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SYSTEM CONCEPT STUDY - VOLUME I

CONCEPT ALTERNATIVES (U)

21 MAY 1968

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1.0 INTRODUCTION

Significant improvements in cost, timeliness, and operational flexibility are achievable in a photographic reconnaissance system characterized by a long-life observational satellite with electromagnetic data return. Such a reconnaissance system will rely on one of several opto-electronic transducers to eliminate the need for the expendables associated with a film recovery system. The launch rate of the system will be greatly reduced by maintaining the observational satellite on orbit for an extended period of time. Rapid return of data via a communication link will increase the assurance of acquiring the desired data on a timely basis and will avoid redundant acquisitions. This, and the reduction in peak work loads, will result in improved efficiency of data processing and photo interpretation facilities.

The key elements of the system are a transducer to convert images to electrical signals and a means for transmission of the video data from the observation spacecraft to a ground station. Sensitivity and resolution of image transducers are fundamental state-of-the-art limitations on achievable performance. State-of-the-art constraints on bandwidth for the transmission of video data limit the information flow rate obtainable and therefore the image capacity. Within these constraints a number of system concepts appear to be technically feasible for the level of performance required for a strategic surveillance mission. Other missions, requiring higher performance, are achievable within the growth potential afforded by foreseeable technological advancements.

The study reported in this document explores conceptual approaches to systems for the electromagnetic return of reconnaissance imagery. A number of representative system concepts are chosen for conceptual design examples, but no attempt is made to select a "best" system concept.

The study report consists of four volumes. Significant results of the study appear in Volume I. Backup material is compiled in Volume II. Technical problems associated with development and growth of reconnaissance systems with electromagnetic data return are discussed in Volume III. Volume IV is devoted to an evaluation methodology that could be applied in the future selection of a concept for implementation.

The material presented in Volume I covers the following principal topics:

a. Definition of system functions and characteristics (Section 2).

b. Identification of alternative concepts for implementation and selection of representative baseline concepts (Section 3).

c. Review and analysis of mission requirements to establish quantitative measures of the desired system characteristics (Section 4).

d. Assessment of hardware capabilities and state-of-the-art limitations for each of the potential subsystems (Section 5).

e. Quantitative identification of internal trade relationships that must be observed in making system design choices (Section 6).

f. Assignment of performance requirements for each of the systems (Section 7).

g. Definition of point design baseline system examples for each of the selected concepts (Sections 8 through 14).

2.0 SYSTEM FUNCTIONS AND KEY PARAMETERS

In seeking to identify alternative conceptual approaches, the system should be viewed both from the implementation viewpoint of defining functions and from the mission viewpoint of identifying critical system characteristics. Functions having a major impact on system characteristics must receive emphasis in the search for alternative mechanizations.

2.1 FUNCTIONAL FLOW DIAGRAMS

A useful framework for examining system concept alternatives is the functional flow. Functional flows depict each specific system function and its relationship to other system functions. A completed set of flows provides a thorough functional description of the system and shows, for a given function, which functions must precede it, which must follow and which are independent. The top flow, depicted in Figure 2-1, provides a gross functional breakdown of the system. Each of the top functions indicated can be expanded into more detailed flows. Expanded functional flows with alternate implementations for the major inline operational functions shown in Figure 2-1 appear in Volume II, Section 3.

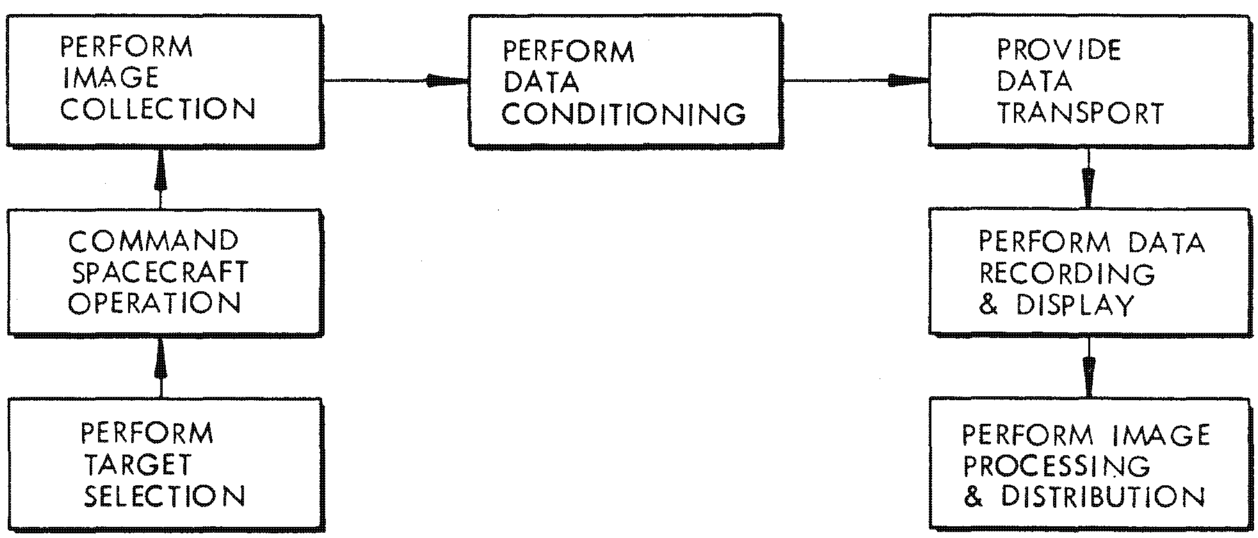

FIGURE 2-1 SYSTEM FUNCTIONAL FLOW

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The Perform Target Selection function of Figure 2-1 is a ground station function including inputing targets and priorities, obtaining and maintaining system status data, obtaining target cloud cover (weather) data, defining, scheduling and updating (as required) a use schedule for the observation satellite (i.e., what targets are to be acquired at what times), and performing a mission simulation to assure the validity of the developed schedule data.

The Command Spacecraft Operation function includes command simulation, command transmission from ground control to the satellite(s), the command reception at the satellite(s) and verification, command acceptance, and the initiation of required functional responses.

The Perform Image Collection function is the observation satellite function of acquiring specific images and includes performing the properly timed attitude control (line-of-sight orientation) functions and image transducing functions.

The Perform Data Conditioning function is the observation satellite function including developing transduced imagery if required, providing a store for imagery data for store-and-forward system concepts, and generating and conditioning (e.g., stored imagery scanning, analog-to-digital conversion, encryption) imagery data signals as required for input to the rf communication subsystem.

The Provide Data Transport function includes the acquisition and tracking of a relay satellite or the ground station by the observation satellite, transmission of imaging data, acquisition, tracking and relaying signals from the observation satellite to the ground station by the relay satellite, and ground station reception and demodulation.

The Perform Data Recording and Display function is a ground station function including recording data in direct electrical form if required, decrypting and converting imagery data from digital to analog data if required, recording the images on film, providing an imagery monitor, and transporting the recorded imagery data to a processing station, should the latter not be colocated with the ground data reception facility.

The Perform Imagery Processing and Distribution function includes screening imagery for useful data, identifying and labeling data as required, and distributing the imagery to the exploitation center(s).

The second-level flow diagrams define many more implementation alternatives than can be covered in this early search for system concepts. Therefore, primary attention is devoted to the three functions having greatest impact on the system characteristics: a) Image Collection, b) Data Conditioning, c) Data Transport. However, the conceptual designs of the baseline systems cover all system functions. Later sections will treat Image Collection as two functions, image transducing and pointing control. In considering alternative concepts, consideration of Data Conditioning will be deferred on the grounds that it is a consequence of the concept selected rather than the determinant.

2.2 CRITICAL FACTORS

System characteristics of importance to a long-life orbital system with electromagnetic return of imagery data are summarized in the following paragraphs. Most of these factors are given quantitative treatment in Section 4 and in Volume II.

Imagery Parameters — Ground resolution, ground frame size, and the scene characteristics to be accommodated are fundamental system design parameters determined by the mission objectives.

Frequency of Access — The allowable time interval between acquisitions of a given aiming point can vary widely with the mission. Demand acquisition of random targets may require frequent access even though the number of images acquired is small.

Data Timeliness — Two time factors are evident: 1) the response time in acquiring a given aim point subsequent to a receipt of a request for imagery - this is closely related to frequency of access; 2) data delivery delay time from acquisition until imagery is available to the user - this includes time for reproduction and distribution in addition to time for data transport and imagery reconstruction.

Image Capacity — Each photographic image represents a very large amount of data. The total number of images to be acquired is of critical importance to a system with electromagnetic data return.

Transmission Time — The rate at which the imagery data must be transmitted and processed is a function of the available transmission time. The transmission time, in turn, is a function of the data storage and transport concept.

Peak Acquisition Rate — In addition to the number of images to be acquired, the time interval during which the targets appear is of importance. The system must be capable of a high acquisition rate if the aiming point set contains concentrations of targets.

Vulnerability/Data Security — The principal susceptibility to hostile interference lies in the communication links. In addition to a jamming threat, electromagnetic data return runs the risk of imagery data interception.

Lifetime/Reliability — System operation without expendables is a significant benefit of electromagnetic data return. However, realization of this benefit demands long life subsystems of reliable, proven design.

Availability — A trade exists between performance available from the subsystem hardware and the time period in which the system is to be operational. The concepts open for consideration and the system characteristics attainable for each concept are functions of the development time allowed.

Growth Potential — Some concepts may offer significant growth in performance by upgrading or retrofit of certain subsystems on later missions as technology developments become available. Such potential is a consideration in the selection of a concept for early implementation.

3.0 ALTERNATIVE SYSTEM CONCEPTS

Image transducing, data transport and pointing control have been identified as the functions having greatest impact on system characteristics. Alternative approaches to accomplishing each of these functions are discussed in Sections 3.1, 3.2 and 3.3. Section 3.4 considers the possible system combinations available from these alternatives. A group of representative system concepts is selected to serve as baselines.

3.1 IMAGE TRANSDUCER CONCEPTS

The transducer alternatives include four devices. Three are applicable to a frame imaging system. The fourth is a quasi-linear array of detectors used in a "broom-scan" mode. These two basic approaches are illustrated in Figure 3-1.

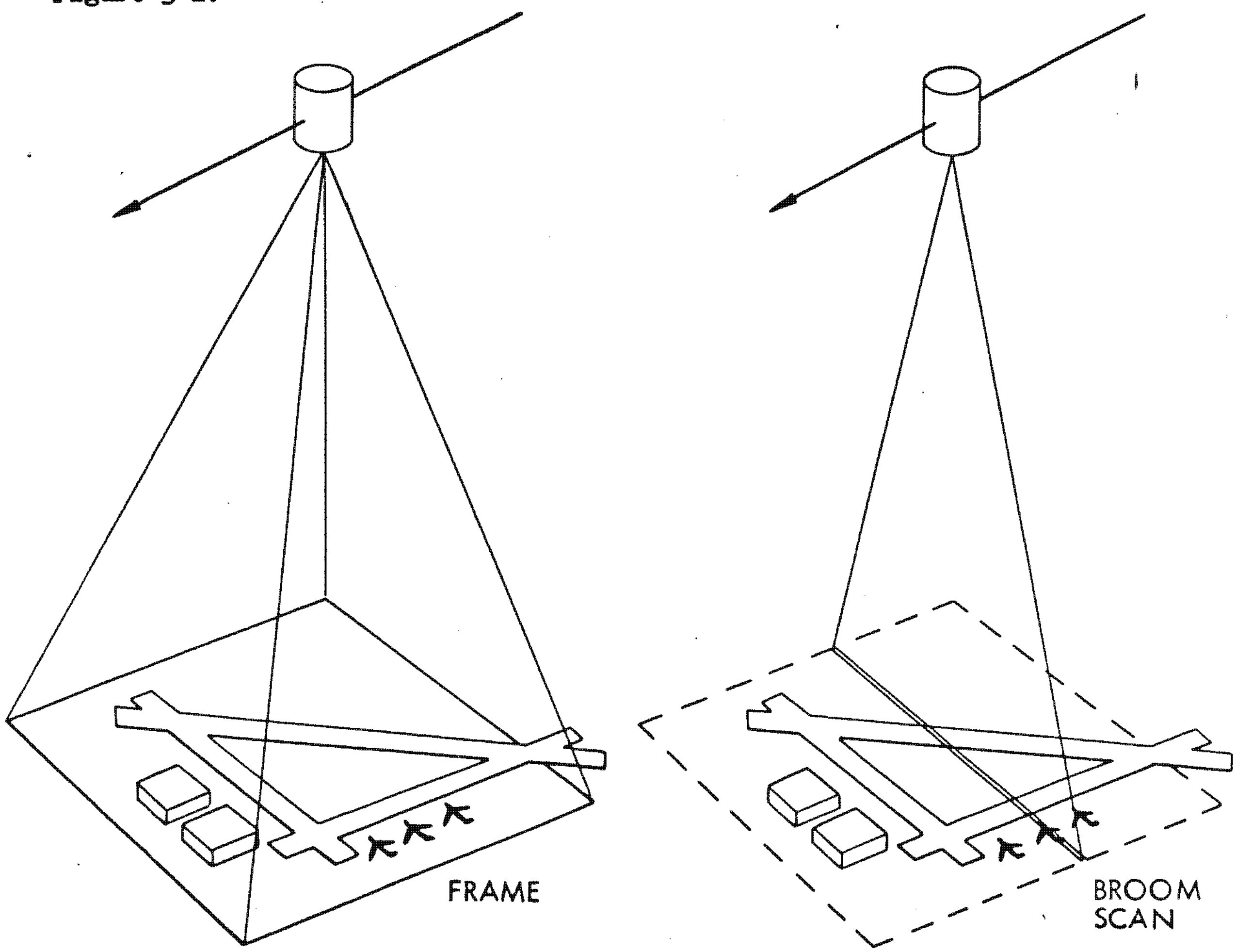

FIGURE 3-1 IMAGE ACQUISITION CONCEPTS

Screened Thermoplastic Film — This transducer is composed of a photoconductive layer and a thermoplastic layer on a suitable base material. An optical image causes the precharged potential across the film to selectively decay through photoconduction. The electrostatic latent image is converted to a proportional surface deformation upon the application of heat. The deformographic image can be stored in the thermoplastic layer for extended periods of time. This image is read out using a phase-sensitive optical scanning technique to provide an appropriate electromagnetic signal for transmission to the ground station. The thermoplastic material can be erased for reuse by a second application of heat.

Return-Beam Vidicon — The return-beam vidicon is a photoconductive imaging device which generates a video signal in response to an optical image. The vidicon offers limited single-frame storage. The vidicon photoconductor will retain an image for several seconds, a storage time inadequate for the store-and-forward concepts. However, this storage can be used as a "buffer" in a direct-readout concept to enable the use of an achievable communication bandwidth. Following readout by electron-beam scanning, the vidicon image is erased and prepared for reuse.

Reconotron — The Reconotron incorporates the features of an electro-optical camera in a dielectric-tape storage configuration. It incorporates many of the performance characteristics of the vidicon with the addition of multiple-frame storage for extended periods of time. It therefore has application in a store-and-forward concept. Conceptually, the Reconotron is used in the same fashion as the screened thermoplastic.

Array of Phototransducers — A one-dimensional array of discrete photodetectors provides a means of implementing a "broom-scan" system. The detectors are arrayed in the crosstrack direction, giving an instantaneous view of a narrow ground strip perpendicular to the ground track with a length equal to the ground frame width. The strip scans the ground in the direction of the velocity vector to generate an electrical equivalent of an optical image. The array of detectors is electrically sampled three times during the time the array is moved a distance of one resolution element on the ground. The output of the array is electrical and must be transmitted in real time or stored in an auxiliary storage device. The scan rate can be adjusted by controlling the optical line-of-sight rate. The bandwidth requirements imposed on the imaging subsystem and the communication or storage subsystem can be reduced by use of a slowed scan. The increased integration time with a slowed scan also permits operation with smaller optics and/or reduced illumination.

3.2 DATA RETURN CONCEPTS

Three alternative methods have been identified for return of the photographic data to the ground control station. They are depicted by the sketches in Figure 3-2.

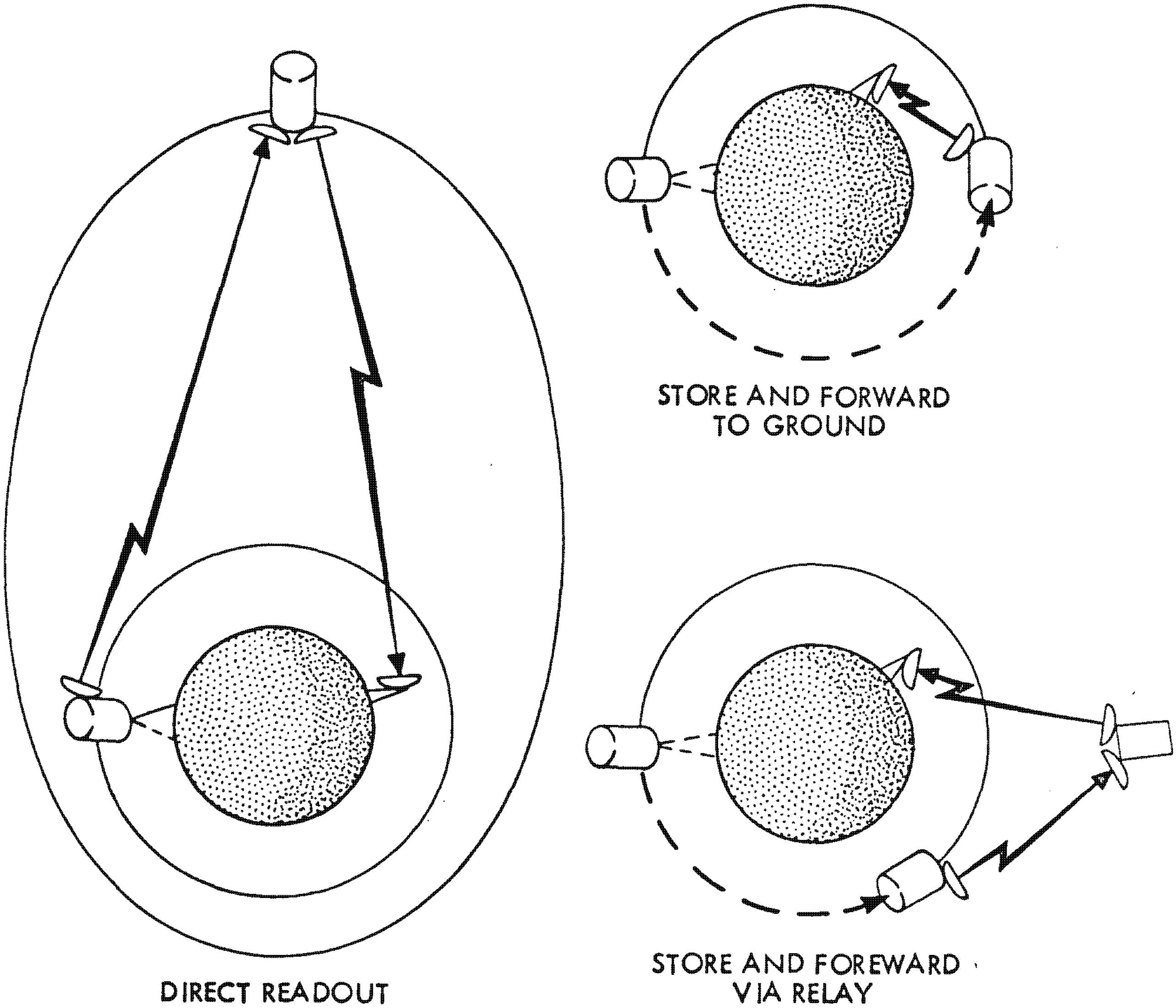

FIGURE 3-2 DATA TRANSPORT CONCEPTS

Direct Readout - In this concept, imagery data is returned immediately to the ground station. One or more relay satellites are required to provide a link from the observation satellite over the target to the ground recovery station within the contiguous United States. The cost and technical difficulty of providing this link are related to coverage requirements of the system. For example, one relay satellite in an elliptical 24-hour orbit with apogee above the north pole can provide a link from an observation satellite anywhere over the Sino-Soviet bloc (during local sunlight periods) to a ground station within the contiguous U.S.A. However, no coverage of large portions of the rest of the northern hemisphere and most of the southern hemisphere would be provided. To obtain a world-wide coverage with direct readout, three relay satellites in equatorial orbit and a more complicated link (including a relay-satellite-to-relay-satellite link) would be required. A two-relay satellite system provides coverage between these two extremes. Unfortunately, the spectrum of coverage vs. complexity of relay links is not well suited to growth, since a limited-coverage one-relay system cannot be improved by simply adding one or two more relays. Rather, the one-relay communication system must be replaced with a more complex multiple-relay system, with relays in markedly different orbits. A detailed discussion of relay orbit alternatives is presented in Volume II, Section 5.

