Some Aspects of the Pointing Problem for Optical Communication in Space

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An examination is made of the problems of pointing the extremely narrow beams envisaged in current concepts of laser communication systems. Atmospheric effects are shown to prejudice seriously reliable direct communication using such beams between interplanetary vehicles and Earth, and they lead to consideration of relay schemes: of these the preferred alternative of a satellite or satellites in a high near-polar orbit is examined in relation to a hypothetical Earth/Venus mission. The difficulties of tracking in sunlight or when luminous bodies such as the moon or Venus enter the receiver field are discussed. Their deleterious effect on communication bandwidth is illustrated by examples. The need to compensate for velocity aberration and propagation time and means for the calculation of beam offset are described. Particular attention is given to the complex case of communication between vehicles when one is in a near-polar Earth orbit or when one is normal to the ecliptic. The importance of beam-deflecting devices in providing rapid response in tracking and precise, correct pointing is explained with reference to a conceptual optical system.

Introduction

IT is apparent from the literature that the future use of optical, as an alternative to microwave, techniques for communication in space over great distances depends upon the concentration of the radiation into extremely narrow beams. Such beams are made possible by the coherence of laser emission, and it becomes relevant to inquire as to how far the narrowing process can be taken and what the limitations are.

Laser communication techniques currently available are relatively crude and demand the narrowest possible beamwidths to provide the maximum signal intensity and signal-to-noise ratio at the receiver. Current studies of ultimate performance of laser communication channels typically refer to beamwidths of 1 arc sec or less, even when they are associated with advanced techniques not yet available, such as frequency modulation, narrow-band reception methods (including optical heterodyne or laser amplification), and improved optical components.[1] The use of very narrow beams, therefore, appears inherent in optical space communication systems. Such beams are sensitive to velocity aberration, propagation time, and refraction effects, which combine to create the beam-pointing or -aiming problem.

This paper attempts to show that the ability to direct an extremely narrow beam at a target receiving station, in order to illuminate it with certainty, may be a critical factor in the feasibility of laser communication in space. The approach is to examine the situations arising during a typical space mission assumed to depend solely upon a laser channel for two-way communication. The channel would have the maximum bandwidth from space to Earth and a command and data link of lesser bandwidth from Earth to the space vehicle which need not necessarily be optical.

The most simple arrangement for the suggested channel would comprise a number of receiving stations around the Earth to maintain optical contact. However, since a brief consideration of the optical properties of the atmosphere shows them to be a serious hindrance to communication with a typically narrow beam, the first step in the approach is to consider the effects of the atmosphere. It is believed evident that extra-atmospheric reception is essential to continuous and satisfactory communication with a space vehicle at interplanetary distances.

The facility for extra-atmospheric reception can be provided by the use of relay satellites. An alternative concept envisages use of an aircraft flying in the upper troposphere in the higher latitudes. This concept offers several potential advantages, such as the removal of many restrictions imposed on the communication equipment by satellite operation, enhancement of reliability through crew supervision and maintenance, cost reduction, and most important of all, relief of the beam-pointing problem.

References made in the course of the paper to reliability in the two-way communication channel relate to unscheduled interruptions in service. Consequently, occultation by the moon or Earth is considered only with regard to the reacquisition problem that it creates. From the communications point of view, conjunctures of the relay satellite with either Earth or the moon are almost as devastating as occultations.

The mission chosen as the basis of discussion is an Earth-to-Venus probe in a favorable period, and following the maximum payload trajectory, a voyage lasting something over 100 days. This trajectory is shown in Fig. 1, which also shows the relative position of Earth and probe (or spacecraft) at various stages in the mission and the state of the illumination of Earth as seen from the probe.

Conditions for optical communication differ greatly in the two directions. The Earth's station will operate mostly in sunlight. It will see the space vehicle transmitter against a sky whose darkness will depend upon the amount of atmospheric scattering of sunlight. The space vehicle must track the Earth station with the sunlit Earth full in its field of view to establish the communication channel. The problems of acquisition, tracking, and beam pointing are comparable with those of the optics in the laser communication channel. Both vary widely during the course of the mission.

Communication Aspects

It appears that optical methods will not compete successfully with radio microwaves for long-range communication until the advanced laser techniques referred to in the intro-

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duction are perfected, enabling optical bandwidths to be reduced to within one or two orders of the information bandwidth. Until then, the latter will be restricted severely by noise considerations. System design and operations will be restricted by the need to avoid situations in which bright sources appear in the acceptance field or in which the background is appreciably luminous.

A brief review of the state of the art will support this statement. Current technology indicates the use of an amplitude-modulated, continuous, helium-neon laser emitting at 6328 Å (4.7×10^{14} cps) or 11,523 Å (2.6×10^{14} cps). The conceivable performance of a channel using the latter radiation is given in Table 1. The listed values are derived, as shown in the Appendix, on the basis of range, 80×10^6 miles; receiver diameter, 100 cm; signal/noise ratio, 1000; beamwidth, 1 arc sec; and optimistically, a total laser emission of 100 mw.

The value given for the information rate H for the channel itself can never be exceeded. It is limited by noise in the optical signal by reason of the quantized nature of the radiation. This introduces an uncertainty in the number of photons arriving at a detector in a particular integrating period, which in a communication context is one-half of the inverse of the information channel frequency.

In ignoring all other sources of noise, it is implied that the receiver has a narrow-band optical filter to exclude unwanted light as far as possible. Such a filter might have a passband of 10 Å. This is equivalent to more than 10^{11} cps, approximately eight orders in excess of the information bandwith. This is the source of the noise problem.

As the received signal intensity varies inversely with the range squared, the situation at intermediate ranges need not be so adverse with regard to either beamwidth or bandwidth because the first can be relieved by widening the transmitter beam. The extent to which the beam is widened depends on how it is desired to share the advantages of possible increases in bandwidth or beamwidth. In this paper it is assumed that increased beamwidth is more desirable, and continuous adjustment of beamwidth to illuminate a constant circle is assumed. The diameter of the illuminated circle at 80×10^6 miles is 400 miles. This constant-illuminated-area concept means that the performance of the communication channel will remain unchanged during the mission as will also the apparent brilliance of the transmitters being tracked.

