solar panels must be large enough to allow for the progressive deterioration that takes place in space. Fuel requirements for station-keeping and attitude stabilization may well be large, possibly of the order of 20% to 25% of the mass of the satellite if existing techniques are employed. With the advent of ion thrusters, less fuel will be required to realize the same manoeuvring functions. These thrusters may eventually replace traditional gas propulsion systems. However, these new thrusters are still in the developmental stage and full evaluation still has to be determined (see Report 843).

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CCIR Documents

[1978-82]: **a.** 10-11S/60 (Canada); **b.** 10-11S/114 (Japan); **c.** 10-11S/114 (USA).

[1982-86]: **a.** 10-11S/14 (France); **b.** 10-11S/59 (Canada).

ANNEX I

DEVELOPMENT OF LIGHT-WEIGHT SOLAR ARRAYS

The purpose of this Annex is to summarize the performance characteristics which might be expected from new developments in the technology of light-weight, deployed solar arrays. It provides a summary of development on light-weight solar arrays and may serve as a basis for determination of future research and development work in this area.

During the past several years there has been a considerable effort by a number of organizations to develop light-weight, deployed solar arrays. Two distinct types of solar arrays have been studied; namely, deployed, rigid arrays and deployed, flexible arrays.

The deployed, rigid arrays have been exclusively the fold-out type either folded around the satellite body during transfer orbit or contained in a flat pack, accordion fold, arrangement during transfer orbit. Deployment occurs in several steps usually commencing with the pyrotechnic release of latches or the cutting of cables. Depending on the type of array, the deployment continues with the solar panels unfolding, followed by the extension of a yoke mechanism to separate the array from the spacecraft. The deployed array is normally locked in place at the panel hinges. With a rigid array, transfer orbit power is readily obtained from the outer side of the stowed panels.

There are two basic types of deployable, flexible substrate arrays; fold-out and roll-out. The fold-out solar arrays use a flat pack concept to contain the solar cell blanket during launch. The deployment sequence begins with the pyrotechnic opening of a box or release of latches or cables holding the array against the spacecraft body. Deployment of the folded blanket takes place by extension of a pantograph, a boom, or a telescopic mast system attached to the blanket. During transfer orbit, the roll-out array is wrapped around a drum attached to the spacecraft. During deployment, the solar cell blanket is rolled out by the extension of a boom which is attached to the blanket. For both fold-out and roll-out systems a yoke is used to separate the array from the spacecraft.

A major advantage of flexible, fold-out systems over flexible, roll-out systems is their inherent higher packing density, since no drum is required. This allows for easier integration of a fold-out system to a spacecraft within a launch fairing. Usually for both types of flexible arrays an additional array is required to provide transfer orbit power during launch to geostationary orbit. In some advanced fold-out, flexible designs this power is provided by incorporating the transfer orbit panels into the flexible array.

Table V shows the weight-to-power and power-to-weight ratios for several deployed, rigid solar arrays. This Table is based on a one kilowatt wing of a two kilowatt solar array system. It first lists the weight-to-power at beginning of life, equinox conditions, including the array with its blanket, deployment, yoke, and stowage systems. The orientation mechanism weight-to-power ratio is broken out separately as is an estimated miscellaneous category to cover redundancy items such as redundant orientation mechanism motor windings and/or electronics, redundant latching, insulation, etc. The power-to-weight ratio at end of life (5 years) summer solstice conditions is also given. Finally, the effect of advanced cells on the power-to-weight ratio is shown. The end effect on the overall array of changing to advanced cells was estimated at 15 per cent which was assumed uniformly. It should be noted in Table V and later, in Table VI that in going from a designed or tested solar array system to a flight-qualified array, extra weight is estimated to provide for redundancy, temperature control, etc. It is even more evident when the Fleetsatcom or CTS and FRUSA numbers are compared to the typical early design numbers. Several examples of rigid solar arrays are shown. The first one listed which is being developed for an operational spacecraft is Fleetsatcom. The Fleetsatcom array is a rigid deployable array, initially folded around the periphery of the spacecraft. It uses conventional aluminium honeycomb substrates and solar cells.