Store and Forward to Ground — The imagery data is stored onboard the observation satellite with readout from storage direct to the ground recovery station when the latter is within direct communication range of the satellite. No relay satellites are required, and world-wide coverage is provided. However, the time delay between data acquisition and data return may be as much as 12 hours.

Store and Forward Via Relay Satellite — Stored data is read out from the observation satellite through a relay satellite to the ground station. Unlike the direct-readout alternative, the data-return link need not be established at the time of image acquisition. This concept requires onboard data storage and introduces a delay between data acquisition and data return of up to one-half hour. However, world-wide coverage is provided with a single, synchronous-altitude relay satellite.

It can be seen that the data-return concept selected can have a strong impact on system coverage, data return time, system complexity and growth capability. Also, the susceptibility of the system to electronic countermeasures (ECM) is determined primarily by the data-return concept. Direct-readout systems offer the ultimate in data recovery speed. However, they offer the least ECM resistance and the most difficult design problems. The necessary high data rates and long-range inter-satellite links require the use of high-powered transmitters and large antennas which in turn require accurate attitude control and beam steering.

Security of the inter-satellite link will be low unless encryption is used. The use of a [redacted] carrier will offer some protection but requires advancements in power amplifiers and receivers.

The store-and-forward-through-relay concept has the advantage that readout need be performed only over the western hemisphere. This tends to reduce the vulnerability of the links. The long distance inter-satellite link is still difficult to implement although less so than in the direct-readout system. The store-and-forward-to-ground concept is easiest to implement and is least vulnerable since all communications are performed within line of sight of ground stations within the contiguous U.S.A.

3.3 POINTING CONTROL CONCEPTS

Two alternatives have been defined for pointing control, as shown schematically in Figure 3-3.

Spacecraft Orientation — When the optics and associated structure dominate the spacecraft moment of inertia, orienting the entire spacecraft to the target is a reasonable approach. During periods when no imagery is being acquired, the spacecraft can be rotated either to an attitude that minimizes drag or to a fixed inertial reference attitude to eliminate the requirement to provide gimbals for primary navigation sensors such as a star tracker or a sun tracker.

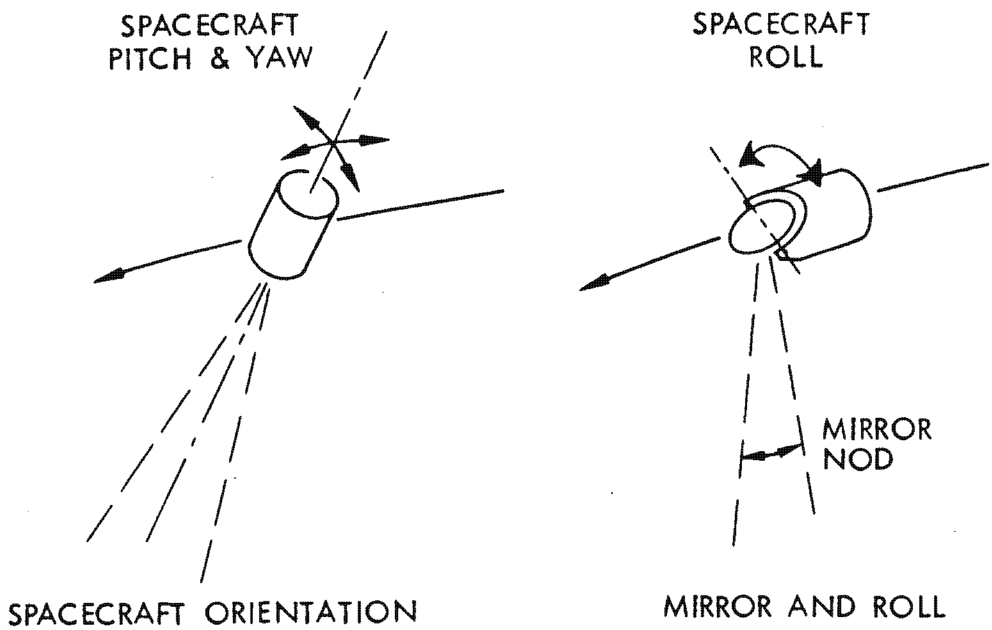

FIGURE 3-3 POINTING CONTROL CONCEPTS

<u>Mirror and Spacecraft Roll</u> — In this concept the spacecraft optical axis is held horizontal and in the orbital plane. A full-aperture, flat, folding mirror is oriented at 45 degrees with respect to the spacecraft optical line of sight. The mirror folds the line of sight 90 degrees to a vertical direction. In-track rotation of the line of sight with respect to vertical is obtained by rotating the folding mirror to an angle greater or less than 45 degrees. Crosstrack rotation of the line of sight with respect to vertical is obtained by rolling the spacecraft.

This alternative has two advantages over the spacecraft-orientation alternative. Drag is minimized because the spacecraft is streamlined, and the torque required for line-of-sight rotation (sensor pointing) is significantly reduced. Sensor pointing requires rotation of the mirror and rotation of the spacecraft about its axis of minimum moment of inertia, rather than rotation of the spacecraft about the two axes of high moment of inertia.

3.4 SELECTED SYSTEM CONCEPTS

Each combination of an image transducer, a data-transport mode and a pointing control method constitutes a potential system concept. Even a small number of options for each function leads to a large number of candidate system concepts. By grouping the three frame-type image transducers together, the number of system concept alternatives is reduced to twelve, as shown by Table 3-I. Entries in the table identify six baseline concepts selected for development to the level of conceptual system definition.

TABLE 3-I SYSTEM CONCEPT ALTERNATIVES

DATA TRANSPORT	POINTING CONTROL	IMAGE TRANSDUCER	
		FRAME	BROOM SCAN
DIRECT READOUT	SPACECRAFT ORIENTATION	VIDICON	
	MIRROR AND ROLL		QUASILINEAR ARRAY
STORE AND FORWARD TO GROUND	SPACECRAFT ORIENTATION	THERMOPLASTIC	QUASILINEAR ARRAY
	MIRROR AND ROLL		
STORE AND FORWARD VIA RELAY	SPACECRAFT ORIENTATION	RECONOTRON	
	MIRROR AND ROLL		QUASILINEAR ARRAY

All three of the frame-type image transducer candidates are represented. Direct readout is used with the vidicon, as it does not provide the long-term image storage that is inherent in the screened thermoplastic and Reconotron transducers. Use of the broom-scan transducer with the store-and-forward concepts implies a magnetic tape recorder for onboard storage.

The full-aperture, flat-folding-mirror-and-spacecraft-roll method of pointing control is not used with the store-and-forward-to-ground transport mode. The inherently short communication time for this concept suggests operation at high altitude where a broadside attitude does not impose a severe drag penalty.

In Section 7 each of these selected system concepts is assigned specific constraints on ground resolution, frame size and operating altitude to establish baseline system design points. Command and navigation methods are also selected for each baseline system.

4.0 MISSION REQUIREMENTS ANALYSIS

This section discusses the derivation of quantitative performance requirements related to the critical system characteristics. A more complete treatment of the analyses summarized here can be found in Volume II.

Particular attention is devoted to the specific mission of strategic surveillance. Requirements defined for purposes of the present study are as follows:

a. The system will have electromagnetic return of black-and-white imagery from the observation spacecraft to the ground recovery station. The latter will be in the contiguous United States, preferably in the [redacted] area.

b. The time lag from imagery acquisition until ground recovery will be no greater than 12 hours.

c. The observation spacecraft will have both a high-resolution and a low-resolution capability. The low-resolution capability will be so implemented as to permit the acquisition of both a high- and a low-resolution image on a given pass over an aiming point. The low-resolution image will include the high-resolution field of view.

d. The high-resolution imagery will have a ground resolution of 0.6 to 1.2 meters. Low-resolution imagery will have a ground resolution of 1/5 the high resolution; i.e., 3 to 6 meters. Resolution is defined as that obtained from an aiming point lying directly beneath the observation spacecraft and having a scene brightness of 800 ft-Lamberts.

e. The ground frame size for high-resolution imagery will be 6 to 15 Km. Ground frame size for low resolution will be five times the high-resolution field of view; i.e., 30 to 75 Km.

f. Capability is desired for stereo observation of a target on a single access. The two images for the stereo pair will be acquired by the high-resolution sensor with an angular separation of 15°.

g. A target set specified for the mission provides data on geographic location, diameter and character of targets. Four levels of acquisition frequency are specified with the following numbers of target images at each level:

Monthly	220
Quarterly	300
Semi-Annually	250
Annually	6200

These numbers include provision for stereo coverage on 50% of the acquisitions and high- and low-resolution coverage on 50% of the acquisitions.

h. The observation spacecraft will have a minimum life of one year. It is desired that the life be failure limited.

4.1 IMAGERY PARAMETERS

The impact of such mission requirements as ground resolution and ground frame size can be assessed quite directly and treated in a parametric fashion to cover the region of probable emphasis. Because both affect the number of elements per image in a square-law fashion, these parameters are very significant. Relatively small changes in these quantities can move a system design from the easily obtainable to the nearly impossible.

For the purpose of this study, it is assumed that three cycles of data will be required to express the imagery data contained in one ground-resolution element (e.g., a 0.6 x 0.6 meter ground area). Number of data cycles per image is plotted in Figure 4-1, for the range of frame size and resolution specified for the strategic-surveillance mission. To minimize the load on data-storage and transmission subsystems, the frame size should be no larger than the size of the target or target complex to be imaged. Some increase in field of view beyond the target dimension must be made to assure that line-of-sight-orientation error does not result in missing part of the

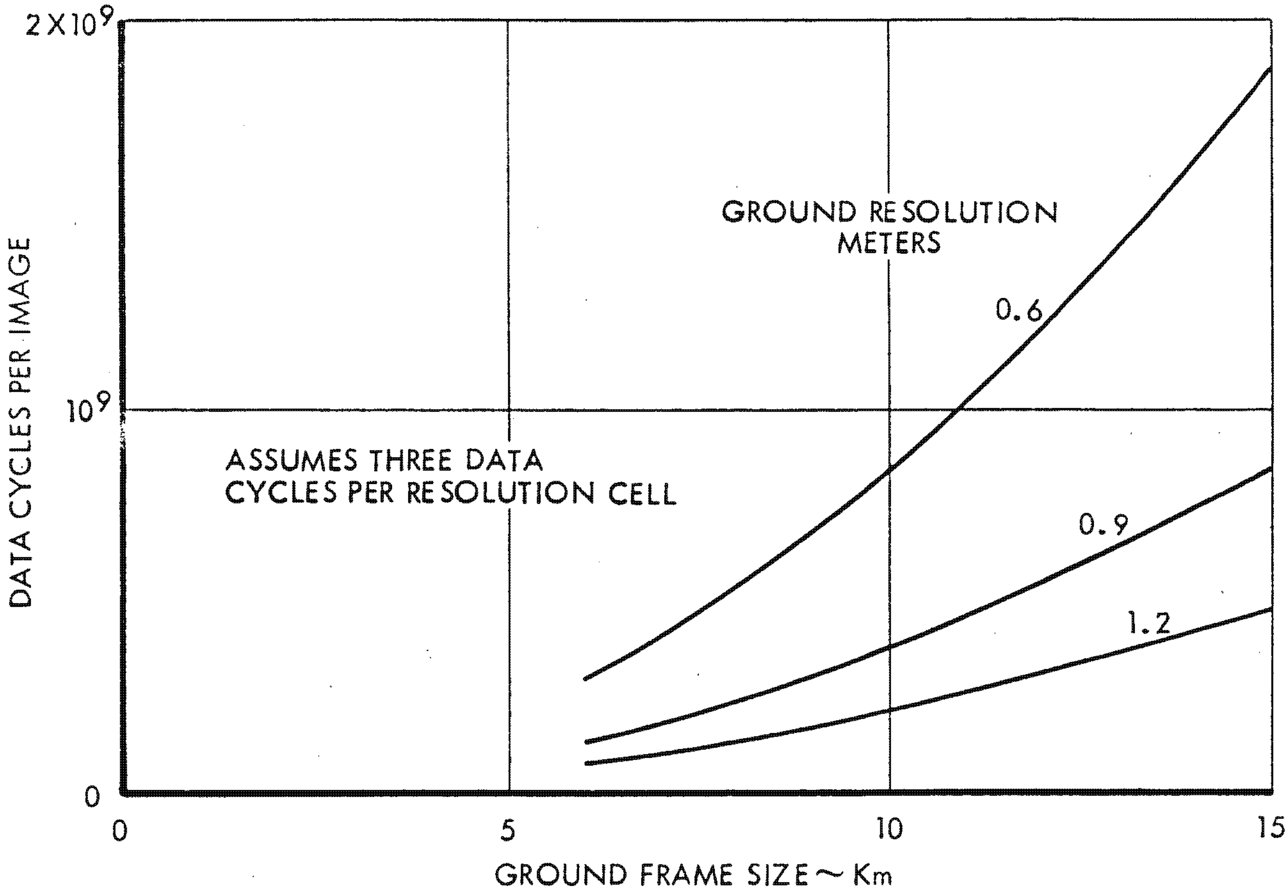

FIGURE 4-1 DATA QUANTITY PER IMAGE

target. As discussed in Volume II, Section 10, about 1 Km of the frame dimension should be allocated to pointing error tolerance. Targets in the specified model have the size distribution shown in Figure 4-2. Fortunately, the great majority of targets are small in diameter. The minimum frame size of 6 Km will accommodate targets up to 5-Km diameter with a 1-Km error tolerance. Selection of a 6-Km frame size requires multiple image acquisitions for only about 10% of the targets. Also, adequate intelligence may be obtainable from a selected 6-Km portion of a large target (e.g., hangar or revetment area of a large airfield). The frame size must be doubled (and capacity increased by a factor of 4) to accommodate 99.6% rather than 90% of the target set with a single image per aim point. The requirement for multiple images for 10% of the target set clearly involves less data than the 300% capacity increase involved in doubling the frame size. Therefore, for the purpose of this study, the ground frame size was specified to be 6 Km.

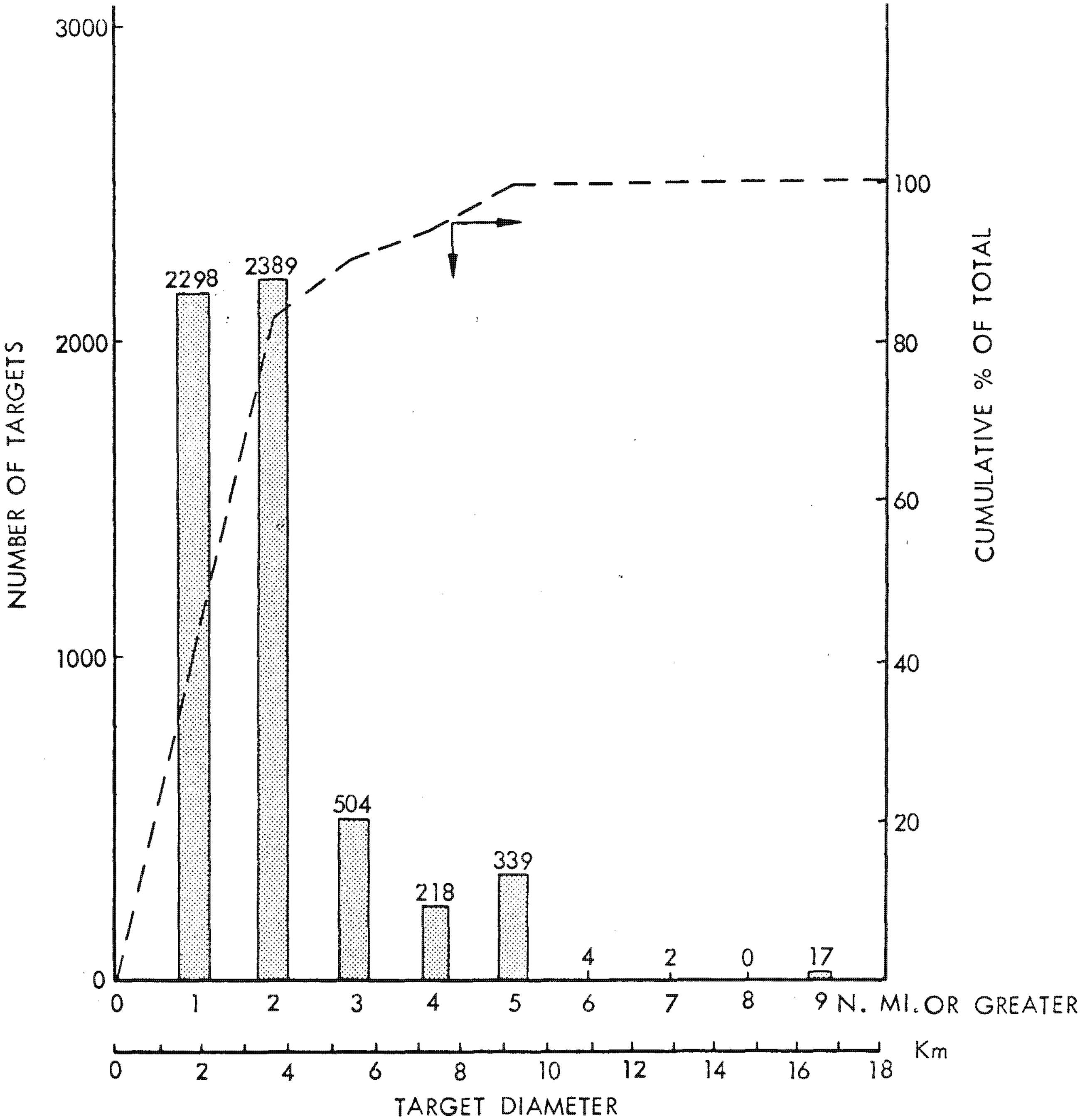

FIGURE 4-2 NUMBER OF TARGETS VS. TARGET DIAMETER

Circular, sun-synchronous orbits are the logical choice to maintain adequate illumination for long term, photographic reconnaissance satellite systems. Circular orbits are not adversely affected by orbit perturbations which make elliptical orbits unsatisfactory. Sun synchronization ensures the most favorable illumination possible throughout the world and throughout the year for imaging operations.