It is relevant to compare the brightness of a transmitter having the parameters indicated previously with that of other heavenly bodies. The magnitude of a source M is determined by the expression $M = 12.5 \log_{10} I_R/I_0$ in which I_R and I_0 are relative illuminances produced by the source and a zero order-of-magnitude star under the same conditions. The energy received from such a star outside the atmosphere is reported to be 0.41×10^{-15} w cm^{-2} Å^{-1} at 10,000 Å. From this it is found that photons would arrive from the laser beam at about the same rate as from a 6.5 magnitude star through a 10 Å filter centered on the laser wavelength (11,523 Å).

There is an average of one star of equal or greater magnitude per square degree of sky, which indicates the probability of false acquisition. It is too high to be ignored in the case of the Venus-bound spacecraft under the circumstances in which it is acquired and tracked. It can be greatly reduced by imposing on the transmission a "signature" or characteristic modulation, unless it can be identified by an inherent characteristic, such as the synchronizing signal of a television channel or a subcarrier frequency.

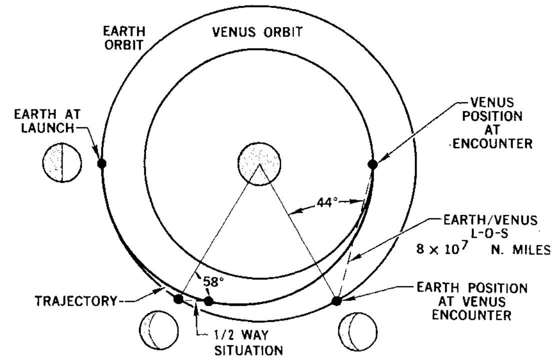

Fig. 1 Earth/Venus mission.

The signature of a directly amplitude-modulated audio channel could be a constant tone outside the voice frequency band. Later discussions of acquisition and tracking assume the transmitted beam to carry such means of discrimination. Inevitably the power available for the communication function is reduced thereby; this could be avoided by the use of a second transmitting laser. Since this use would add to system weight, lower reliability, and add another component necessitating precise alignment, it does not appear worthwhile.

Earth, moon, and Venus cannot be excluded at all times from the acceptance field of the receiving telescope on either the spacecraft or relay satellite, and it is relevant to estimate the effect of their coming into full view on channel performance. Their effect at maximum range of 80×10^6 miles is shown in Table 1.

The reduction in data rate caused by the sunlit Earth or moon may be tolerated in the Earth-to-spacecraft channel intended for data and command transmission, but the effect on the ability of the spacecraft to track and discriminate the relay satellite cannot be thus tolerated. The appearance of Venus in the field of the relay satellite receiver makes both communication and tracking virtually impossible. The unavoidable situation is typical of those that will arise repeatedly in interplanetary missions and emphasizes the need for narrow-band receivers if the laser is to be an effective communication device.

Atmospheric Effects

The following phenomena impede direct optical communication between space and the Earth's surface at visible or near infrared frequencies: 1) clouds and precipitation, 2) absorption, 3) mass refraction, 4) turbulence, and 5) scattering. Clouds are completely opaque to these frequencies, and the transmissions of rain and snow are negligible

Table 1 Background effects

Noise source	Photons/sec $= N$	Photons arriving in interval $\frac{1}{2} B$ sec $= n$	Signal/noise power ratio $= S = n_s^2/(n_s + n_n)$	Maximum information rate $= H$ bits/sec
1 Channel only	$N_s{}^a = 1.1 \times 10^5$	$n_s{}^a = 1000$	1000 (30*db*)	1000
2 Earth	4.3×10^8	4.3×10^6	0.25	62
3 Moon	1.0×10^7	1.0×10^5	10	340
4 Venus	7×10^9	7×10^7	0.014	2.0
5 Day sky	6×10^{10}	6×10^8	0.0017	0.23

[a] Effective signal photons.

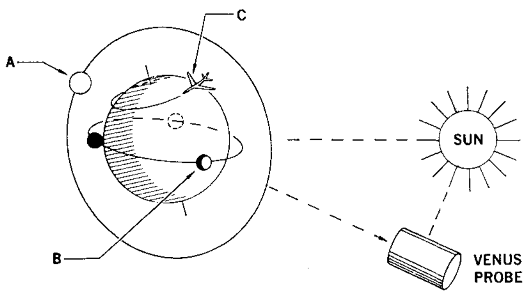

Fig. 2 Relay system concepts.

for communication purposes. The probability of interruption can only be reduced by establishing stations where clouds and rain are infrequent. To what extent this is possible is beyond the scope of this paper. Atmospheric absorption is largely dependent on moisture content and is about 50% for the normal thickness of the atmosphere. The moisture is concentrated in the lower layers, 98% being below 40,000 ft. Such "windows" as exist in the water vapor absorption spectrum do not coincide with available laser emissions.

The systematic refraction of the atmosphere follows the laws of physical optics. The effect varies with elevation; its magnitude is known, and navigators and astronomers are accustomed to correcting for it. The maximum displacement occurs at the horizon, causing a star to be displaced upward by about 36 arc min. Superimposed on these systematic displacements are random rapid movements caused by turbulence and thermal inhomogeneities at different levels in the atmosphere. These movements have been shown to comprise a wide range of frequencies and amplitudes and are accompanied by changes in brightness.[2] The combined effects contribute to what is referred to as the astronomical quality of "seeing." The unaided eye sees the effect as "twinkling." There is strong evidence that much of this disturbance, shown to comprise frequencies of 1 to 1000 cps, is caused by turbulence in upper layers of the troposphere. During darkness, deviations of 2 or 3 arc sec are commonly observed; instantaneous deviations of as much as 30 arc sec have been reported. Stars are not easily observed during daylight, but large movements of observed objects caused by thermal convection are an every day experience when using a telescope, and it can be assumed that even at mountain elevations daylight conditions are likely to be worse than those noted by astronomers. Even these exceed the 1-arc sec beam of the advanced laser. Another effect is the "shadow-band pattern" of the illumination falling on a telescope objective which is caused by interference between light reaching the viewing objective through neighboring air cells of differing refractive index. The effective aperture of an objective may be reduced to nearly half its nominal value from this cause.