TABLE V – *Comparison between deployable rigid solar arrays*

Type of array	FLEETSATCOM TRW - conventional, rigid foldout	MBB-ICS foldout (carbon fibre)		MBB-ULP (very light materials)		Matra foldout (glass fibre technology)		Flight type arrays Post 1980 (estimate)
		A	B(1)	A	B(1)	A	B(1)	
Array, including deployment and stowage, at beginning of life, Equinox (kg/kW)	54.0	31.0	31.0	18	18	28.6	28.6	
Orientation mechanism (kg/kW)	7.7	(4.3)	(3.4)	(4.3)	(3.4)	(4.3)	(3.4)	
Miscellaneous (2) (kg/kW)	Included in above	(1.5)		(1.5)		(1.5)		
Total at beginning of life, Equinox (kg/kW)	61.7	36.8	34.4	23.8	21.4	34.4	32.0	25
Total at beginning of life, Equinox (W/kg)	16.2	27.2	29.1	42.0	46.7	29.1	31.3	40
Total at end of life, 5 years, summer Solstice (W/kg)	11.4	20.9	22.4	27.7	30.8	20.4	21.9	26
Total at end of life (5 years), summer Solstice if advanced cells are used (W/kg)	13.1	24.1	25.8	31.9	35.4	23.5	25.2	Included in above
Reference	[1]	[2]		[3]		[4]		

( ) Assumed value, since a real value is not provided.

(1) Column B uses a lighter weight suggested by the Royal Aircraft Establishment (RAE) for the orientation mechanism and excludes any miscellaneous items.

(2) Includes any redundancy required in the orientation mechanism; for example, motor windings, any insulation required, redundant latches, etc.

The Messerschmitt-Bölkow-Blohm GmbH (MBB) improved composite structure (ICS) array uses aluminium honeycomb substrates and carbon fibre reinforced epoxy (CRFP) facesheets in a flat pack design with the deployment energy supplied by spiral springs on the panel hinges. This concept will be used on the ESA, OTS and Marots satellites. The ultra light-weight panel (ULP) is an advance on the ICS system using the same deployment approach but including a carbon fibre framework and very light-weight solar panels. The Matra system uses a flat pack design with aluminium honeycomb substrates and glass fibre faceskins. Deployment is by springs and hinges along with a cable and pulley system.

The MBB-ICS system shown utilizes 125 micron solar cells whereas the Matra analysis is based on 200 micron solar cells. Consequently, the MBB-ICS system is lighter and shows less degradation than the Matra system, at the end of five years. However, the availability of 125 micron cells in the large quantities necessary for production spacecraft is questionable. These systems may have to depend on the heavier cell.

The last column represents an estimate on weight-to-power for advanced flight-type systems for use in the post-1980 time period.

TABLE VI – *Comparison between deployable, flexible solar arrays*

Type of array	SNIAS flexible foldout	RAE flexible foldout	CTS flexible foldout	Hughes FRUSA flexible roll-out	Flight-type arrays Post-1980 (estimate)
Array, including deployment and stowage, at beginning of life, Equinox (kg/kW)	23.0	16.6	37.7	35.8	–
Orientation mechanism (kg/kW)	4.3	3.4	Included in above	Included in above	–
Miscellaneous (1) (kg/kW)	1.5	–	Included in above	Included in above	–
Total at beginning of life, Equinox (kg/kW)	28.8	20.0	37.7	35.8	18
Total at beginning of life, Equinox (W/kg)	34.7	50.0	26.5	27.9	56
Total at end of life, 5 years, summer Solstice (W/kg)	22.9	36.2	17.5	18.4	37
Total at end of life (5 years), summer Solstice if advanced cells are used (W/kg)	26.3	41.6	20.1	21.2	Included in above
Reference	[5]	[6]	[7]	[1]	

(1) Includes any redundancy required in the orientation mechanism; for example, motor windings or electronics; any insulation required; redundant latches, etc.