The orbit plane of a sun-synchronous orbit rotates about the Earth polar axis in the same sense and at the same mean rate that the Earth revolves about the sun. Consequently, the satellite overflies a given latitude at the same local sun time on each pass. For near Earth orbits the inclination required for sun synchronization is characteristically 97° so that areas between 30° and 70° latitude are overflown within one-half hour of the same local sun time. Sun-synchronous orbits thereby enable imaging to be conducted at favorable local sun time throughout Eurasia during the entire year.

Sun-synchronous orbits are possible due to an orbit perturbation (nodal regression) caused by the oblateness of the Earth.

An additional orbit perturbation due to oblateness causes elliptical orbits to be unsatisfactory for long-term surveillance systems. This perturbation, called apsidal rotation, causes elliptical orbits to rotate within their orbital plane. As a result, the orbit perigee that has been carefully oriented over a limited surveillance zone for resolution purposes slowly drifts away at about 4° per day. This rate is too fast for long-term coverage of a limited zone of interest, and it is too slow for world-wide coverage. The propulsion required to counteract this perturbation is prohibitive for long-term operations.

The only remaining major orbit characteristic to be selected for a specific system is altitude. Altitude selection is influenced by target access requirements, optical system performance, and orbit decay due to drag.

4.2 ACCESS FREQUENCY

The frequency of access to any given aiming point is related to the size of the ground area that the observation satellite can access at any given instant. The access area is determined by the maximum permissible angle between local vertical and the optical line of sight and by the altitude of the satellite. The relationship of access frequency to altitude for given view-angle limits is shown in Figure 4-3. The derivation of this relation-

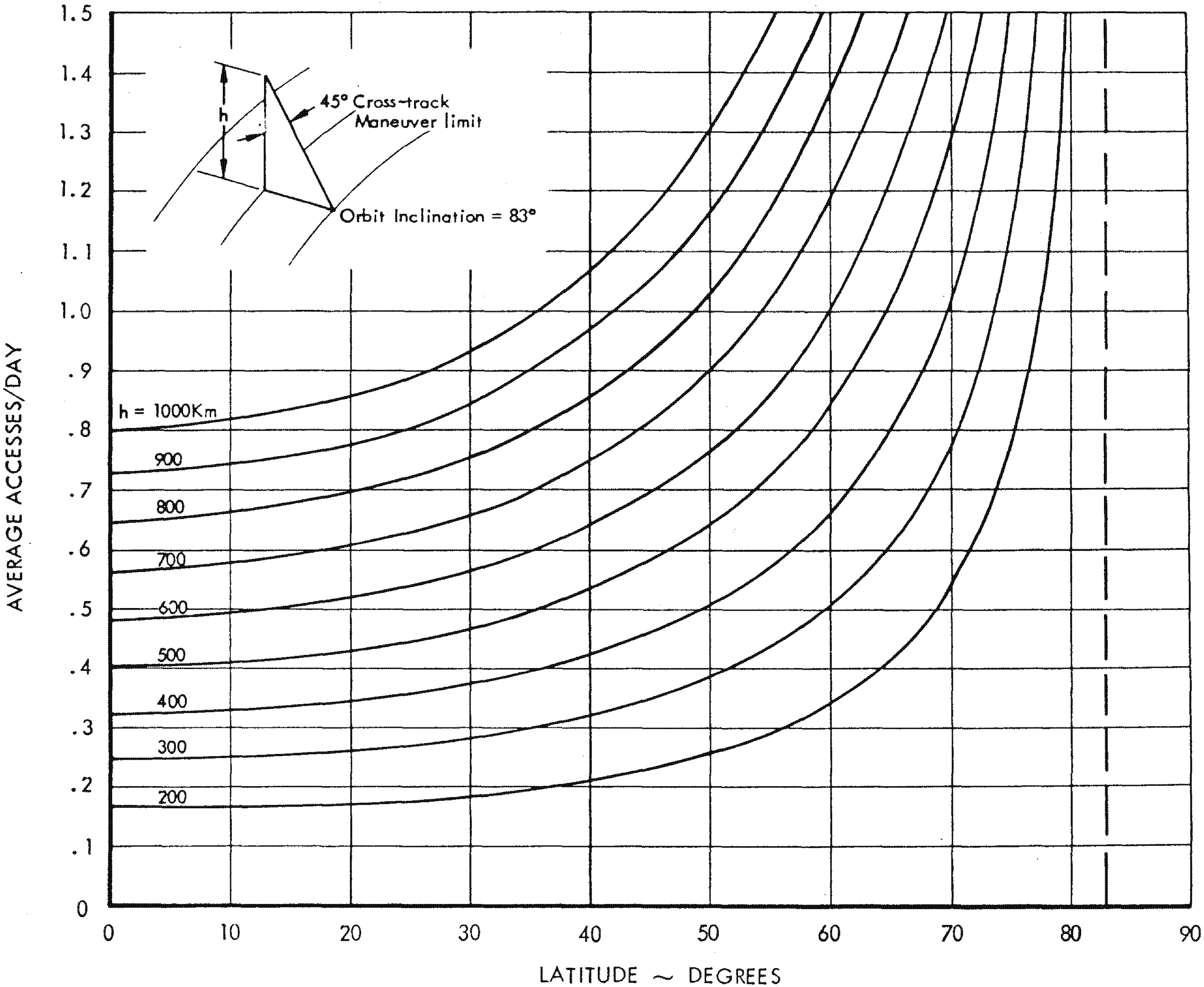

FIGURE 4-3 ACCESS FREQUENCY VS. ALTITUDE & LATITUDE [POINT TARGET]

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ship can be found in Volume II, Section 4. As shown in Figure 4-3, access frequency increases linearly with altitude. This argues for high-altitude operation. Conversely, the weight penalty and technical problems of large optics make it desirable to operate at as low an altitude as the mission requirements permit.

4.2.1 Coverage Constraint on Altitude

Some altitudes result in incomplete coverage as shown in Figure 4-4. The peaks shown result from orbital periods that result in periodic repetition of the same ground path. A daily repeat occurs at 277- and 568-Km altitudes. A repetition of the ground path occurs every two days at a 400-Km altitude. At these altitudes some regions between ground paths are never accessible. Thus, to assure coverage completeness, the candidate altitudes are limited to the nominal ranges of 215 to 250, 305 to 410, and 430 to 540 Km. As shown in Figure 4-4, these ranges assure complete coverage in three to seven days for the latitudes indicated and a maximum permissible viewing angle of 45 degrees. It should be noted the coverage period in Figure 4-4 is the time required for complete coverage. During this period, much of the area will be covered more than once. Figure 4-3 represents a smoothed average access rate, and should be used to determine expected accesses provided the required frequency of image acquisitions of a given target is low enough (a few per year).

4.2.2 Drag and Orbitkeeping Considerations

Orbits in the altitude range of 200 to 600 Km decay due to atmospheric drag. A circular orbit remains essentially circular but its altitude decreases at an ever-increasing rate as illustrated in Figure 4-5. If a system is to achieve complete coverage for the duration of the mission, the orbit altitude must not decay through altitudes at which incomplete coverage occurs. For the example indicated in Figure 4-4, the orbit must begin between A and B and must not decay below B in the time interval of interest.

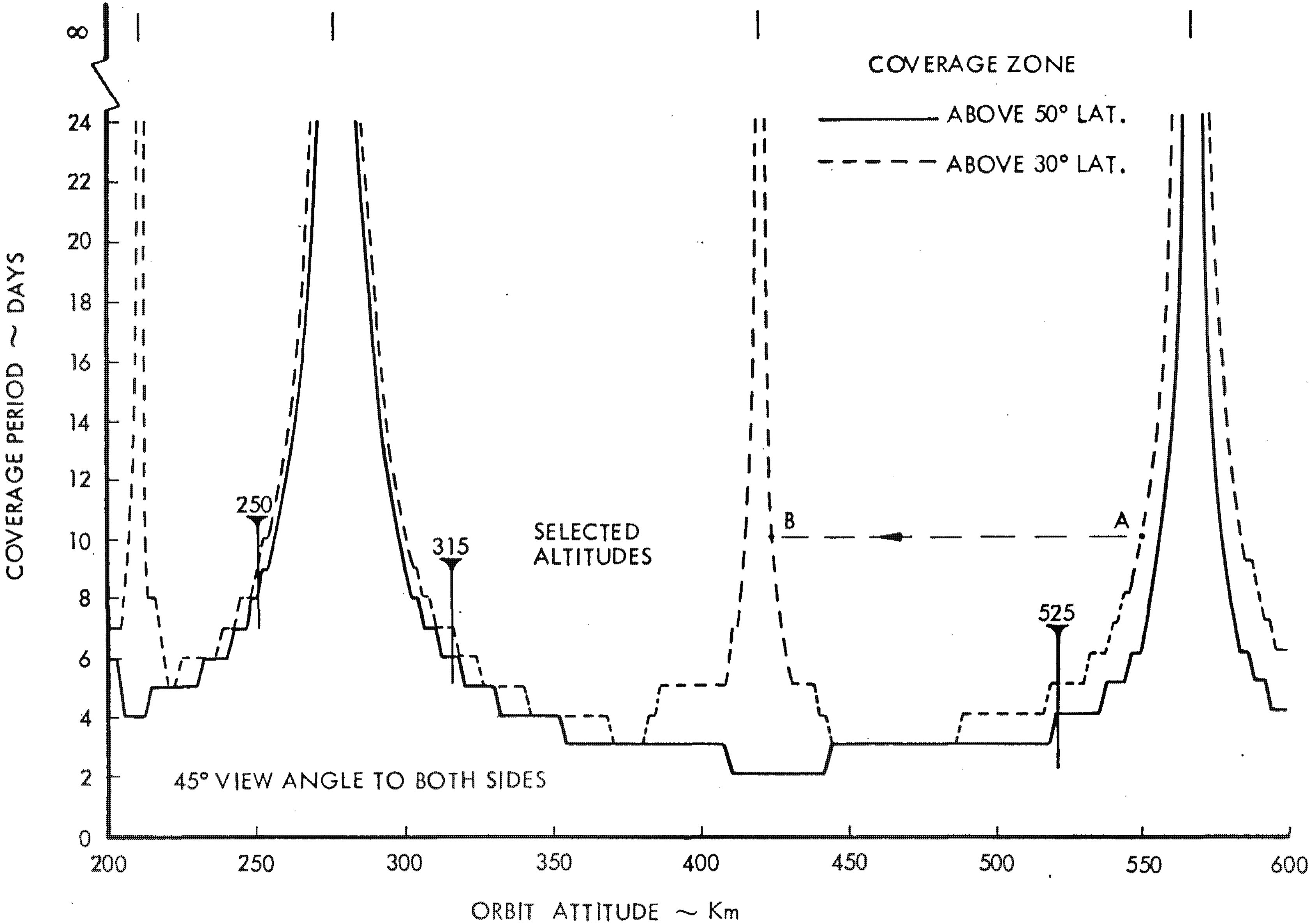

FIGURE 4-4 COVERAGE VERSUS ALTITUDE

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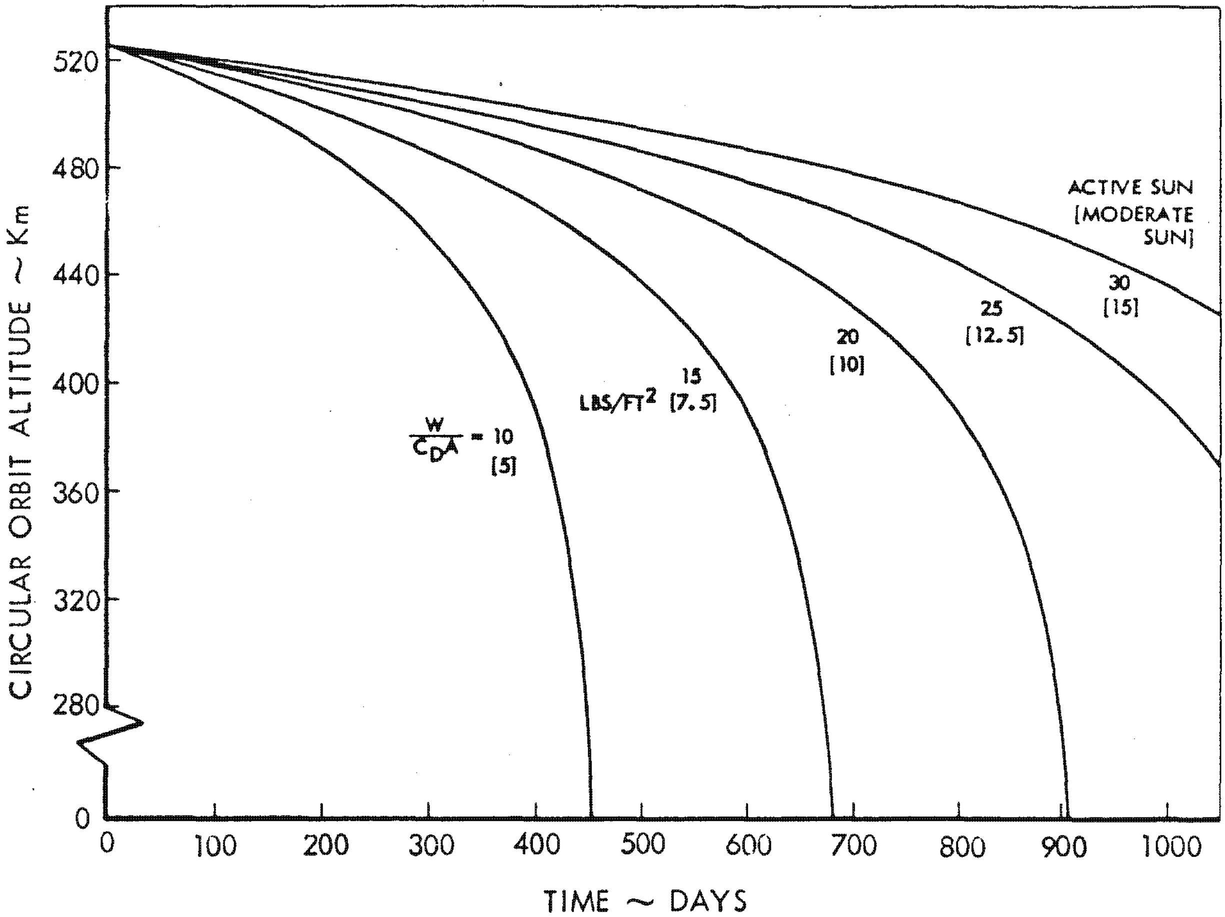

FIGURE 4-5 CIRCULAR ORBIT DECAY PROFILES

At the lower altitudes it is necessary to employ orbitkeeping (restoration, by means of propulsion, of altitude losses due to drag) to avoid incomplete coverage. Figure 4-6 shows the amount of orbitkeeping propellant required per year per 100 ft^2 of drag area. Drag areas of 100 to 200 ft^2 are typical for the spacecraft of interest. The data of both Figures 4-5 and 4-6 represent the severest conditions (greatest atmosphere density) anticipated for the time period of 1970 through 1978. Upper atmosphere density varies with solar activity which follows an eleven-year cycle that is minimum in 1975-1976 and maximum in 1968-1969 and 1979-1980. Orbitkeeping does not preclude low-altitude orbits but it does impose a complexity and weight penalty. Orbitkeeping is feasible with storable propellants at altitudes as low as 215 to 250 Km. In this altitude range, orbitkeeping maneuvers must be performed about once a week.

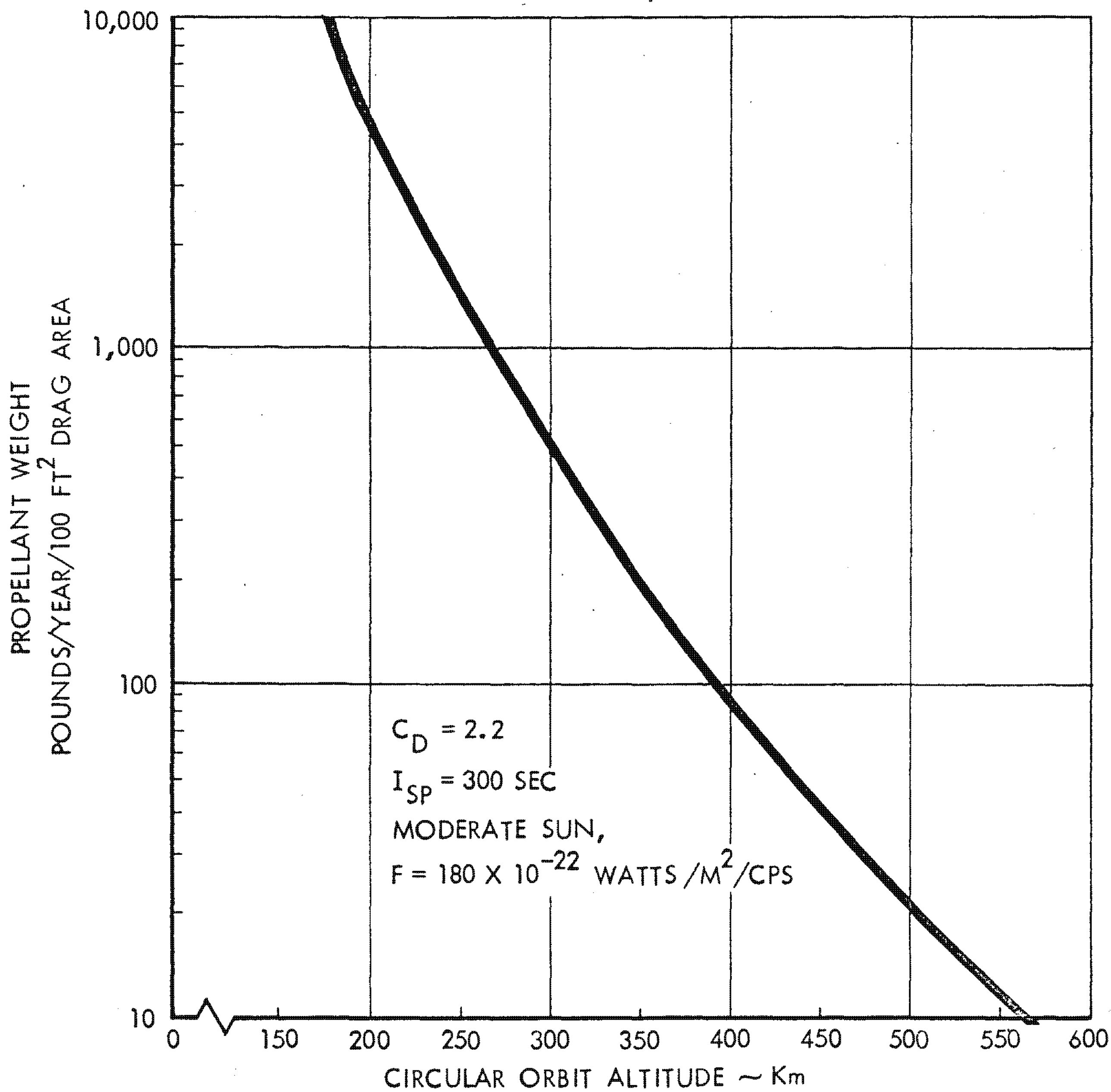

FIGURE 4-6 ORBIT KEEPING PROPELLANT REQUIREMENT [ANNUAL PER 100 SQ. FT. DRAG AREA]

It is possible to achieve a one-year mission without orbitkeeping from initial altitudes of about 500 Km under moderate and high solar activity conditions (1969-1972, 1978-1983). Under quiet sun conditions (1973-1977), it is possible to get a one-year operation free of orbitkeeping from 410 Km. The orbit will decay to 310 Km after one year.