Since both absorption and random-refraction effects are principally thermal in origin, they are at their worst under the conditions of full sunlight, to which the Earth terminal receiver will be exposed during most of the Earth/Venus mission. Making allowance for the most adverse combination of these effects, it is concluded that they can reduce signal strength by a factor of 10, or 3 db, and make acquisition, tracking, and beam-pointing with the required precision unreliable and therefore impracticable.

One of the effects of scattering is commonly observed: the luminosity of the day sky makes it impossible for the observer to see any but the brightest stars. This luminosity is due to the scattering of both direct sunlight and that reflected from the Earth's surface. The intensity of reflected light varies from place to place according to the surface albedo. The luminosity diminishes rapidly with increasing altitude, and the intensity that can be accepted in a space-to-Earth laser channel depends on signal strength and tolerable signal/noise ratio. The effect of the day sky, as seen from the ground, on the hypothetical laser channel is shown in line 5 of Table 1. These data show both communication and tracking to be effectively prevented.

Published data on the spectral radiance of the sky at satellite altitudes and at angles in excess of 30° to the sun are meager and variable. It appears that the effect on channel performance will be slight in comparison with, for example, Venus.

System Concepts

An alternative to direct communication between an interplanetary vehicle and Earth is the use of relay satellites, providing an optical link with the space vehicle and radio links to Earth. Such satellites could be employed in either of two configurations illustrated in Fig. 2. One would consist of three or four satellites in a low-altitude, low-inclination orbit (B), the other of one or possibly two satellites in a high-altitude, polar orbit (A). In the first, at least three satellites would be necessary to provide continuous line-of-sight to the space vehicle. In the near-polar orbit, the altitude might be 10,000 miles, enabling one satellite to provide continuous line-of-sight to a deep space vehicle under favorable conditions.

The relative disadvantages of injecting a satellite into a high-altitude polar orbit have to be weighed against those of multiple satellites in more convenient low-inclination orbits. The latter has significant disadvantages. Since the satellites are seen mostly against the background of the sunlit Earth, an unacceptable signal/noise ratio results, and the difficulty of tracking is increased. It is expected that these disadvantages will be overcome with improved laser equipment, but the problem of the periodic break in the channel will remain because the use of narrow beams means that each satellite must be followed in turn, one being acquired after the previous one has been relinquished before disappearing behind the Earth. It is impracticable to illuminate the whole path in which the satellites are seen. This path would be 3000 miles wide if the satellites were injected into an orbit with 22° inclination. Consequently, the preferred concept is that based on a high-altitude polar orbit.

Whether one or two satellites are used in this polar orbit arrangement depends on the requirements of the mission. If only one is used, then, for a considerable part of the mission, the plane of the orbit is only slightly inclined to the line-of-sight, causing communication to be interrupted every 5 hr when the satellite comes in line with, or is occulted by, Earth. If this condition can be accepted for the first 30 days of the mission, an orbit having a favorable aspect can be found for the remainder. The alternative is two satellites in separate orbits. In either case a suitable orbit is approximately circular with an altitude of 10,000 statute miles; a period of 10 hr; angular velocity, 0.01°/sec; and orbital velocity, 20,000 fps. Occultation by the moon, occurring at most twice each 29 days, lasts a maximum of 20 min.

Another concept having attractive features is represented at C, Fig. 2. This envisages an aircraft flying west in the upper troposphere, in the higher latitudes, with its ground speed equal and opposite to that of the Earth's surface below it. This aircraft would thus travel through space at Earth's orbital speed and from the spacecraft appear stationary in relation to Earth.

The advantages offered by this scheme include the simplification of the aligning process of the transmitters since the relative motion of the aircraft and space vehicle will be the same as the steady motion of Earth and a significant reduction in the amount of data required. It is applicable to either a northern or southern circum-polar track according to the inclination of Earth's axis. Choice of track must also take account of turbulence in order to minimize residual scintillation and aircraft disturbance. The performance of the DC-8 aircraft flying at 40,100 ft during the Douglas Aircraft Co.-National Geographic Society 1963 solar eclipse expedition [Project Aerial Photography of the Eclipse of the Quiet Sun. (APEQS)] demonstrated the probability that the stability demands of a rapid-response beam deflection tracker can be met.

Acquistion and Tracking

One obvious implication of the narrow beams inseparable from optical communication is the need for the transmitter to point its beam at the associated receiver with extreme precision which, because of the high relative velocities of the vehicles, means that precise compensation must be made for velocity, or Bradley aberration, and propagation, or transit time. The magnitude of the combined effects is often many times greater than conceptual beamwidths.

Velocity aberration and the related Bradley effect cause apparent displacement of a body moving relative to an observer. Transit time, up to 7 min in the Earth/Venus mission, requires a signal to be aimed ahead of the moving target for its interception. Figure 3a illustrates these effects: A_0, A_1, and A_2, are the successive apparent positions of an omnidirectional source A, in a plane normal to the line-of-sight, at times t_0, t_1, and t_2, such that $t_1 - t_0 = t_2 - t_1 = \Delta t$, Δt being the time taken for light to travel the distance AX. Thus if V is the across-range velocity of the source and c the velocity of light, the distances A_0A_1 and A_1A_2 are equal to $V\Delta t$, and AX is equal to $c\Delta t$. Photons reaching X_1 at time t_1, when A is at A_1, will have been emitted from A_0 at t_0. They will have traveled the path A_0X_1 and appear to come from A_0. Further, if at time t_0, X wishes to send a signal to A, this must be aimed ahead at A_2. In other words, when the target is apparently at A_0, an outgoing signal must be aimed at A_2. The angle of offset α is defined by the ratio of the velocities and is independent of distance; thus $\alpha = \tan\alpha = 2V/c$. It is clear that A must make a similar allowance when sending a signal to X, since to A, X seems to be moving in the opposite direction, X_0, X_1, and X_2, representing apparent positions corresponding to A_0, A_1, and A_2.