The data in Tables V and VI are based on the following sources:

[1] BILLERBECK, W. J. and CURTIN, D. J. [1974] Flexible solar array applications in communications satellites. Proc. Intersociety Energy Conversion Engineering Conference, San Francisco, California, USA.

[2] CRABB, R. L. and SCHNIEDER, K. [1973] Development of an advanced lightweight rigid solar array. Conf. Rec. Tenth IEEE Photovoltaic Specialists Conference, 306-316.

[3] KOELLE, D. E. [1974] Advanced lightweight rigid solar arrays based on carbon fibre technology. Internat. Astronaut. Fed., XXVth Congress, Amsterdam, Holland.

[4] LARSSON, H. [1973] Problems of development and test of large lightweight solar arrays. Proc. Internat. Congress, "The Sun in the Service of Mankind", Paris, France.

[5] BARKATS, G. [1973] Development of flexible, fold-out solar array. Proc. Internat. Congress, "The Sun in the Service of Mankind", Paris, France.

[6] TREBLE, F. C. [1974] The RAE lightweight solar array. Royal Aircraft Establishment Technical Report 73172. (Numbers contained in this report were upgraded to a 1 kW system by F. C. Treble.)

[7] SACHDEV, S. S., QUITTNER, E. and GRAHAM, J. D. [September, 1974] The Communications Technology Satellite deployable solar array sub-system. International Conference on Photovoltaic Power Generation, Hamburg, Germany (Federal Republic of).

Table VI is similar to Table V except that it concerns flexible substrate solar arrays. It includes data on several solar arrays. The array developed by AEROSPATIALE uses a pantograph, fold-out system with launch stowage in an aluminium honeycomb-walled box. The pantograph is spring loaded and self deploys when released. The rate is controlled by a winch and motor. The solar cells are mounted on a Kapton substrate designed in modular form to be usable for different power levels.

The Royal Aircraft Establishment solar array, based on the background of a 280 W hardware development programme, as proposed, would use a pneumatically actuated telescopic mast to deploy a light-weight flexible, foldout panel using 125 micron wrap around contact solar cells. The light-weight orientation mechanism is based on design estimates by Hawker-Siddeley Dynamics. RAE also indicates that transfer orbit power could be provided by a light-weight rigid panel which would be part of the flexible array. As with the MBB-ICS system, the RAE estimates are based on using the 125 micron cells; consequently, the degradation rate is less than with other systems.

The Communications Technology Satellite (CTS) array is a flexible, fold-out array deployed by a Bi-Stem boom. This satellite was launched in January 1976. The data given is based on direct scaling of the actual array. Since the CTS array consists of two wings with approximately one kilowatt total at end of life, it is not optimized as a one kilowatt/wing system. Consequently, the numbers shown are heavier than would be expected in a two-wing, one kilowatt per wing array. On the basis of one kilowatt per wing, the CTS system would be expected to achieve at least 20 W/kg at the end of life.

The flexible rolled-up solar array (FRUSA) system is a flexible roll-out solar array deployed by a Bi-Stem boom. The FRUSA array was built by Hughes Aircraft Company and launched in 1971. It performed successfully and provided several months of useful data.

The last column in Table VI is the estimate on weight-to-power for flight-type flexible solar arrays for use in advanced flight-type systems in the post-1980 time period. It must be stressed that in both Tables V and VI these are estimates based on information available at present. Particular missions or new, unique types of arrays could change these estimates.

ANNEX II

Annex II gives a representative summary of pointing accuracies obtained with American communication satellites in 1976.

TABLE VII

Item	Error (degrees)	
	North-South	*East-West*
Long-term variations:		
– station keeping	±0.02	±0.01
– attitude determination	±0.01	±0.01
– antenna thermal distortion	±0.04	±0.04
– attitude drift	±0.1	±0.1
Sum of square roots	±0.11	±0.11
Short-term variations:		
– spin (pitch)	not available	±0.035
– nutation	±0.04	±0.02
Total error	±0.150	±0.185