Satellite position prediction, essential for system operation, is influenced by atmospheric drag. Atmosphere density uncertainty contributes directly to in-plane position error. This component of error grows as the square of the prediction interval. Since the magnitude of the density uncertainty increases drastically at the lower altitudes, tracking and navigation operations must be more current to support lower-altitude surveillance satellites. Although these tracking and navigation requirements do not preclude lower-altitude surveillance orbits, less stringent tracking and navigation requirements are an advantage of higher-altitude orbits.

4.2.3 Orbit Management

Orbit management pertains to changing (and/or maintaining) the orbit of a satellite to achieve a desired effect. In this application, the desired effect is coverage flexibility (increased observation opportunities when needed). Such orbit management must be achieved by the use of propulsion. The coverage flexibility afforded by orbit management and the propulsion ΔV required to implement it are presented in the following paragraphs.

There may be specific sites that, on occasion, require daily coverage for a short period of time. Such coverage is possible if the propulsion is available to transfer back and forth between a complete coverage altitude and the nearest daily-coverage altitude. It may also be possible to employ daily coverage to obtain sufficient annual coverage of a high-density area at the expense of less critical areas.

Circular orbits at 277-Km and 568-Km altitudes result in daily (but incomplete) coverage. The ΔV required to transfer to these daily-coverage orbits from initial orbit altitudes of 200 to 600 Km is shown in Figure 4-7. Since the daily coverage strips alternate longitudinally with strips of no coverage, phasing of the transfer is required to ensure the desired longitudinal positioning of the daily coverage strips. During any one coverage period of the initial orbit the required phasing should be closely approached and the transfer can be made at ΔV's approaching the minimum ΔV's shown. The ΔV required for transfer within two days under worst phasing condition is shown in the upper left-hand corner of Figure 4-7.

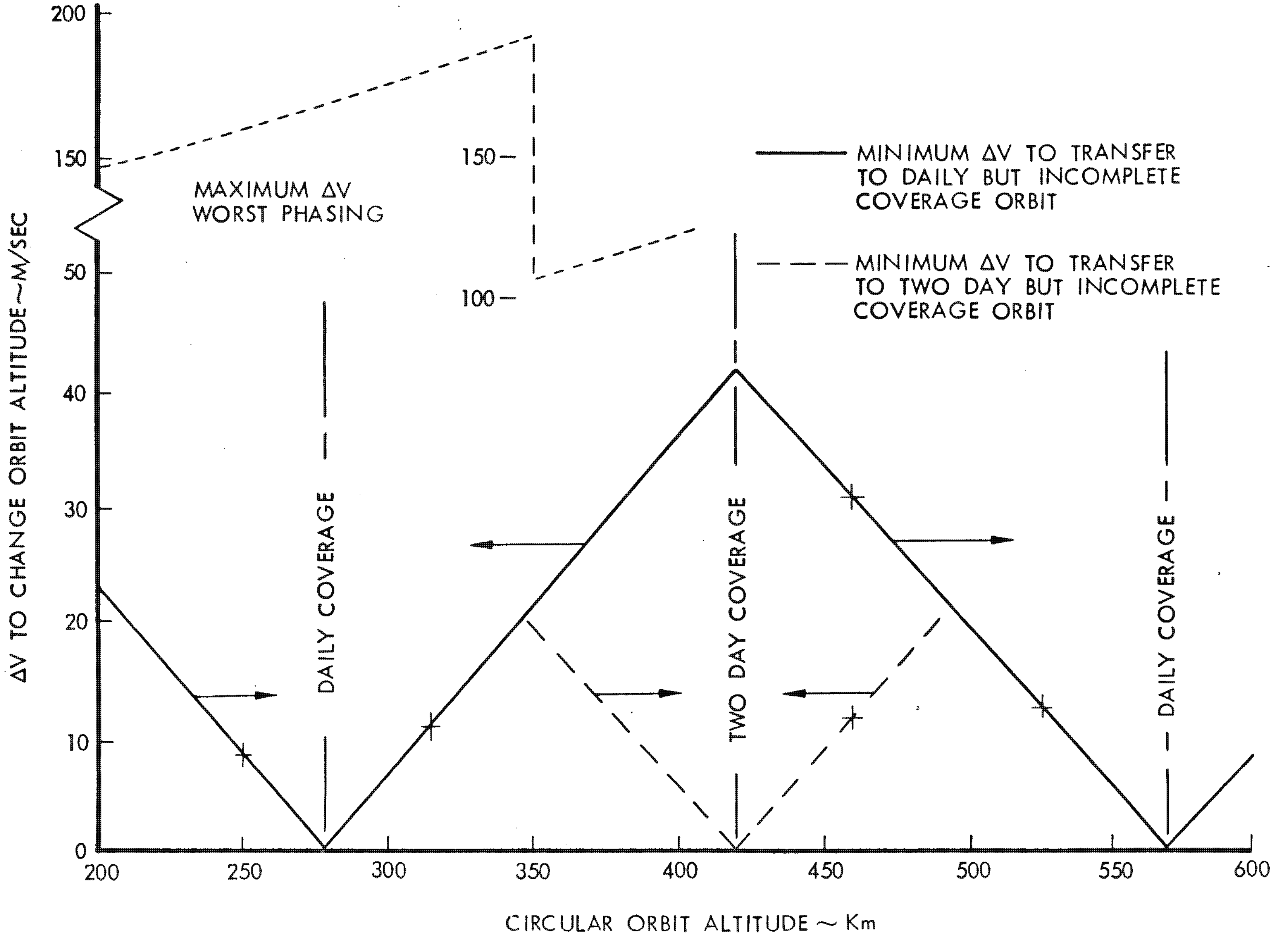

FIGURE 4-7 COVERAGE FLEXIBILITY ΔV

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When the nominal operational altitude is 200 to 250 Km it is feasible to use drag to return from daily coverage operations at 277 Km. See Figure 4-8. By stabilizing the satellite in its high-drag attitude ($W/C_DA = 50$), the time required to return to normal operations at 250 Km is about four days. It requires ten days if the normal drag attitudes ($W/C_DA = 25$) are maintained. Upon reaching 250 Km, the normal attitudes and, thereby, the normal 7- to 8-day orbitkeeping cycle are resumed. In this case propulsion is required only to transfer one way (up to the daily-coverage altitude, 277 Km).

It is not feasible to use orbit decay to transfer to the daily-coverage altitude from altitudes of 300 to 400 Km, because phasing is not necessarily achieved. In this case, propulsion is required to transfer both to and from the daily coverage operation. It is not feasible to return to normal operation at 400 to 500 Km from daily operation at 568 Km by means of orbit decay because of the extremely low decay rates at these altitudes.

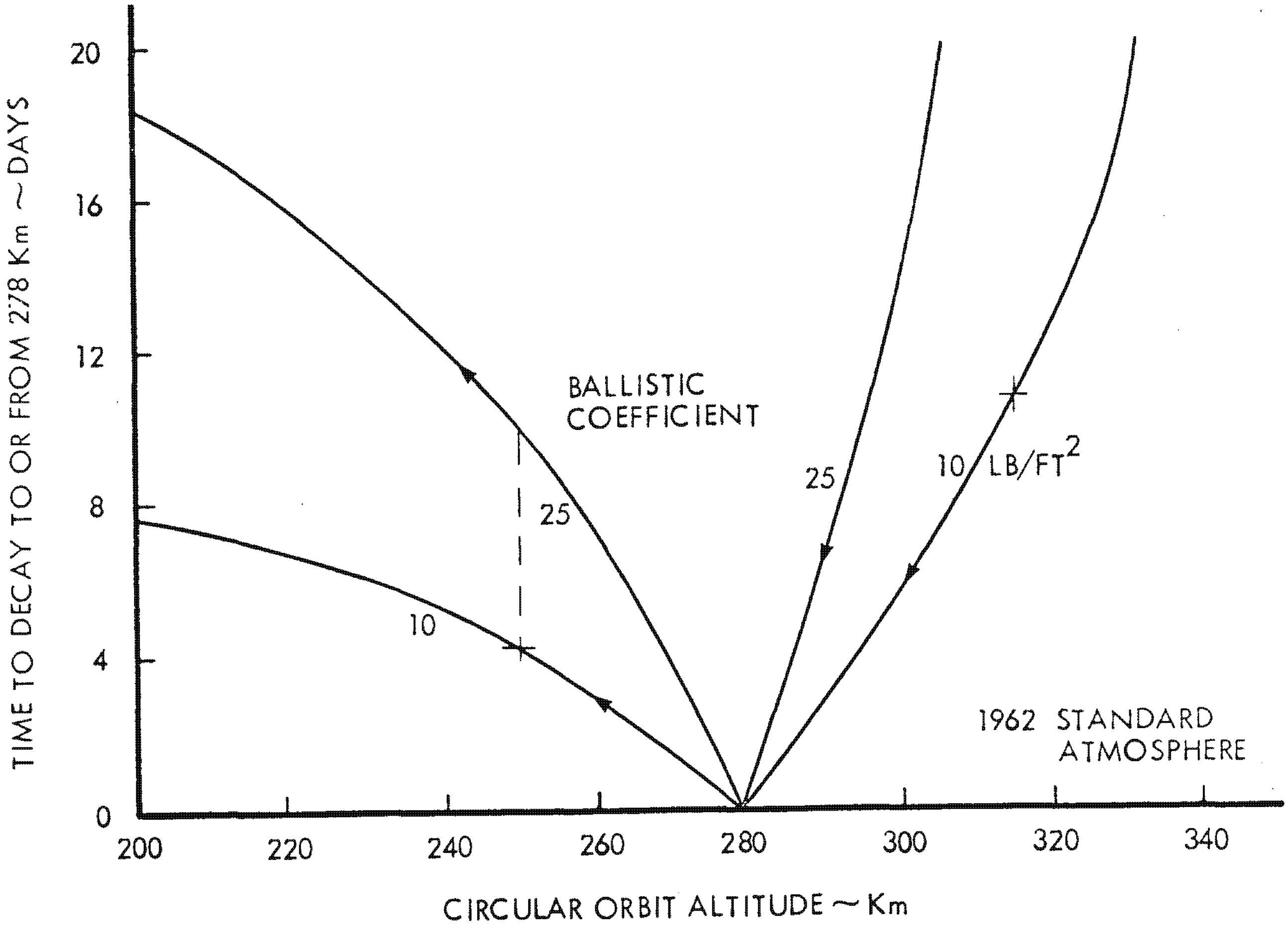

FIGURE 4-8 ORBIT DECAY TIMES

From operational altitudes of 200 to 250 Km, periods of daily coverage can be achieved wholly within the system's normal orbitkeeping budget. From altitudes of 300 to 400 Km and 400 to 550 Km, periods of daily coverage can be achieved at nominal increases in stationkeeping propellant. Nominal circular orbit altitudes of 250, 315, and 525 Km are favorable from the standpoint of complete coverage with optional daily coverage.

4.3 DATA TIMELINESS

For the strategic surveillance mission the allowable data delivery delay, excluding ground functions, is specified as 12 hours. Response time is not a critical parameter for this mission. Should a mission require fast response and/or delivery, it would impose a severe constraint on concept selection.

4.4 IMAGE CAPACITY

A simple count of aiming points, weighted by frequency of observation, defines the basic image capacity requirement for the system. This work load naturally breaks down into a daily increment and for each day to the number of images required for each orbit revolution. Because the aiming points tend to be concentrated in certain geographic areas, some allowance for bunching must be made in determining reasonable "per-day" and "per-orbit" capacities from an annual image capacity. Also, allowance must be made for loss of system operating time due to targets being obscured by cloud cover. Statistically, the average target will be free of weather interference on about one day out of three.

Considering these factors plus the requirement for stereo pairs on half of the acquisitions and an overlaid low resolution image on half of the acquisitions, a daily capacity of 75 images appears appropriate for the strategic surveillance mission.

4.5 TARGET CONCENTRATION

One of the most significant characteristics of an orbital reconnaissance system is its low duty factor. The observation satellite will be over areas of interest only about 5% of the time. In addition, targets are not uniformly distributed but tend to be grouped. Figure 4-9 presents a map showing the strategic surveillance mission target distribution in 10 degree latitude by 10 degree longitude blocks. Finer-scale grouping of targets is also found within the blocks. Therefore the system cannot be designed on an average-work-load basis, but must be able to handle peak rates.

Analysis of the strategic surveillance target data resulted in selection of a limiting "worst-case" area to use in establishing peak rate requirements. This analysis is presented in Volume II, Section 4. As can be expected, peaks in the frequency with which targets are encountered result from clustering of targets (density) rather than number of targets alone. Density is effectively increased when the target cluster contains many targets requiring several observations per year, and the demand on the system is further increased when a given high density extends over a relatively large region. Thus, the 10° latitude x 10° longitude block around Moscow imposes a greater peak rate requirement than does an adjacent block which contains a greater number of targets.

Essentially all of the 816 target images in the selected "worst-case" area can lie within the access swath of an observation satellite operating at 525-Km altitude with a crosstrack viewing angle limit of 45 degrees from vertical. At this altitude the observation satellite can be expected to have 94 clear accesses per year to this "worst-case" area. For reasons reported in detail in Volume II, Section 4, the access swath of an observation satellite operating at one of the lower candidate altitudes, e.g., 250 Km, will contain most of the "worst-case" area targets, but the number of clear accesses is reduced to about 50 per year. Thus, operating at a lower altitude appreciably affects the peak acquisition rate.

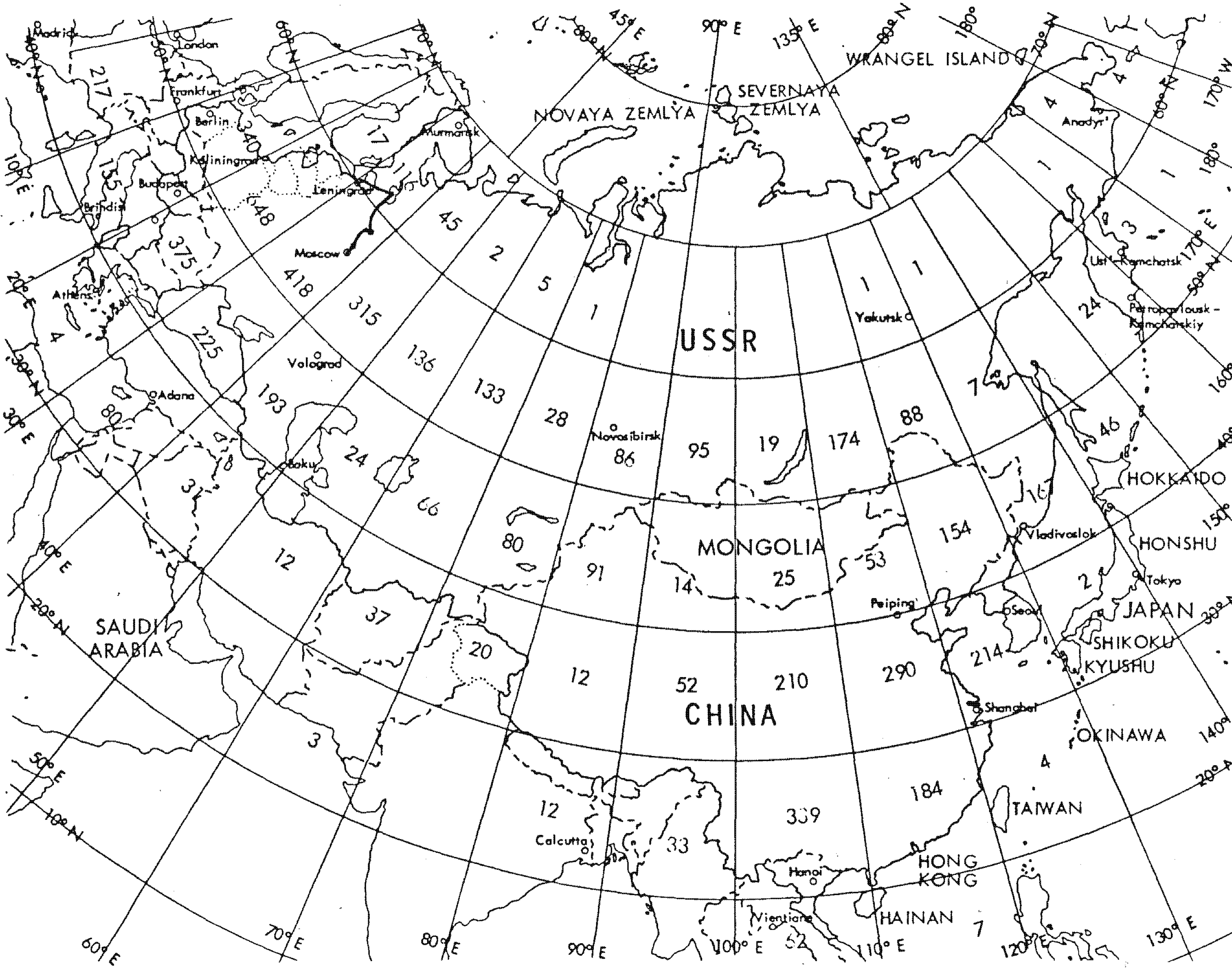

FIGURE 4-9 GEOGRAPHIC DISTRIBUTION OF TARGETS

ZA / TOP SECRET

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4.6 POINTING FREEDOM AND AGILITY

In addition to timely access and observation from desired viewing angles, the ability to handle concentrations of aiming points in relatively small geographic areas is dependent upon the freedom given to the line-of-sight orientation and the time required to execute repositioning maneuvers within this angular freedom.

By treating crosstrack and fore-aft pointing independently, the next two sections provide quantitative insight into their dependency on target characteristics and mission requirements. Section 4.6.3 discusses the interplay between the two axes of pointing control in handling a realistic two-dimensional distribution of aiming points.

4.6.1 Crosstrack Repositioning

Within the crosstrack pointing freedom required to access aiming points, a repositioning maneuver is required for each aiming point. The magnitude of these maneuvers depends upon the size of the region in which targets must be acquired on a single pass (orbit revolution).

For the altitude range of interest the number of orbit revolutions per day lies between 15 and 16, resulting in a spacing of 22-1/2 to 24 degrees longitude between ground traces. The average spacing for clear passes during a year (assuming one day in three is clear) is therefore less than .2 degree or a distance of about 12 kilometers at the upper middle latitudes where target concentrations are most likely to occur. If aiming points were uniformly distributed this interest area could be divided into subswaths of this width, each of which would be covered on a separate orbit revolution. In such a situation a conservative estimate of nominal magnitude of crosstrack repositioning maneuvers is

$$\theta_C = \frac{\text{Subswath Width}}{\text{Altitude}} \times 57.3 \text{ degrees.}$$

If the interest area must be covered in a time period shorter than a year, the number of clear passes available is reduced and the subswath width increased proportionately. Figure 4-10 shows these results for the three selected altitudes.