At this stage we are concerned with acquisition and tracking, and it is relevant to stress that it is a very different function from that of acquiring and tracking a star for spectroscopy or observation. Communication demands rapid acquisition and lock-on tracking, and yet one vehicle cannot acquire a second vehicle unless the latter has already correctly aimed its transmitter, which requires a knowledge of both the position and velocity vector of the first. One may then inquire how the operation of mutual acquisition starts. The answer is that some external reference and a planned operational sequence is required. The procedures for acquisition and tracking differ appreciably in the two directions, but they have much in common. They employ the common frame of reference indicated in Fig. 3b, together with the coordinate system used.

The procedure for the acquisition of a relay satellite depends on the spacecraft's locating the satellite with reference to Earth by using coordinates computed from stored ephemeris and trajectory data, and recorded time. This operation requires that the orientation of the vehicle be defined and a common frame of reference established. The most convenient reference for this mission is the plane through Earth, the space vehicle, and the sun. The optical axis of the receiving telescope on the space vehicle is therefore maintained coincident with the line-of-sight to Earth. Rotation about this axis is defined by reference to the sun. The use of Cartesian coordinates is preferred because, as will be apparent later, it simplifies the correction for aberration and transit time, since the satellite orbit will generally appear as an ellipse.

The relay satellite has no equivalent of the Earth to aid its acquisition of the spacecraft, but it is assumed that the previously described procedure will enable the spacecraft to aim its transmitter correctly, and that it will not fail to find the transmitter in the area of the sky defined by on board information.

The beamwidth under discussion calls for tracking accuracy of about 0.25 arc sec. It is believed that such accuracy cannot be maintained by vehicle attitude control alone. The implied response would cause excessive consumption of energy, and permissible errors might be exceeded by limit-cycle amplitudes or by the perturbations of a manned craft resulting from crew movements. These difficulties are overcome by a combination of telescope alignment with beam-deflecting techniques. Telescope alignment can be achieved most easily by attitude control when it is an integral part of the vehicle; otherwise the telescope must be slewed by a servo system, and the tendency of this additional control loop to degrade beam-pointing performance should not be overlooked. In either case the telescope control can be relatively leisurely. No unnecessary energy is expanded in cyclic motion of the vehicle, precise tracking being accomplished by beam deflectors with negligible time lag and power consumption.

Consideration must be given to the stage in the mission at which optical communication can usefully begin. Relative motions of the spacecraft and the relay satellites, or even ground stations, give rise to high angular velocities and wide angular displacements, and operation in early stages greatly aggravates the acquisition, tracking, and beam-pointing problems. A requirement for communication soon after launch can only be met in a manner that in later phases of the mission prejudices either tracking and beam-pointing performance, or reliability because of added complexity. For this reason acquisition will be assumed to start at some distance between 0.5×10^6 and 10^6 miles from Earth when the range of movement of the deflectors does not exceed $\pm 0.75°$. Once initiated, mutual acquisition proceeds according to the following sequence (an account of the mechanization is given later under "Optical Concepts"):

1) The space vehicle Earth tracker acquires Earth, and the telescope axis is brought into line by attitude control or slewing. At some stage in this process the tracking error falls below a predetermined level, and the second operation is initiated.

2) Control signals, generated by the computer from ephemeris data and the clock, cause the beam deflector of the tracker-receiver to direct continuously the line-of-sight to the predicted position of the relay satellite while the transmitter beamdeflector aims the beam as required by the satellite's velocity vector. Accuracy is not critical at this stage, since the constant-illuminated-area concept implies a beamwidth 80 times that used at the extreme range of 80×10^6 miles.

3) The relay satellite acquires the spacecraft transmitter by virtue of its broadened beam and proceeds to track it, assuming its "signature" modulation has been correctly identified.

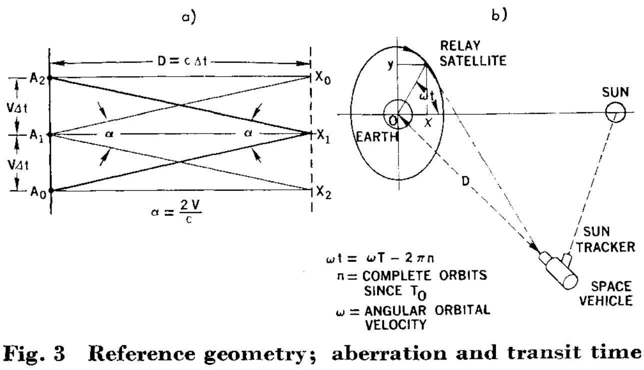

Fig. 3 Reference geometry; aberration and transit time errors.

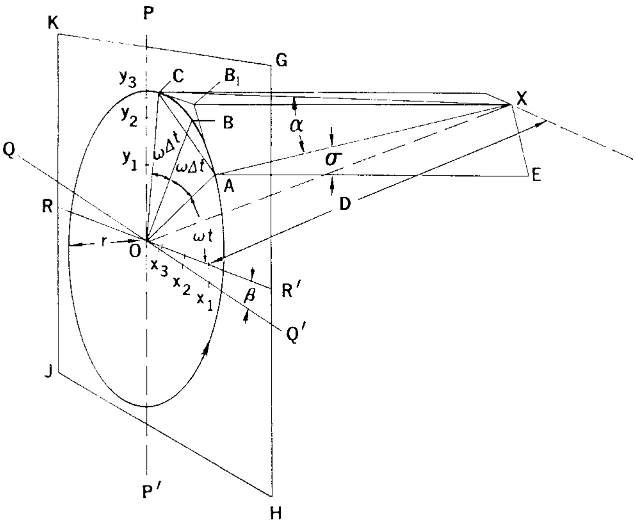

Fig. 4 Compensation for orbital motion.

4) The relay satellite directs its transmission to illuminate the space vehicle, the necessary correction being generated from ephemeris data.

5) When the space vehicle tracker-receiver is illuminated and the transmission identified, it is switched to the direct tracking mode in which the beam deflectors follow the relay satellite in its orbit while the telescope axis remains aligned with Earth. Communication can now proceed.

Subsequently, tracking is monitored by reception of the identification signal. This signal will disappear at times of occultation or conjunction. Under these circumstances, in the case of the space vehicle tracking the relay satellite, the situation remains under control with Earth being tracked by the coarse tracker. The orbital motion of the satellite during occultation by the moon is not sufficient to take it out of the tracker field. The acquisition should therefore be automatic; if not, the initial reacquisition routine could be followed.

On the other hand, if after occultation the relay satellite fails to reacquire the space vehicle, the reacquisition problem is more complex, and a reference point becomes necessary. In both instances the probability of reacquisition is increased if the vehicle is undisturbed during occultation. It should be possible to devise logic to insure this since the process of occultation will be gradual and detectable as such by the tracker-receiver circuitry.

Beam Pointing

If we refer again to Fig. 3a, it is apparent that unless the angle α can be accommodated within the transmitted beamwidth, it is necessary that a correcting offset be introduced between the tracked position of the target and the direction in which the laser transmitter is aimed. In view of other sources of error, it is considered that permissible error resulting from this effect should be less than one quarter of the beamwidth. It is therefore relevant to determine the order of magnitude of this angle for possible situations, and consideration is now given to the motion of Earth and satellites as seen from the space vehicle in the reference plane, and relative to the optical axis of its telescope. This point of view is taken for convenience only, since it is obvious that, to a satellite, the space vehicle will appear to execute equal and opposite motions. In other words, tracking and beam-pointing problems at both ends of a relay satellite-spacecraft communication channel are similar except for the following important difference: the center of the space vehicle's apparent orbital motion is not marked by the equivalent of Earth. The method of correction is explained by reference to Fig. 4.

A, B, and C represent successive positions, at time intervals, of Δt, of a target in Earth orbit, radius r. This will first be assumed to lie in the plane $GHJK$, normal to a line OX from Earth's center to a communicating vehicle X, with diameters coinciding with POP' and ROR'. Energy emitted from A and reaching X will travel in plane AB_1XE. Similarly, energy emitted from X when the target is at B must follow the vector XC if it is to intercept the target. Thus, when the target ap pears to be at A, the transmitter must be aimed ahead at C. It is apparent that $AC = 2r \sin\omega\Delta t$, and that $\alpha = 2r \sin\omega\Delta t/D$. Angle $B_1AC = \omega\Delta t$ also has to be taken into account. When these expressions are related to the 10,000-mile orbit, α is found to reach a maximum of 26 μ rad or 5.3 arc sec. Obviously compensation is necessary.

Cartesian coordinates are preferred because they correspond to the orthogonal motions of the beam deflectors and are obtained directly from the parametric equations of the orbit. This appears as a circle when its plane is perpendicular to the line-of-sight, the first satellite position A, Fig. 4, being defined by

$$x_1 = r \cos\omega t \qquad y_1 = r \sin\omega t \tag{1}$$

Correction for aberration and transit time is made when the space vehicle points its transmission to C, defined by

$$x_3 = r \cos\omega(t + 2\Delta t) \qquad y_3 = r \sin\omega(t + 2\Delta t) \tag{2}$$

The orbital plane will generally be at an angle β to the plane normal to the line-of-sight appearing as an ellipse with axes lying along POP' and QOQ'. In this case the apparent position would be

$$x_1 = r \cos\omega t \cos\beta \qquad y_1 = r \sin\omega t \tag{3}$$

and the point to which the transmission is directed is defined by

$$x_3 = 4 \cos\omega(t + 2\Delta t) \cos\beta \qquad y_3 = r \sin\omega(t + 2\Delta t) \tag{4}$$

However, in normal operation the satellite is tracked without reference to Earth, the offset vector AC being related directly to the position A. The coordinates of C transferred to this new reference become

$$\begin{aligned} x(A) &= r \cos\beta[\cos\omega(t + 2\Delta t) - \cos\omega t] \\ y(A) &= r[\sin\omega(t + 2\Delta t) - \sin\omega t] \end{aligned} \tag{5}$$

Assuming that the beam deflectors are linear devices, it is seen that the accuracy of the offset vector is unaffected by displacement of the telescope axis in the X and Y planes but is sensitive to rotation about these axes. In other words, accuracy of correction for velocity aberration and propagation time depends on accurate orientation of the vehicle about the telescope or optical axis by reference to the sun.

The compensation for these effects thus requires considerable facilities for computation and data storage. It is considered that the necessary data cannot be obtained with the required accuracy from tracking information. Since complete ephemeris data must be stored for the initial acquisition of the relay satellite by the spacecraft, no serious penalty is involved in updating stored data by normal microwave telemetry. This also calls for no extra facilities since complete dependence on optical communication with very narrow beams is clearly impracticable.

Attention is now directed to the problem of correctly aiming the beam in the opposite direction from the relay satellite transmitter to the space vehicle. The actual path in space of such a satellite is trochoidal, but this is not apparent to the satellite tracker on the space vehicle because its beams deflectors follow the satellite on its apparently circular or elliptical path whereas the telescope follows Earth's heliocentric motion. To the relay satellite, the space vehicle appears to execute equal and opposite motions that will appear trochoidal since, in this case, there is no equivalent to Earth to aid the separation of the circular and linear motions. However, some way must be found to separate the circular motion from the steady motion in the reference plane to prevent early saturation of the beam deflectors in the latter.