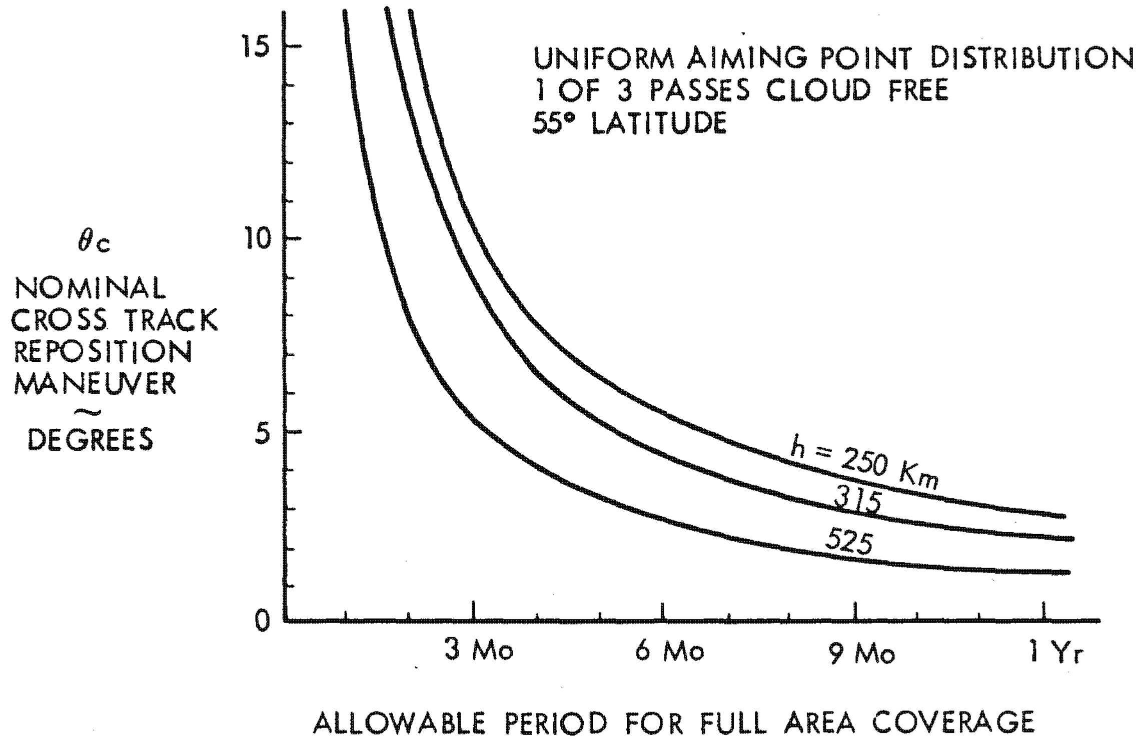

FIGURE 4-10 CROSSTRACK MANEUVER MAGNITUDE

Concentration of aiming points as depicted in Figure 4-11 can be handled by using narrow subswath widths in the dense region and wider subswaths in the sparse regions so that the number of acquisitions per clear pass remains essentially constant. This leads to larger crosstrack repositioning maneuvers in the sparse regions. As an extreme example, if 80% of the aiming points were concentrated along a line, 20% of the clear passes would have to cover the entire region of sparse targets. This would lead to crosstrack repositioning maneuvers five times as great as the values given in Figure 4-10. However, 80% of the acquisitions would require no crosstrack repositioning, resulting in the same <u>average</u> repositioning maneuver.

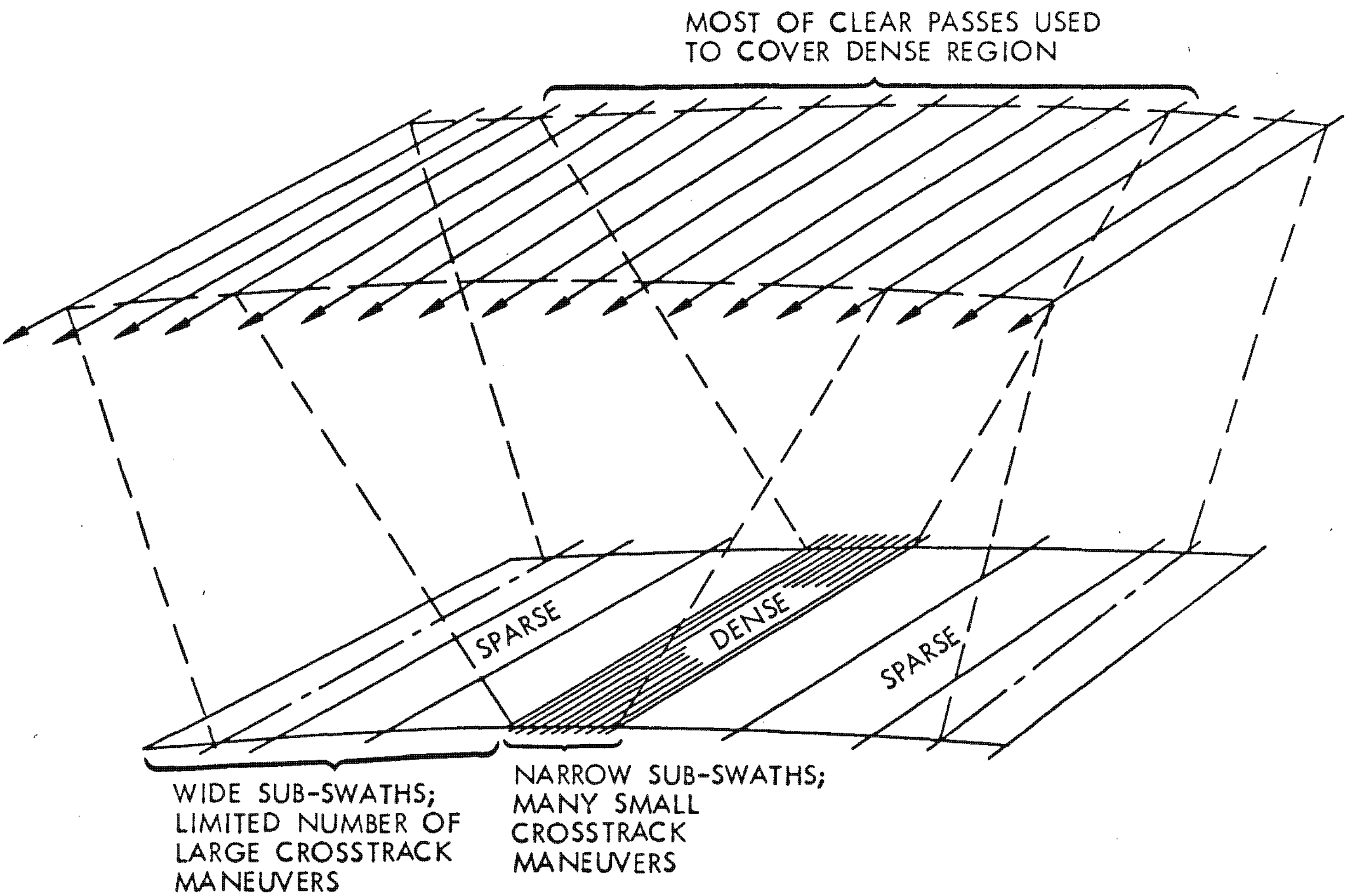

FIGURE 4-11 CROSSTRACK AIMING POINT CONCENTRATION

This one-dimensional treatment of aiming-point distribution assumes the crosstrack line-of-sight orientation for a given orbit revolution can be established prior to entering the interest area and that fore-aft pointing freedom is available to accommodate any concentration of targets in the downrange direction. Section 4.6.3 contains a qualitative discussion of the pointing control trades associated with a realistic two-dimensional aiming point distribution.

4.6.2 Fore-Aft Repositioning

The basic constraint on peak image acquisition rate is the picture interval,

$$t_p = t_I + t_R ,$$

where t_I is the imaging time required for all functions associated with a single image acquisition and t_R is the repositioning time required to reorient the line of sight from one aiming point to the next. Each orbit pass can be divided into segments of length equal to the satellite motion during one picture interval. These segments are directly analogous to the subswaths in the crosstrack dimension. Again the aiming-point concentration can be handled by providing pointing freedom to allow access to the dense region from outlying segments.

If the number of clear passes available is at least as great as the number of aiming points in a concentration, a single acquisition can be made on each pass and no fore-aft pointing is required. Fore-aft pointing may also be avoided if the picture interval is very short, allowing multiple acquisitions within the concentration. When fore-aft pointing is required, the needed pitch freedom is proportional to the picture interval as suggested by Figure 4-12.

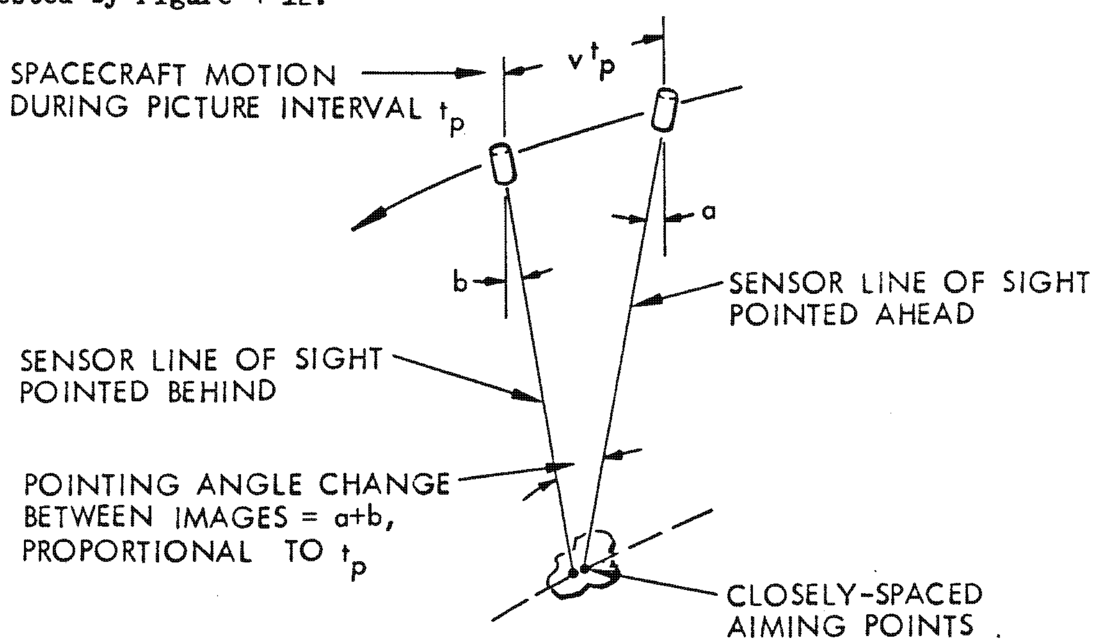

FIGURE 4-12 FORE-AFT SENSOR POINTING

Figure 4-13 gives quantitative results for a "worst case" target concentration drawn from the strategic surveillance target model. The increased pitch freedom required at low altitude results from the smaller number of accesses occurring during a year.

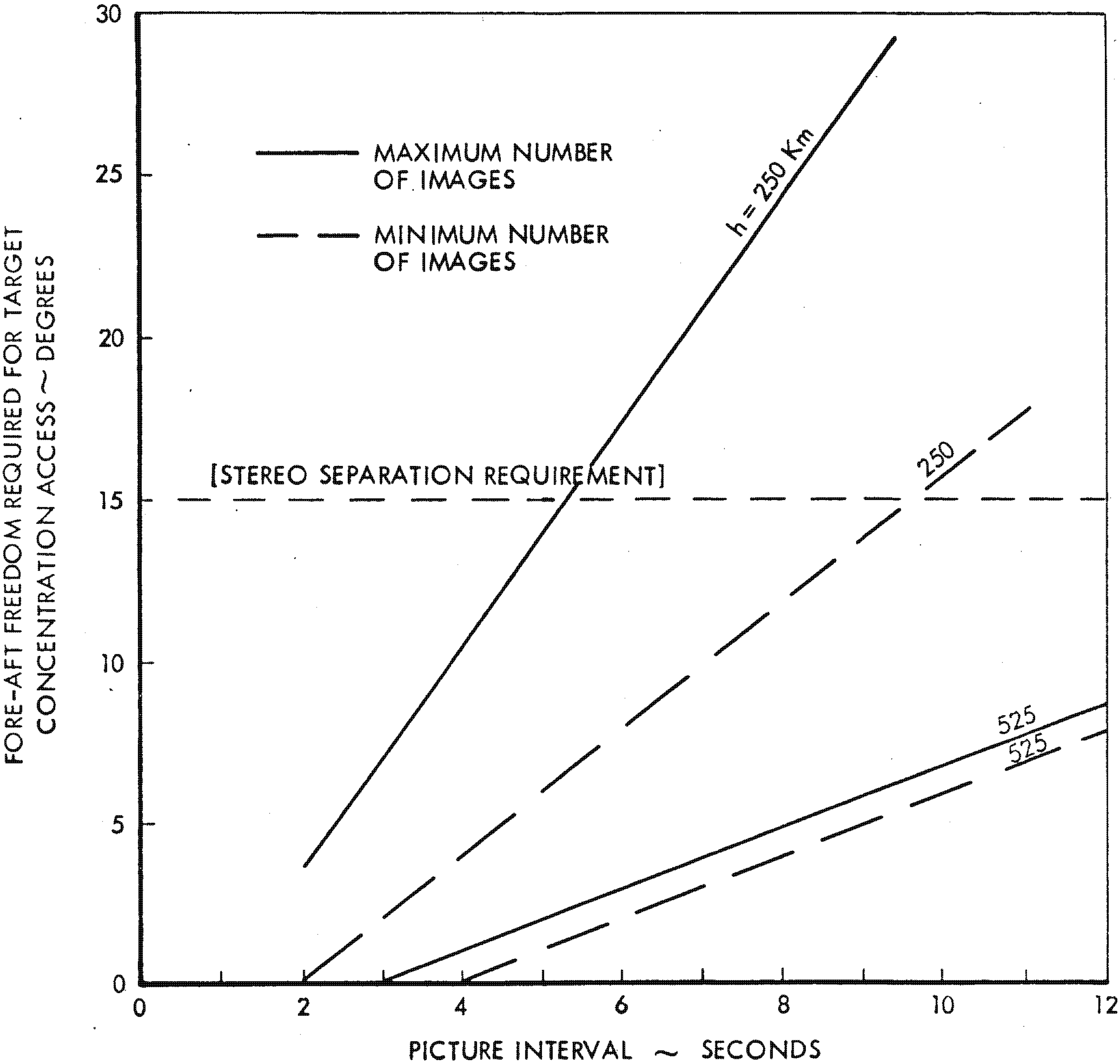

FIGURE 4-13 FORE-AFT POINTING FREEDOM REQUIREMENTS

4.6.3 Trades Between Fore-Aft and Crosstrack Pointing

As depicted in Figure 4-14, the number of imaging opportunities for an area of aiming-point concentration is proportional to the product of crosstrack and fore-aft pointing freedom. This suggests the existence of a trade relationship in which a reduced requirement for pointing freedom about one axis can be achieved by increasing pointing freedom about the other axis. This trade opportunity can be particularly significant in a situation where there is interference between several aiming point concentrations.

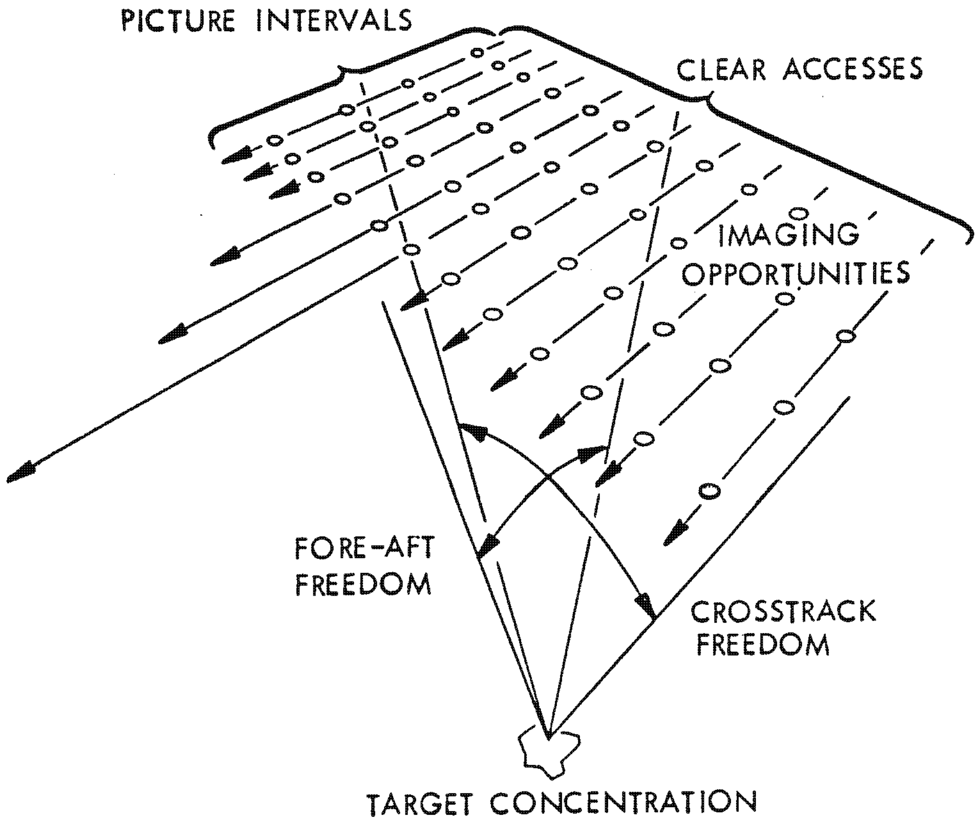

FIGURE 4-14 IMPACT OF POINTING FREEDOM ON IMAGING OPPORTUNITIES

A simplified example of such interference is shown in Figure 4-15. For case a no fore-aft pointing is used, but large crosstrack pointing maneuvers are required on many passes. By using fore-aft pointing to achieve multiple acquisitions within a single concentration, case b eliminates the large