One possible approach is illustrated in Fig. 5. A_2, B_2, and C_2 represent successive positions of the relay satellite in its terrestrial orbit, and A_1, B_1, and C_1 represent equivalent apparent positions of the space vehicle. The proposed scheme is to determine, with computer equipment and data identical to that on the space vehicle, the theoretical position of the spacecraft relative to the center of its hypothetical orbit, to compare this with the actual deflection demanded of the tracking beam deflector, and to use the difference vector to control the satellite axis. With the axis of the satellite pointing at Q and the space vehicle transmitter tracked at A_1, QA_1 represents the deflection imparted by the beam deflectors, whereas if the satellite axis had been correctly pointing at R, the deflection would have been RA_1. Thus QR is the correction to be applied to the vehicle attitude, and, although some smoothing may be required, this information is quickly available. Once the correct attitude has been achieved, the method of correction for aberration and transit time is the same as that employed on the space vehicle, although a small error will remain to generate the necessary signal to cause rotation of the vehicle in the orbital plane.

Optical Concepts

The optical functions on which acquisition, tracking, and beam pointing are based have been indicated in previous paragraphs. It is now desirable to draw these together in a physical concept if the implications of astronomical range optical communication are to be seen in their true perspective. The essential features of such a concept are illustrated in Fig. 6.

The main unit is the tracker-receiver telescope; its chief feature is a large parabolic mirror that is the receiving telescope objective. The remainder of the optics are mounted as one unit on the telescope axis. This unit comprises a tracker-receiver pointing toward the mirror and in tandem with it and the laser transmitter pointing in the opposite direction. This construction promises the rigidity necessary to withstand launch and other accelerations. The tracker-receiver includes a beam deflector providing rapid response in tracking. It is also furnished with a narrow-band optical filter matched to the transmission to be received.

External to the telescope and with its axis parallel is the primary tracker whose function is to control the telescope alignment. An orthogonally mounted pair of single axis beam deflectors enables the image of a bright source in the field to be centered on the receiver without complete alignment of the telescope. This deflector pair follows the motion of the satellite as it moves relative to Earth. The transmitter is also furnished with a similar two-axis deflector pair.

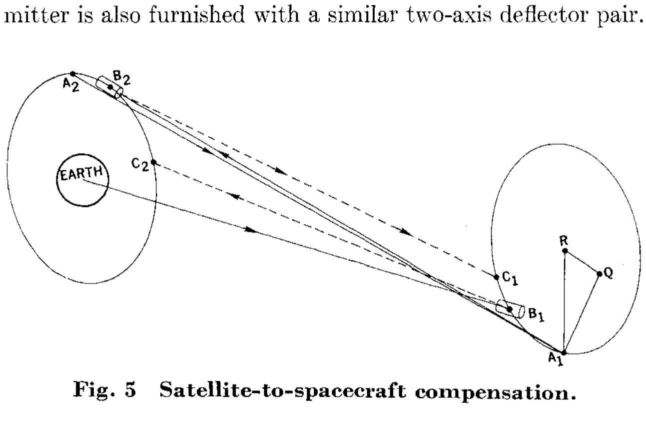

Fig. 5 Satellite-to-spacecraft compensation.

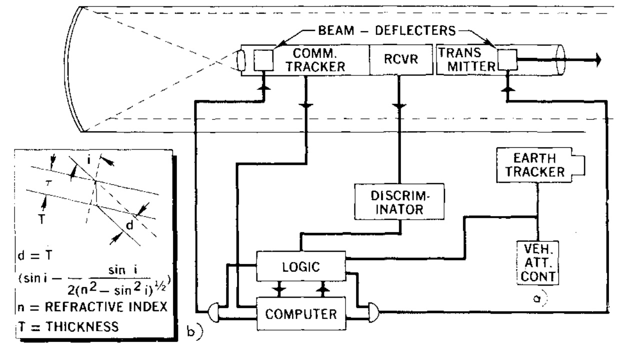

Fig. 6 Beam-pointing functional concept.

This pair directs the beam to compensate for aberration, transit time, and misalignment, and its operating signals are computed from tracker-receiver information and ephemeris data.

Beam deflectors may take various forms. Electro-optical solid-state devices of suitable range would be most attractive if they were available. The alternatives are refracting or reflecting devices that fortunately can be small enough to be embodied in practical mechanical designs capable of rapid response, require little power for their operation, and are compatible with the space environment. One refracting type uses parallel sided prisms who seinclination relative to the optical axis causes lateral displacement of the image. Figure 6a implies the use of a preferred arrangement one gimbaled prism. The relationship between beam displacement, angle of inclination, and thickness is shown in Fig. 6b. Linearity is important for accurate correspondence between tracker and transmitter. The device shown is linear within 1% up to 15° deflection. The change in optical path length with inclination is negligible in the configurations contemplated. This design may be the more complex mechanically, but losses resulting from reflection occur at two faces only.

Whatever form the device takes, wide dynamic range is required. Threshold sensitivities of 0.1 to 0.2 arc sec appear desirable, whereas tracking of a satellite in a 10,000-mile polar orbit from a distance of 10^6 miles calls for a deflection range of $\pm$ 0.75°. Difficulties in maintaining correspondence between transmitter and receiver beam deflectors over such an angle within fractions of an arc second are to be expected. However, the situation is not as critical as it appears because at short range transmitter beamwidths can be increased.

The operation of this concept will be best understood by following through the sequence on a spacecraft establishing communication with a relay satellite in a 10,000-mile polar orbit. It is assumed that Earth has been acquired either through a wide-angle tracker or other reference, and the Earth tracker in the telescope is generating error signals. These will first pass to the vehicle attitude control system where, after integration, they will cause the image of Earth to be centered in the Earth tracker. As these signals gradually decrease, a moment arrives when the Earth tracking error combined with the displacement of the satellite relative to Earth (as determined from ephemeris data) falls within the range of the tracker-receiver beam deflector. This moment is determined within the computer whose logic then causes the signals to be switched to the beam deflector. This should bring the satellite into the field of the tracker-receiver, in which case the characteristic modulation of the transmission will be "recognized" by the discriminator. This recognition initiates the next step, the connection of the beam deflectors to the receiver-tracker, closing this self-contained loop that will continue to track the satellite. Thereafter the operation is independent of ephemeris data except for reacquisition after occultation or equipment failure, which is indicated by loss of the discrimination signal.