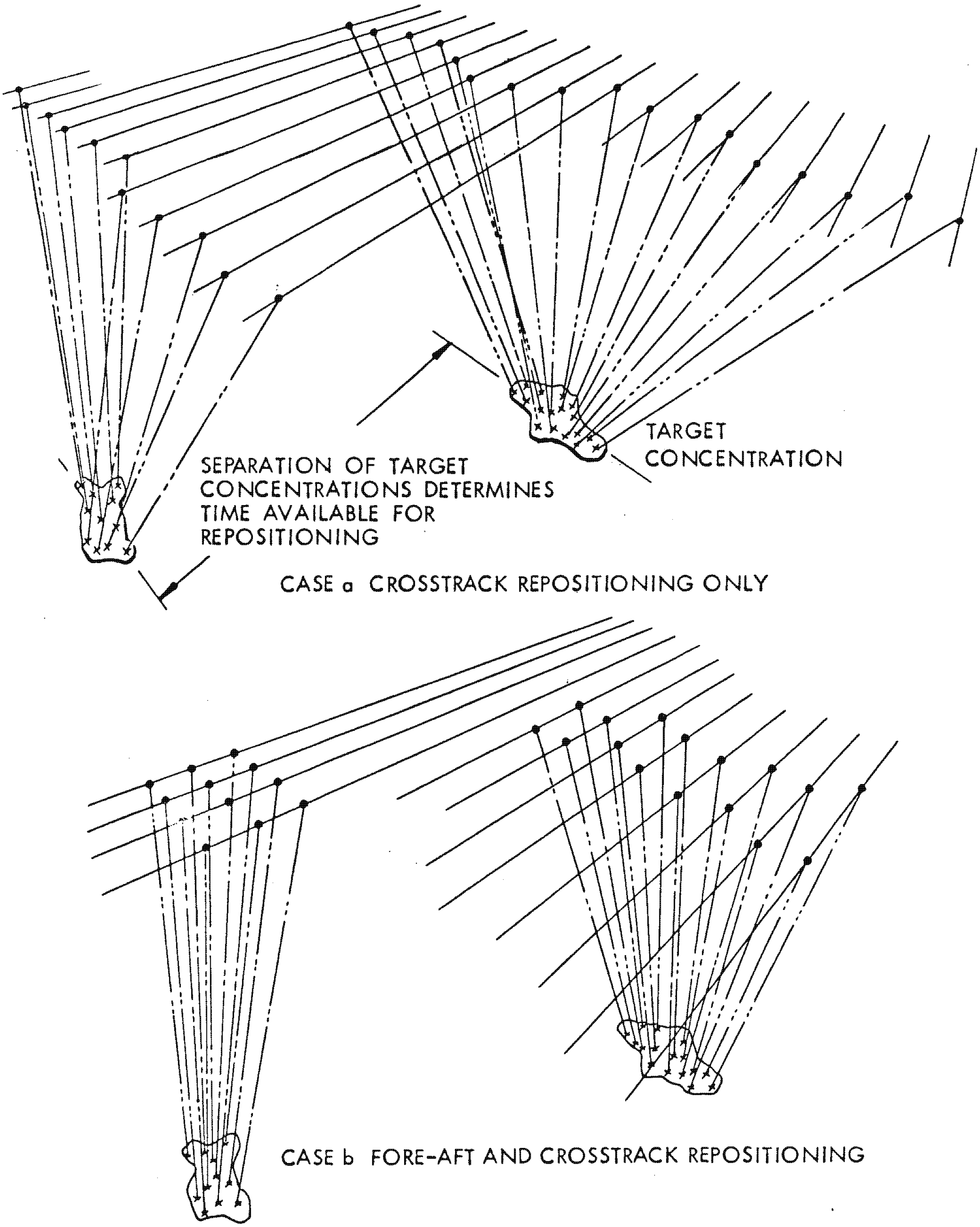

FIGURE 4-15 POINTING FREEDOM TRADE

crosstrack maneuvers. However case b requires many small pointing maneuvers within each concentration. A variety of solutions exist between these two extremes. The most efficient tactic is not easily determined and will depend upon the detailed nature of the aiming-point distribution, maneuver-time-versus-repositioning-angle performance of the system, and any additional constraints on pointing-angle freedom.

A trade also exists between picture interval and pointing control. For example, if fore-aft freedom is fixed, a shortened picture interval permits acquisition of more images on each pass through a target concentration and reduces the need for large maneuvers between concentrations. The same effect was achieved in case b, above, by using fore-aft pointing. Crosstrack freedom may be limited if a short picture interval is provided so that relatively few passes are required to acquire the targets in the concentration. These passes may be assigned to revolutions which pass near the concentration.

4.6.4 Maximum Allowable Picture Interval

The trade between downtrack maneuver freedom and picture interval presented in Figure 4-13 cannot be extended indefinitely. Constraints imposed by viewing angle limits, regional coverage requirements and interference between target concentrations limit the allowable picture interval. Since they are basically independent phenomena, they must be examined individually to determine which represents the most severe restriction. For the target model used, stereo-pair acquisition with 15° separation presents the limiting constraint.

The acquisition of stereo pairs on one orbit revolution with a single sensor requires that the line of sight be reoriented and a second image acquired by the time the observation satellite moves the distance required for the stereo base. Figure 4-16 presents the picture interval (repositioning time plus time on target required for image acquisition) as a function of altitude and stereo base.

For coverage of target concentrations, Figure 4-13 shows an increasing requirement for fore-aft pointing freedom as picture interval is increased.

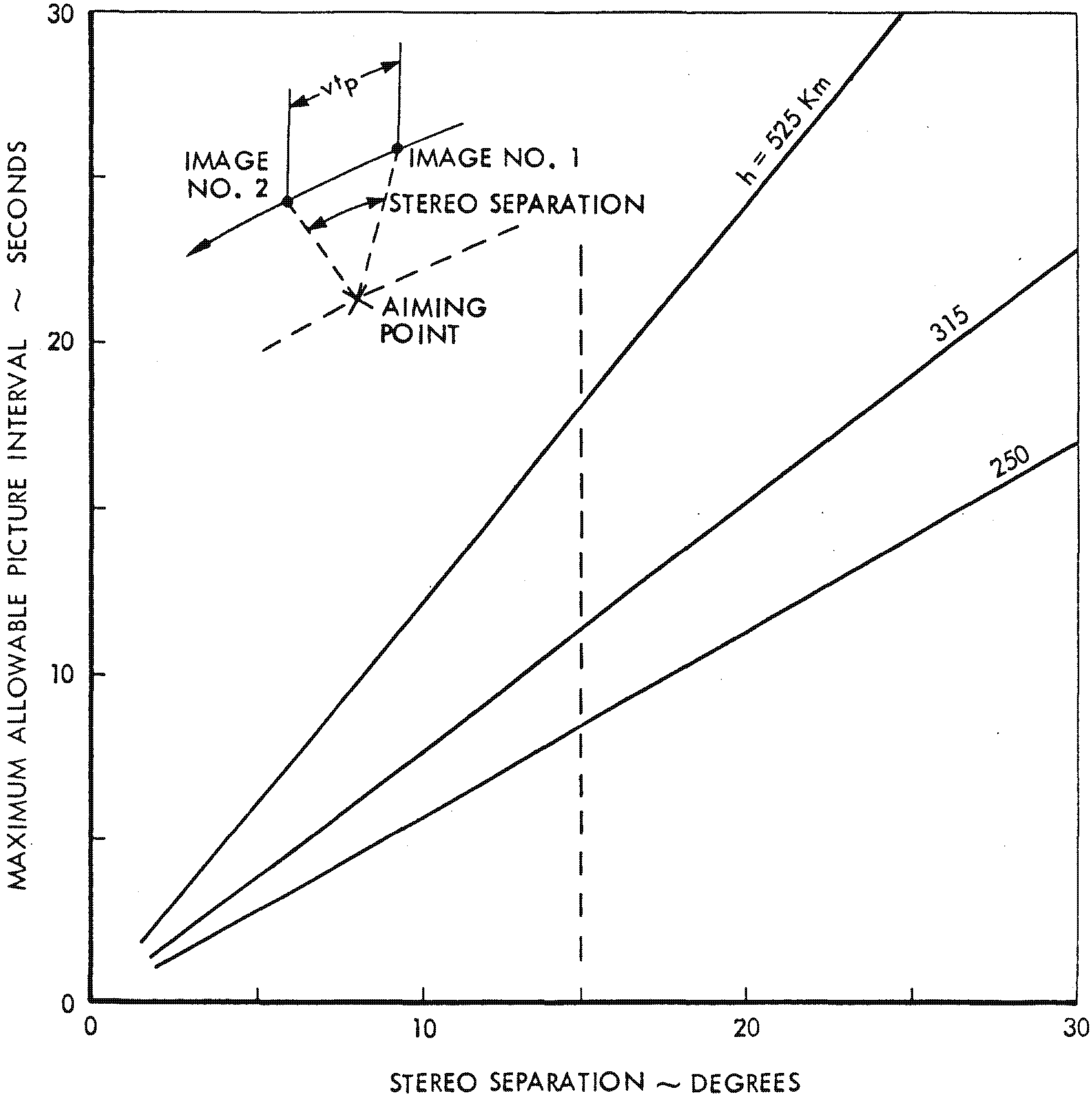

FIGURE 4-16 STEREO CONSTRAINT ON PICTURE INTERVAL

Depending upon altitude, this fore-aft pointing freedom can be expressed in terms of distance, or time, along the orbit path. When the interval devoted to a target concentration approaches the interval between adjacent concentrations, the available time on any orbit revolution is completely absorbed in target acquisitions and crosstrack repositioning between target concentrations. Further increase in picture interval leads to an inability to meet mission requirements. Accurate assessment of this limitation on picture interval involves detailed determination of aiming-point acquisition sequences and assumption of pointing-control performance capability. Although such analysis was not carried out, approximate results indicate this constraint lies beyond the 15° stereo coverage constraint on picture interval.

When targets are randomly distributed over a broad region rather than grouped in dense concentrations, the average picture interval is simply the total time available for image acquisition divided by the total number of images. If a small amount of maneuver freedom is provided to accommodate statistical fluctuation in distribution, this average picture interval is a reasonable approximation of the maximum allowable picture interval. The section 20° to 30° East longitude, 50° to 60° North latitude consists mainly of randomly distributed targets and requires 754 target images per year. With a sun-synchronous orbit having 15 revolutions per day and a 1/3 weather factor, the total time for image acquisition is 630 minutes per year. The corresponding 50-second picture interval is clearly less demanding than the stereo limit.

4.7 IMAGERY DATA RATE

With the imposition of time constraints on image acquisition and data transmission, system capacity appears in the form of data rate. Where data must be handled in electronic form, the information rate (bandwidth) is a basic measure of technical difficulty.

4.7.1 Acquisition Rate

Imaging time is dependent upon the transducer used. For the frame-imaging alternatives the information is stored directly as an image and does not appear immediately as serial information in an electrical signal. Thus image acquisition time is limited only by shutter operation time, and a very high effective data rate is possible.

The broom-scan concept requires conversion of the image information into electrical waveforms in a real-time manner. By use of slowed scan, the imaging time can be varied to accommodate bandwidth limitations of hardware. Figure 4-17 shows the trade between data rate and imaging time for the specified range of frame size and ground resolution.

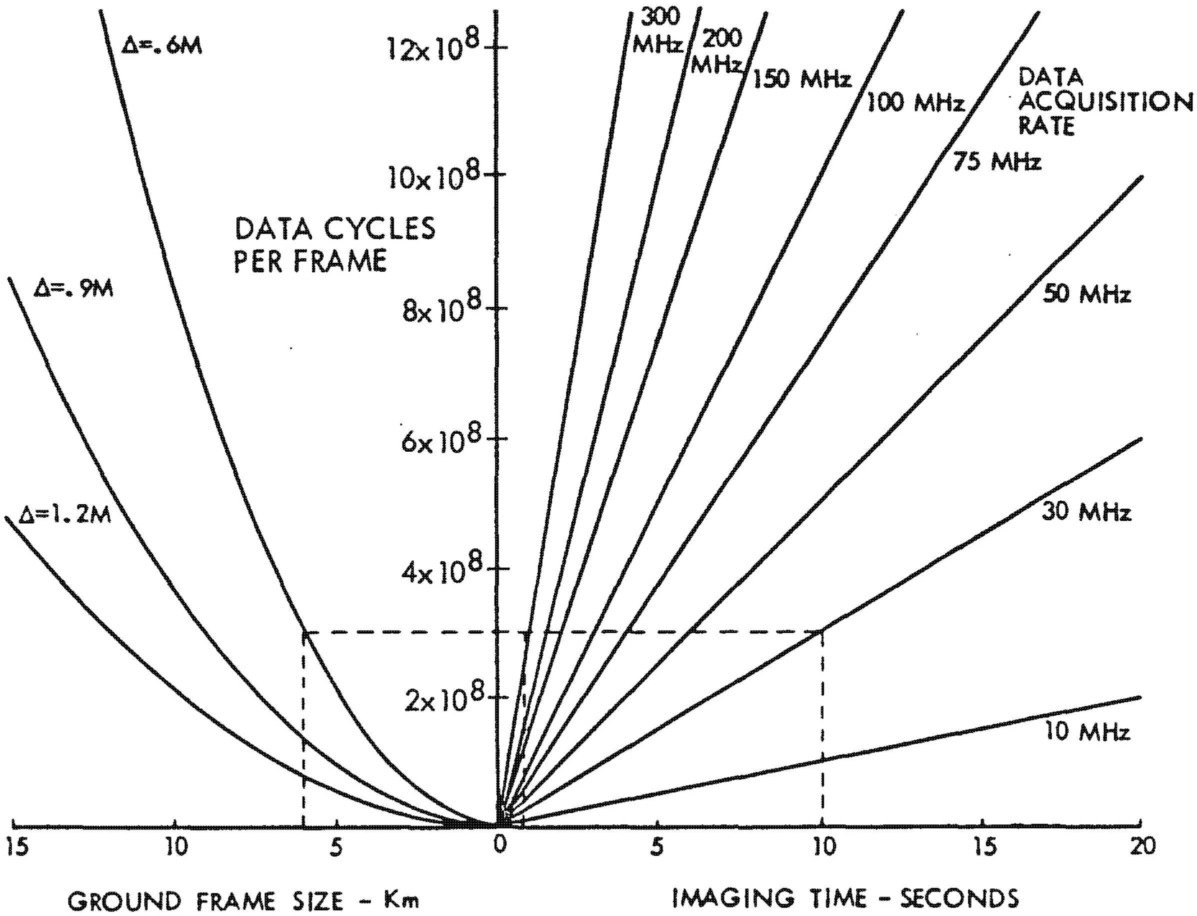

FIGURE 4-17 DATA ACQUISITION RATE

4.7.2 Communication Rate

The data rate (video baseband) that must be handled by the communications subsystem will depend upon the data-transport concept employed. In the direct-readout concepts the output data rate will equal the data acquisition rate if a broom scan is used, or, if a vidicon is used, will depend on the readout time employed.

Figure 4-18 shows the communications baseband width required as a function of ground resolution, number of stored images, and readout time, for those systems that employ onboard data storage. Specific system examples are indicated by the dashed lines which show the maximum readout time available for the given system every 12 hours.

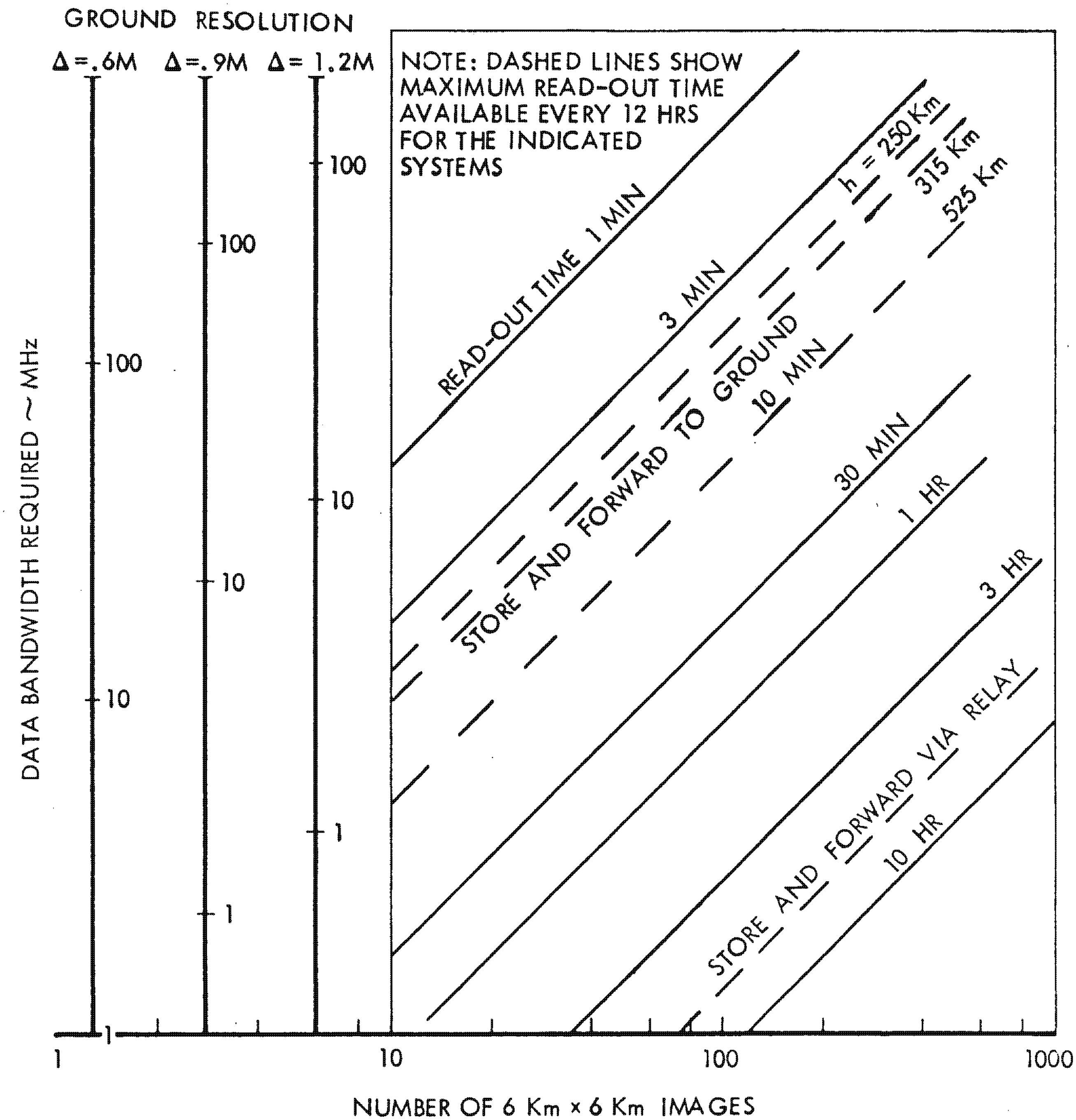

FIGURE 4-18 COMMUNICATIONS DATA RATE FOR STORE AND FORWARD CONCEPTS

5.0 SUBSYSTEM PERFORMANCE CAPABILITY

Achievable system performance will depend primarily on the subsystems used to accomplish the data transport, image acquisition, and sensor pointing functions. The following sections summarize the performance potential of these subsystems as developed in Sections 6 through 10 of Volume II.

5.1 OPTICS

The maximum optical diameter which might be selected is constrained by technological limits. Current high performance optical systems are estimated to be in the [redacted] Exploratory development of an [redacted] diameter optical system has been underway for some time. A maximum optical diameter of [redacted] has been considered to be an upper limit for this study.

Imaging systems were conceived which incorporate a full-aperture, folding flat mirror. These systems were limited to smaller optical diameters because of the difficulty associated with fabricating the large diagonal mirror. A [redacted] system is considered to be the upper limit for a system of this type. A [redacted] diameter system would require a folding flat with a major axis of approximately [redacted]. The manufacturing of such a flat mirror is assumed to be as difficult as manufacturing a [redacted] meter diameter primary mirror.

5.2 IMAGE TRANSDUCERS

Transducer sensitivity and resolution will determine the optical system required to achieve a given ground resolution from any selected altitude. The two types of transducers considered are significantly different in the exposure-time flexibility they offer.