Ephemeris data is still required, however, for the direction of the transmitter beam to compensate for transit time. The details of this operation were discussed earlier under "Acquisition and Tracking." At this time it is sufficient to note that this compensation is applied to the position of the satellite as "seen" by the tracker-receiver.

The preceding concept is applicable also to the relay satellite tracking the space vehicle, except that there is no need for the Earth tracker as such, although some form of wide-angle tracker may be necessary to aid acquisition of the spacecraft. Clearly, the equipment described is bulky and heavy, and considerable power requirements will add to system weight. It appears improbable that the efficiency of gas lasers can be greatly increased so that acceptable over-all efficiency depends on achieving the high gain possible with optical "antennas." The injection diode laser shows great superiority over the gas laser in efficiency, but the extended nature of the source raises doubts concerning the possibility of obtaining the same precision of collimation as with a gas laser.

No attempt is made to discuss conceptual forms of the tracker-receiver since designs having the features already mentioned as essential to successful operation of a laser channel, particularly laser predetection amplifier and its optics, are likely to be revolutionary and bear little relationship to existing designs using postdetection amplification.

Appendix

Estimation of Laser Channel Performance

The following symbols are used in this discussion (although it is difficult to justify any particular numbers as being representative of the state of the art, the values in parentheses are used in estimating current possibilities):

P_T = total laser transmitter power, w (0.1)
P_R = receiver power, w
I_R = intensity at receiver, w/cm^2
Υ_T = transmitter transmissivity (0.2)
Υ_R = receiver efficiency or transmissivity (0.5)
d = diameter of receiver optics, cm (100)
R = range from transmitter to receiver, cm (1.48×10^{13} max)
θ_T = transmitter beam divergence, rad (5×10^{-6})
H = information rate, bits/sec
B = information bandwidth, cps
h = Planck's constant, w/sec^2 (6.6252×10^{-34})
ν = carrier frequency (2.6×10^{14} for 11,523 Å)
N = rate of photon arrival, photon/sec
N_S = rate of effective signal photon arrival, photons/sec
n = photons arriving in period $\frac{1}{2}B$
n_s = effective signal photons arriving in period $\frac{1}{2}B$
S = signal/noise power ratio (30)

Only a fraction (Υ_T) of P_T will be available as signal power depending on the method of modulation, the inherent absorption of the modulator, and intensity variations within the beam. Then the signal power intensity at the receiver is

$$I_R = 4\Upsilon_T P_T / R^2 \theta_T{}^2 \pi \tag{A1}$$

and the effective signal power P_R is

$$P_R = d^2 \Upsilon_R \Upsilon_T\, P_T / R^2 \theta_T{}^2 \tag{A2}$$

Noise is inherent in this quantized radiation because of the uncertainty of the number of photons arriving in each integrating period. In communication theory this period is $\frac{1}{2}B$, thus $n_s = N_S/2B$. Statistical theory states that the possible variation from the mean is $n_s{}^{1/2}$; consequently, if all other sources of noise are neglected, the minimum signal/noise power ratio can be established as

$$S = (n_s / n_s{}^{1/2})^2 = n_s \tag{A3}$$

In the presence of background noise, the expression becomes

$$S = [n_s{}^2/(n_s + n_n)] \tag{A4}$$

It is obviously advantageous for n to be as large as possible for the same power output, implying low values of ν to be preferable, and this factor makes the *He-Ne* laser attractive. Since photon energy = $h\nu$, N_S can be determined from P_R for any particular frequency by substitution in Eq. (A2), and it becomes possible to relate this to channel information rate through the classical formula

$$H \rightarrow 2B \log_2(S + 1) \tag{A5}$$

in which B is established from the specified channel signal/noise ratio in the absence of external noise. Thus an approximation maximum for H can be expressed in terms of received signal power and background luminosity. The figures in Table 1 were arrived at by substituting given values in the foregoing appropriate equations.

Data in Table 1†

The data in this table relating to Earth and moon are based on a value for the solar flux outside the atmosphere of 58 ergs/cm^2-Å-sec at 11,000 Å. The Bond albedo of Earth, the ratio of the total light reflected from it to the total light incident on it, equals 0.34. The area on Earth disk is 1.28×10^{18} cm^2. The total reflected flux in a 10 Å band centered on 11,000 Å is 7.5×10^{20} ergs/sec = 7.5×10^{13} w. Using the channel parameters listed previously, for which $h\nu = 1.75 \times 10^{-19}$ joule, and $2B = 122$, it is found that the maximum rate of photon arrival N at the detector is $N = 4.3 \times 10^8$ photons/sec; and $n = 3.5 \times 10^6$ photons in interval $\frac{1}{2}B$ sec. Assuming a quantum efficiency of 100% (although it can fall below 10% in a photomultiplier), $S = 900/(900 + 3.5 \times 10^6)^{1/2} = 0.48$, and $H = 122 \log_2 1.48 = 70$ bit/sec.

In the case of the moon, the Bond albedo is 0.07, and the disk area is 9.4×10^{16} cm^2. It is thus determined in a similar manner that the total reflected flux is 5.5×10^{11} w (in the same 10 Å band). The maximum rate of photon arrival at the detector is $N = 1.05 \times 10^7$ photons/sec. The remaining figures appearing on line 3 of Table 1 are $S = 2.8$ and $H = 230$ bit/sec. With regard to Venus, the spectral radiance of a zero magnitude star I_0, at 10,000 Å, above the atmosphere is given as 0.41×10^{-15} w-cm^2-Å. Venus has a visual magnitude of -4.08; therefore $I_{\text{Venus}} = I_0 \log^{-1} 1.65 = 1.8 \times 10^{-14}$ w/cm^2-Å from which the data on line 4 of Table 1 is derived.