5.2.1 Frame-Type Transducers

The required exposure times as a function of optical diameter for the three frame-imaging transducers are illustrated in Figure 5-1 for a specific set of operating conditions. A requirement of a static resolution of 0.6-meter from 525 kilometers with a minimum scene brightness of 800 foot-Lamberts is assumed for the combinations of diameter and exposure time given by these three operating curves. If a system operating point below these curves was selected, the transducer would not receive sufficient energy to generate the required output signal. Operation in the region below the curves will result in degraded performance. Selection of an imaging system located above the illustrated operating curve for each transducer will result in a margin of additional performance. Some margin of performance is desirable to increase the flexibility of the system to handle off-nominal conditions that will be encountered in most missions.

The region from which an imaging system can be selected to satisfy other performance requirements can be illustrated by constructing a similar parametric constraint diagram for any specific conditions. Specific examples have been developed for resolutions of 0.6, 0.9, and 1.2 meters and for altitudes of 250, 315, and 525 Km. These are illustrated in Volume II, Section 6.

Figure 5-1 shows that a significant advantage can be gained through the use of a thermoplastic transducer. The projected performance characteristics of this image recording material will permit the use of a system operated at the effective diffraction limit. The short exposure intervals will also eliminate the need for differential image motion compensation and will relax the requirements on the attitude control system. Vigorous pursuit of the thermoplastic development is strongly indicated.

The study of various imaging system alternatives has involved the evaluation of other parametric trades in addition to the basic one described here. Consideration has been given to format size requirements, the time required to complete the exposure of a total frame, readout constraints, and total capacity.

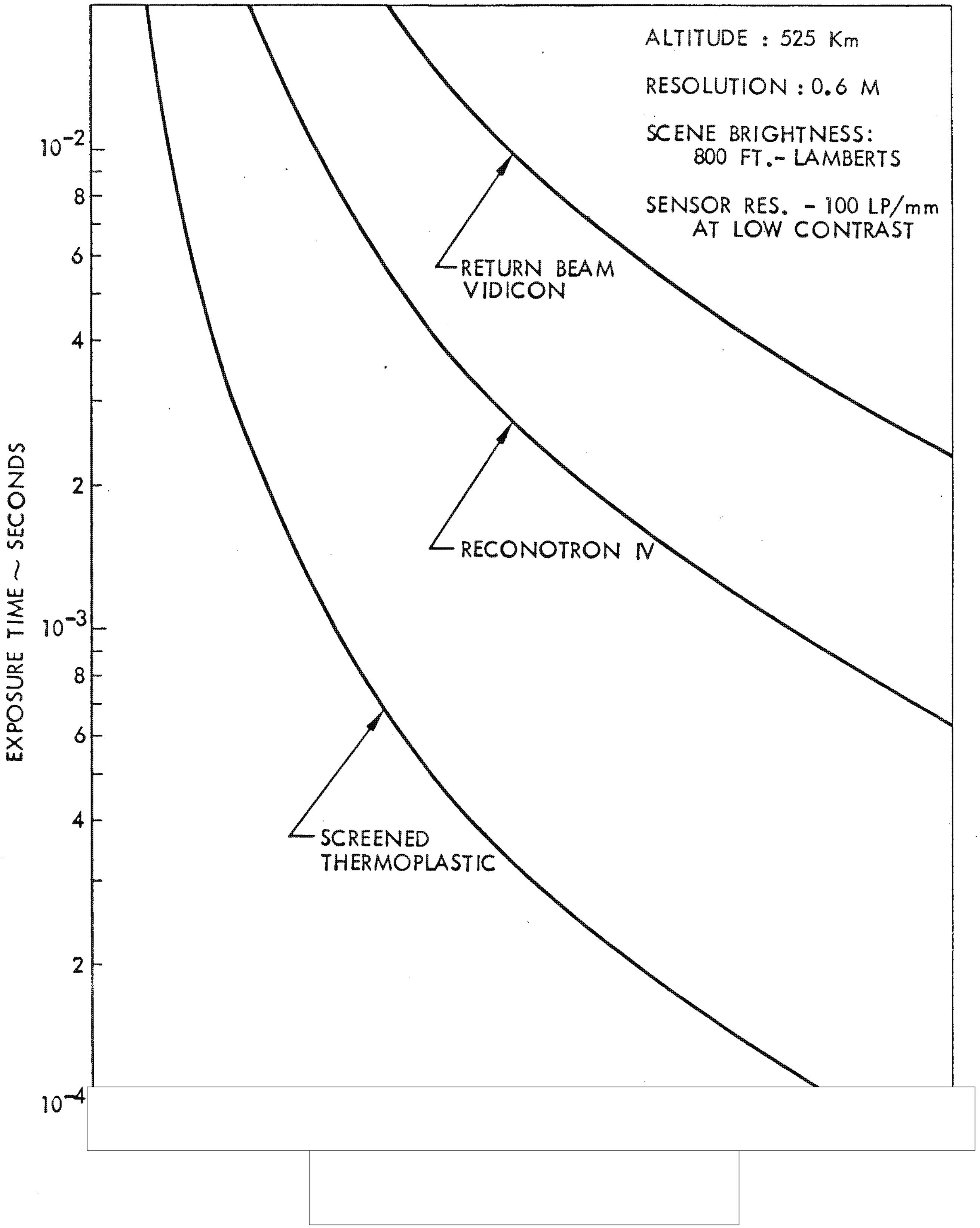

FIGURE 5-1 FRAME IMAGING PARAMETER TRADES

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Vidicon systems are of particular concern since they do not incorporate multiple frame storage. The maximum image acquisition rate that can be realized with the vidicon is dependent on the readout time, the time required for erasing and preparing the vidicon, and the time required for the shutter to complete the exposure. Repositioning of the optical axis can be accomplished during the readout-erase-and-prepare period. The thermoplastic and Reconotron transducers have a multiple-frame storage capability. The total photo interval for these configurations is equal to the shutter actuation time and the repositioning time. Such systems, therefore, offer a greater maximum image acquisition rate.

5.2.2 Broom-Scan Transducer

The broom scan represents an alternative concept for acquiring an optical image. In this concept a line array of discrete detectors is used to generate an equivalent image by periodically sampling the array as its projection scans the ground scene of interest. If scanning is accomplished at satellite velocity, such a system operates like a continuous strip camera. A high data bandwidth and a high detector sensitivity are required for this mode of operation. Slowing the scan is a technique for reducing the technical difficulties involved in the implementation of a broom-scan system. With slowed scan, the scanning rate is reduced by applying an angular rate to the optical line of sight so as to partially track the target. The exposure time (detector integration time) is dependent on the velocity with which the array projection moves across the target scene. The time interval involved in the acquisition of a single broom-scan image is a parameter of significance to the overall system.

The basic parameters of a broom scan-imaging system are interrelated as illustrated in Figure 5-2. The basic trade which exists between optical diameter and imaging time is shown for a system using a quasi-linear array of phototransistors. Three levels of performance are illustrated for operating at

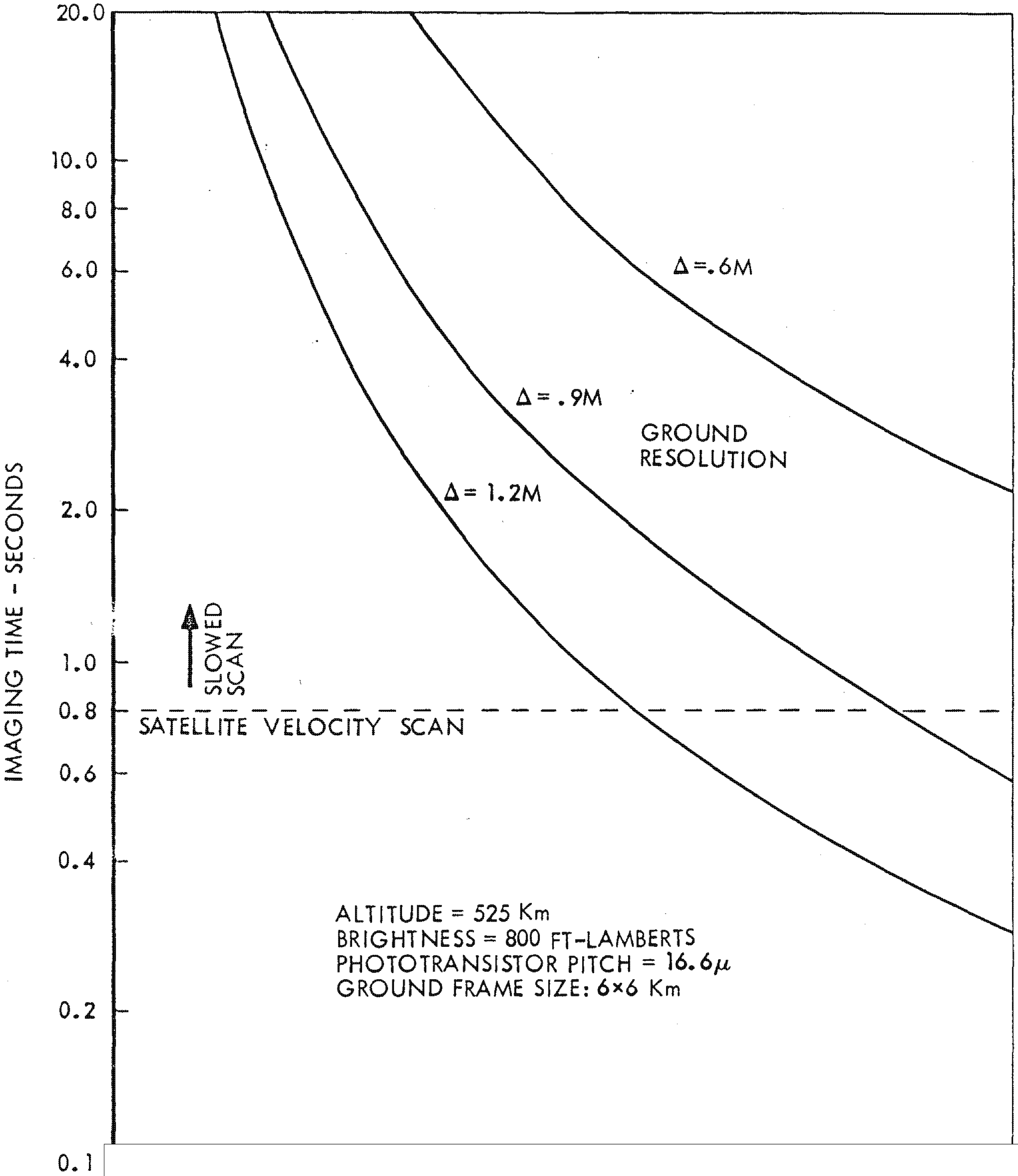


FIGURE 5-2 BROOM SCAN PARAMETER TRADES

an altitude of 525 Km. The selection of a broom-scan imaging system can be made based on the constraints illustrated in Figure 5-2. A family of these diagrams for the operational altitudes and resolutions of interest has been developed for examining the parameters which are pertinent to the various concepts studied.

5.3 ONBOARD DATA STORAGE

The onboard data storage can be provided by three types of recording media: magnetic tape, thermoplastic film, and electrostatic tape (Reconotron). The performance limitations of the equipments associated with each of the three media are given in the following paragraphs.

5.3.1 Magnetic Tape Recording

The use of a multi-channel onboard tape recorder to store the imaging signals provides the capability for transmitting the data at selected times, within reasonable bandwidths. Multiple channels are required to handle the high imaging data rate over the selected target. By use of magnetic tape for storage, many pictures can be accumulated for later recall at a more convenient time and data rate. Transmission time to the ground is extended when the data rate is slowed but the communications bandwidth and required power output can be reduced.

Information can be recorded on magnetic tape at bandwidths of 10 MHz and slightly beyond. Wider bandwidth can be handled by multiple channels. Up to 32 separate channels on a single tape appears to be feasible, permitting a total bandwidth of 320 MHz to be recorded. Current recording techniques employ FM methods for internal processing. The signals are recovered in a continuous data stream without special synchronization requirements, switching transients, or dead time.

The recorded video data rate can be reduced by a ratio as high as 100:1 by slowing down the speed of the magnetic tape recorder. Reasonably sized reels will hold up to an hour of recording (10,000 feet of 4-inch tape) during any one day. This is equivalent to 1,600 6-Km frames with a resolution of 0.6 meter. A recording density on the tape of 1×10^6 cycles per square inch provides this capability. The magnetic tape can be used for up to 1000 replays. This is compatible with a 1000-hour operational life capability for the read/write heads. Incorporation of these advanced features is feasible in a magnetic tape recorder, however an extensive developmental program is required.

5.3.2 Thermoplastic Storage

The resolution of thermoplastic recording has been demonstrated at 100 line pairs per millimeter. Reuse of the plastic media is limited to between 25 and 50 times. If one hundred pictures a day are required for one year with a reuse capability of 36, then 1000 thermoplastic plates will be required. Readout is accomplished by rotating mirror and laser beam. Bandwidths demonstrated by such devices approach 100 MHz with analyzing spots as small as 5 microns. Thermoplastic recording is not developed fully at this time but the concept offers significant potential for future increase in storage capability.

5.3.3 Reconotron

The use of a Reconotron to store the imaging signals provides both a storage and sensor capability in a single instrument. This allows system readout of photographic information at reasonable times and bandwidths. Electron-beam readback of stored data approaches 50 MHz from as many as 4 possible channels.

The limitation on number of pictures is a function of the amount of electrostatic film which can be placed inside the evacuated envelope. Reuse of a given section of storage media appears to be over 1000 times, thus only one day's capacity need be carried onboard.

If a nine-by-nine-inch format can be provided with 120 line pairs/mm, a system can be developed which has [redacted] resolution and a frame size of 9.15 kilometers on a side.

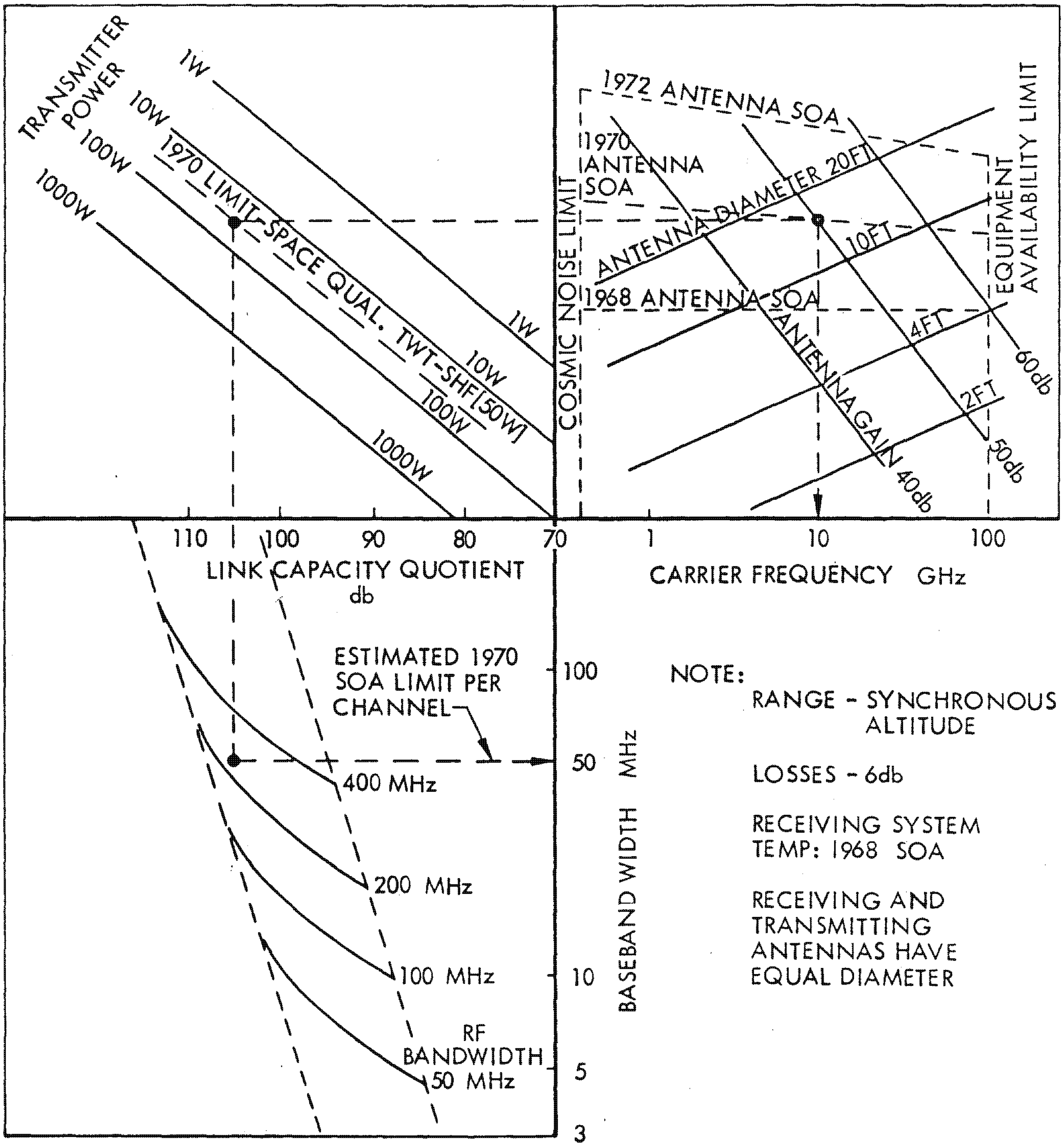
5.4 COMMUNICATIONS

The ability of the communications subsystem to handle the desired data rates will depend primarily upon the data readout alternative used and on the state of the art of communications techniques and equipment. In general, the direct-readout concept will be the most difficult to implement because of the necessary combination of high data rates and long-range inter-satellite links. Such links have yet to be demonstrated in space and will require operating near the state-of-the-art limits in antenna gain, diameter and pointing accuracy, transmitter power and equipment linearity and bandwidth. The store-and-forward-via-relay concept will encounter similar but less difficult problems. The data rate requirements will be less since the time available for readout is much longer. This combined with a more efficient modulation technique will result in lower requirements for transmitter power and/or antenna gain. For both of these alternatives the link from the relay satellite to the ground terminal will be relatively easy to implement. The store-and-forward-to-ground system is by far the easiest to implement from a communications point of view. Although the data rates are relatively high, the communications range is short and the receiving terminal can use high performance, ground-based equipment. The primary problem is to obtain the desired linearity over the relatively wide bandwidth required. This problem can be solved by employing parallel channels. The number of channels is limited only by the increase in system complexity, cost and spectrum requirements.

An estimate for the 1970 time period for the upper limit of performance of alternative communications concepts is given below. The most difficult links to implement in each concept are discussed since these links will determine the capability.

5.4.1 Satellite-to-Satellite Links

Figure 5-3 summarizes the basic trades available between the design parameters associated with an intersatellite communications link. Detailed development of this chart appears in Volume II, Section 8. The relationship



NOTE:

RANGE - SYNCHRONOUS ALTITUDE

LOSSES - 6db

RECEIVING SYSTEM TEMP: 1968 SOA

RECEIVING AND TRANSMITTING ANTENNAS HAVE EQUAL DIAMETER

FIGURE 5-3 BASIC DESIGN TRADES FOR A SATELLITE-TO-SATELLITE LINK between baseband width (data rate) and link capacity quotient is shown for analog modulation.