Day Sky

The spectral radiance of the day sky is stated to be 2.5×10^{-8} w/cm^2-sr-Å at 10,000 Å[1]. An extrapolated value of 2.0×10^{-8} at 11,523 Å will be assumed for the purpose of this calculation. The angular field of the telescope must also be specified, and a circular beam 10 min wide or 6.6×10^{-6} sr is selected as representative.

From these figures it is determined that the radiation reaching the detector through a 10 Å filter is 1×10^{-8} w, and $N = 6 \times 10^{10}$ photons/sec, $n = 8 \times 10^8$ photons in interval $\frac{1}{2}B$ sec, and $S = 0.01$.

References

[1] Bayley, D. S., "Technical note on optical communication I," GPL Div., General Precision Inc., Pleasantville, N. Y., Rome Air Development Cente r, Griffiss Air Force Base, Rome, N. Y.,

† Unless otherwise indicated, data used in the following calculations are taken from Ref. 3.

RADC-TN-61-117, Contract AF 30 (603)-2203 (May 1961).
[2] Peterson, A. M. (ed.), "Upper atmosphere clutter research, Part XIII: Effects of the atmosphere on radar resolution and accuracy," Stanford Research Institute, Menlo Park, Calif., Rome Air Development Center, Griffiss Air Force Base, Rome, N. Y., RADC-TR-60-44, Contract AF 30 (602)-1762 (April 1960).
[3] Allen, C. W., *Astrophysical Quantities* (The Athlone Press, London, 1963).

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Rendezvous Guidance of Lifting Aerospace Vehicles

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An explicit guidance method is described for space vehicles that 1) use atmospheric forces to permit maneuvers during ascent to orbit; 2) ascend directly to an orbital rendezvous; and 3) perform a hypersonic, in-atmosphere cruise before light-off of rocket-powered ascent to orbit. Hypersonic cruise capability will give the vehicle great flexibility to reach targets because it can, in effect, move its launch point relative to earth. The guidance system, however, must accept variations in position, heading, and time of rocket ignition and still must complete the rendezvous. This method can accept the moving-rocket ignition point because place and time of rendezvous are not specified in advance but depend upon the geocentric lead angle between vehicle and target at ignition. Guidance constraint of slope at injection allows reasonable variation in time of ignition, with near-minimum fuel penalty. Simulation results for a representative future lifting ascent vehicle show allowable variation in position at rocket ignition of 80 miles on either side of target orbit plane, with allowed time variation of about 3 min at any position. The vehicle can start farther from the target plane by making a maneuvering ascent in a different plane and then a dog-leg.

Introduction

THE class of future aerospace vehicles that will make use of aerodynamic forces instead of minimizing their effects during ascent to orbit is appearing ever larger on the horizon. These vehicles may evolve from hypersonic aircraft that extend their region of operation to orbit either directly or with staging, or they may embody new design concepts. They have at various times been called recoverable or reusable boosters, or simply aerospace planes. A typical ascent for such a vehicle will begin with a hypersonic in-atmosphere cruise that brings the vehicle to the light-off point where rocket-powered ascent begins (Fig. 1). The vehicle can maneuver during the rocket-powered ascent in order to arrive at a cutoff point from which it will coast to its destination in orbit.

The reusable characteristic will give this type of vehicle economy that will motivate its development for future high-traffic-density missions such as logistic shuttles. Equally important, the in-atmosphere cruise capability and the maneuver during rocket ascent will give it considerable flexibility to reach any of a large number of orbital destinations directly from the same starting point. This will make the vehicle particularly suitable for quick reaction missions such as rescue and satellite inspection.

This paper describes a method for explicit guidance during the maneuvering rocket-powered ascent, for vehicles and missions having these characteristics: 1) use of aerodynamic lift to maneuver during ascent, 2) ascent directly to an orbital rendezvous, and 3) use of hypersonic cruise prior to the rocket-powered ascent. These vehicle-mission characteristics, shown in Fig. 1, have a strong impact on the guidance requirements.

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When a vehicle is launched from a fixed site on earth to reach an orbital target, without a hypersonic cruise, fuel economy generally dictates that the launching wait until the earth's rotation brings the launch site close to the target orbit plane. If the target is not then in a favorable location in its orbit, the ascending vehicle will enter a parking or phasing orbit. If a direct ascent to the target is required, the launching may have to take place when the launch site is some distance away from the target orbit plane in order that the ascending vehicle will meet the target upon arrival in orbit; the velocity increment required to turn into the target orbit plane may be considerable. References 1–4 are among the papers that discuss rendezvous starting from earth-fixed launch sites.

A vehicle capable of hypersonic cruise before rocket ignition will generally direct its cruise path so as to have one component toward the target orbit plane and another in the direction of the target velocity in orbit. That is, prior to the rocket ascent, the vehicle will move its launch point or point

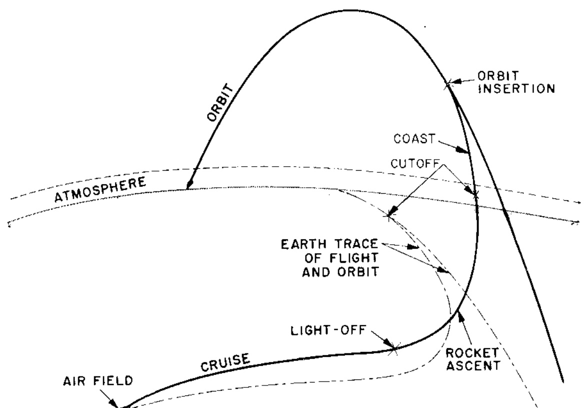

FIG I PHASES OF LIFTING ASCENT

Fig. 1 Phases of lifting ascent.