Similar trades can be developed for digital modems. However, unless data compression is used there is little difference in the required link capacity quotient for the same data rates and output quality. A single-channel bandwidth limit of approximately 200 MHz is estimated for 1970 for the type of link being considered here.

The dated dashed lines indicate the antenna state of the art for the present and expected near future. The region of interest on the chart does not extend below approximately 800 MHz where both the antenna diameters and cosmic noise levels become excessive, nor above approximately 100 GHz where suitable RF equipments are lacking.

Based on the expected 1970 state of the art for transmitter power, RF bandwidth and antenna diameter, the data rate for an inter-satellite link of this communications range will be limited to approximately 50 MHz.

5.4.2 Satellite-to-Ground Links

The data rate that can be handled from the observation satellite to the ground station will be limited primarily by the available spectrum and/or the number of channels that can be provided. The required transmitter power and antenna gain will be well within the state-of-the art. Single-channel baseband data rates between 75 and 100 MHz should be possible for this type of satellite-to-Earth link by 1970. A channel in this case may consist of a number of subchannels multiplexed onto a single carrier.

5.5 POINTING CONTROL

The performance of the optical imaging system depends on the attitude and attitude rate accuracies to which the optical axis is pointed. The optical axis must be repositioned rapidly to accommodate peak rates of image acquisition. The pointing control requirements that support the sensor subsystem are assumed to be provided by the spacecraft attitude control subsystem. The attitude control subsystem characteristics used for predicting pointing performance capability are defined in Volume II, Section 10.2.

5.5.1 Pointing Accuracy

Pointing accuracy is divided into two performance areas; attitude position accuracy and attitude rate accuracy. Attitude accuracy defines the ability to point the optical axis towards an aiming point to place the intended target area within the optical field of view and to compensate for computed image motion. Table 5-I lists the error sources and 1 σ errors based on existing state of art. The 1 σ attitude accuracy required to capture the target area within the field of view is .04 degree. The limiting factors on pointing accuracy are the orbit position error and the attitude reference error.

TABLE 5-I ATTITUDE ERROR SUMMARY

ERROR SOURCE	1 σ ERROR
STAR TRACKERS	.02 Degree
IMU (90° MANEUVER)	.02 Degree
ORBIT DETERMINATION (525 Km Altitude)	.02 Degree
ATTITUDE CONTROL EXECUTION	NEGLIGIBLE
1 σ RSS FOR 45° PITCH, 45° YAW = (Approximately same for all three axes.)	.04 Degree

Attitude rate accuracy defines the ability to control the angular rates of the optical axis relative to the line of sight between aiming point and spacecraft. Rate errors are contributed by attitude rate errors, orbit velocity errors, and attitude reference errors. The estimated 1 σ values of these errors are given in Table 5-II. The limiting factor on attitude rate accuracy is the performance of rate sensors at the present state of art. If fast-imaging sensor development is successful, the rate accuracy requirement will be eased to the extent that it may be met without a rate sensor.

TABLE 5-II ATTITUDE RATE ERROR SUMMARY

ERROR SOURCE	1σ ERROR (Pitch Axis)
RATE GYROS	3 $\widehat{\text{sec}}$/sec
ATTITUDE REFERENCE	2 $\widehat{\text{sec}}$/sec
DIFFERENTIAL IMC	2 $\widehat{\text{sec}}$/sec
RATE CONTROL EXECUTION	4 $\widehat{\text{sec}}$/sec
1σ RSS PITCH RATE ERROR	$\boxed{8\ \widehat{\text{sec}}/\text{sec}}$ *
* Other Axes Somewhat Less	

5.5.2 Repositioning Time

Repositioning time consists of two primary factors:

(1) Spacecraft rigid-body maneuvering time;
(2) Optical structure settling time.

Spacecraft rigid-body maneuver time includes the theoretical time required for a given control authority (acceleration and/or rate limited) to change the spacecraft from one attitude to another. It also includes the control system settling time.

The optical structure imposes a limit on the control acceleration that can be used. If large accelerations are used, the optical structure is deflected beyond the tolerances permitted for image quality. Thus, after the rigid-body attitude is satisfied, the structural vibrations must be allowed to damp. From a simplified treatment of an idealized optical structure having a fundamental resonance frequency of 20 cps and a damping of 1% of critical, it was concluded that control acceleration less than 0.5 rad/sec^2 can be tolerated by the optical system.

Figure 5-4 shows the best repositioning times which are expected under the limitations discussed, [redacted]

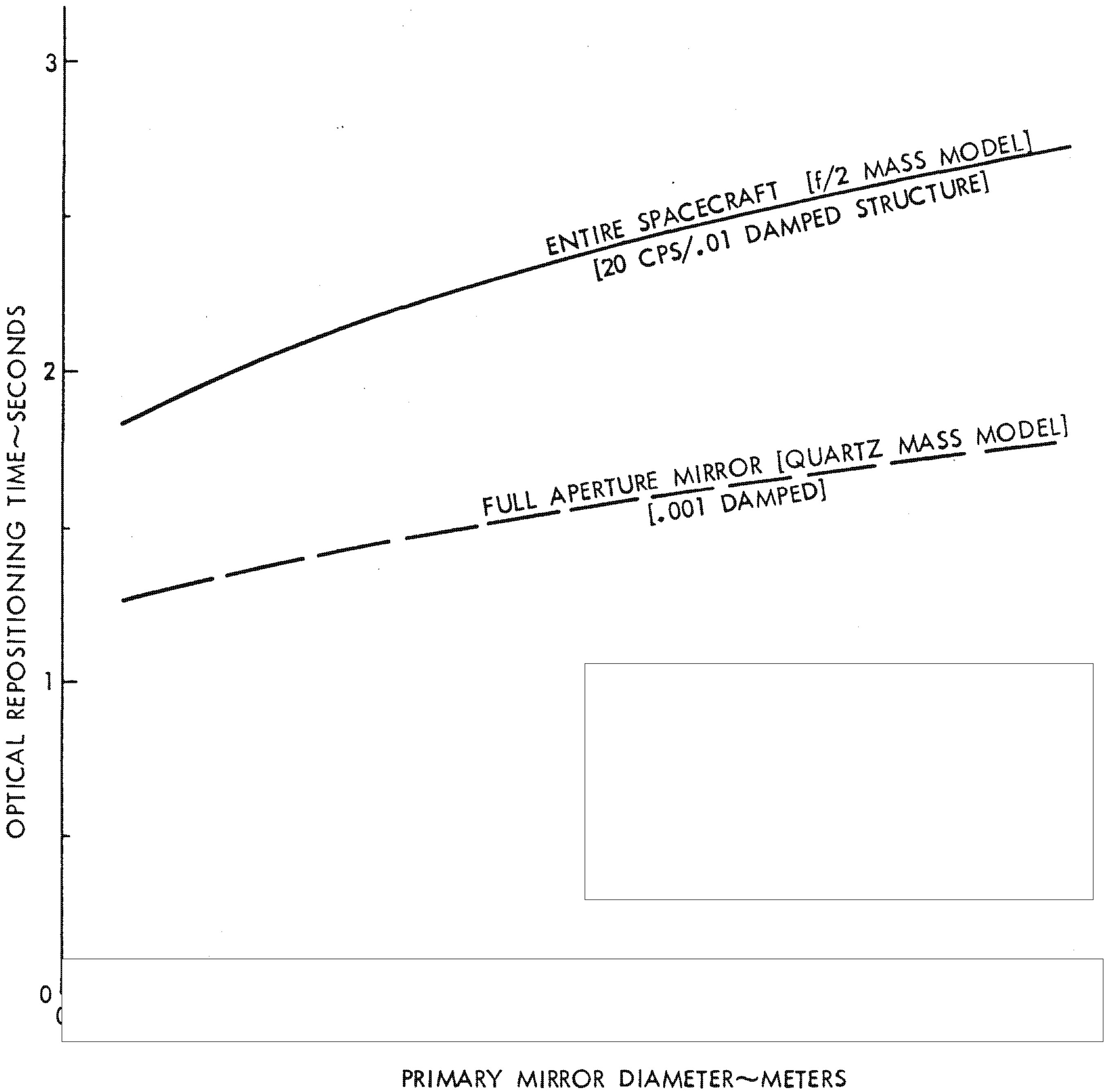


FIGURE 5-4 PREDICTED REPOSITIONING PERFORMANCE

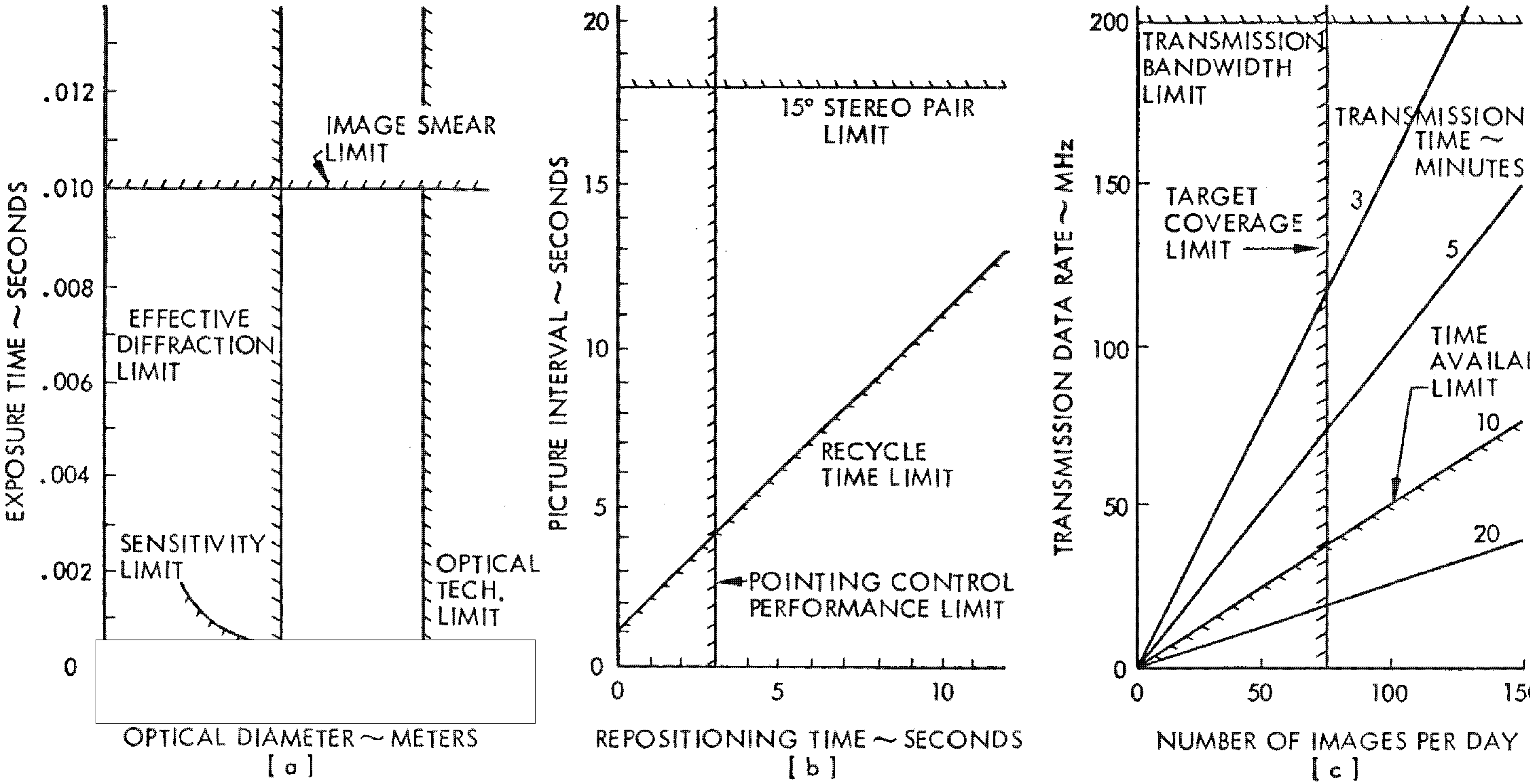
6.0 DESIGN TRADE CONSTRAINTS

Subsystem design values are derived from the interface constraints of other subsystems as well as from the basic mission requirements. These interdependencies among subsystems represent trade-off opportunities in system design. For a frame-imaging system these trade relationships and design constraints exist almost independently for the sensor, data transmission and pointing control subsystems. All three areas are interrelated for a broom-scan system. A graphical treatment of design trade constraints is introduced in this section and applied to subsystem parameters selection for the baseline system examples.

6.1 FRAME IMAGING SYSTEM

The basic trade is between exposure time and optical diameter as presented in Figure 6-1. Performance for a particular transducer (screened thermoplastic) forms the lower boundary shown in Figure 6-1(a). If a system operating point below this curve is selected, the transducer does not receive sufficient energy to generate the required output signal. Operation in this region would thus result in degraded performance. Selection of an imaging system located above the illustrated operating curve for each transducer will result in a margin of additional performance. Some margin of performance is desirable to increase the flexibility of the system to handle off-nominal conditions that will be encountered in most missions.

The region from which allowable imaging systems can be selected is significantly reduced when one considers the various constraints imposed by existing technology, physical limits, and limits imposed by other related subsystems. The upper bound on optical diameter is taken from Section 5.1. The allowable optical diameter is limited on the lower side by diffraction. An "effective diffraction limit" equal to twice that of the theoretical diffraction limit has been assumed. This provides a small margin for residual uncorrected aberrations, manufacturing tolerances, mechanical misalignment, and degradation due to thermal effects.



ALTITUDE = 525 Km, RESOLUTION = .6 M, FRAME SIZE = 6 Km

FIGURE 6-1 FRAME IMAGING SYSTEM DESIGN TRADE CONSTRAINTS

Approved for Release: 2021/04/08 C05098493

An upper limit on the allowable exposure time is indicated in Figure 6-1(a). Such a limit is imposed because of the degrading effects of image motion. The residual error level achievable with image motion compensation is analyzed and reported in Volume II, Section 6. It is shown that a nominal exposure time of 0.01 second will limit the dynamic smear effects to a tolerable level for a system which accomplishes differential image motion compensation. Thus, exposure time represents a tie between the sensor subsystem and the pointing control subsystem.

The repositioning-time trade relationships for the pointing control subsystem are portrayed in Figure 6-1(b). The system performance parameter involved is picture interval, having a minimum value equal to reposition time plus about one second for shutter operation. Target coverage imposes an upper bound on picture interval, as discussed in Section 4.6.4. The lower bound on repositioning time (from Figure 5-4) closes the boundary on permissible design choice for picture interval and repositioning time.

Communication parameter constraints, shown in Figure 6-1(c), are taken from Figure 4-18. A store-and-forward-to-ground concept is assumed for this example, allowing 10 minutes transmission time from a 525-Km orbit altitude. Mission requirements place a lower bound on the number of images to be transmitted. Upper bounds on data bandwidth depend on number of parallel channels used. For a store-and-forward-via-relay concept, the transmission time constraint is greatly relaxed. However, the much greater communication distance leads to a much lower limit on data bandwidth, imposed by RF power and antenna gain constraints.

Output data rate is the principal impact of the sensor subsystem on the storage or communication subsystem. The thermoplastic and Reconotron transducers provide independent scanning of the stored image, allowing data rate to be selected to suit transmission capability. The vidicon readout time is one of the contributors to picture interval. By increasing data bandwidth, readout time is reduced and a corresponding reduction occurs in minimum picture interval until the vidicon recycle time becomes shorter than the pointing repositioning time. The trade is bounded by the bandwidth and storage time capability of the vidicon. Stereo-pair target coverage imposes an upper bound on picture interval, resulting in a completely bounded region within which the choice of subsystem design must lie (Figure 6-2).

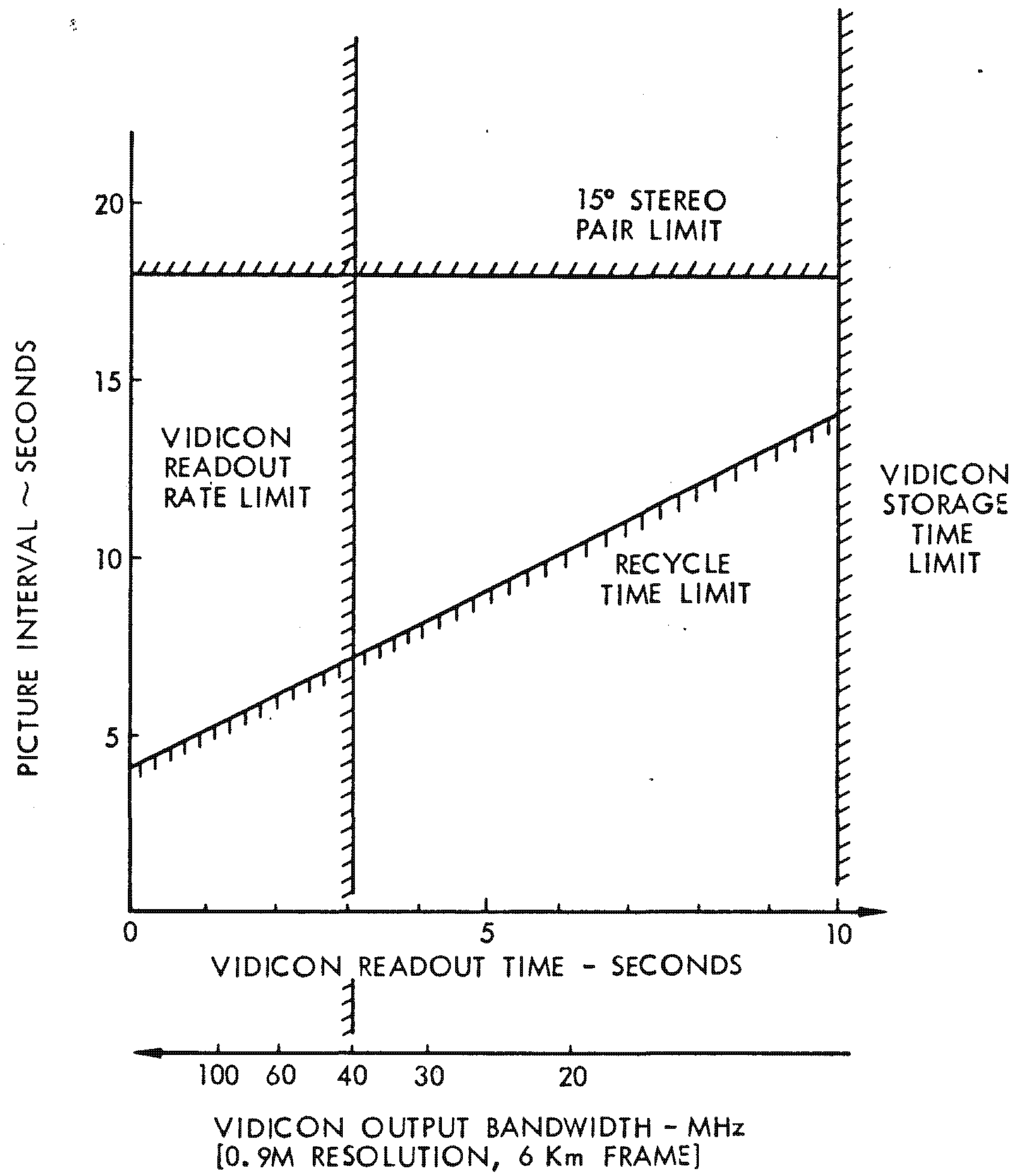
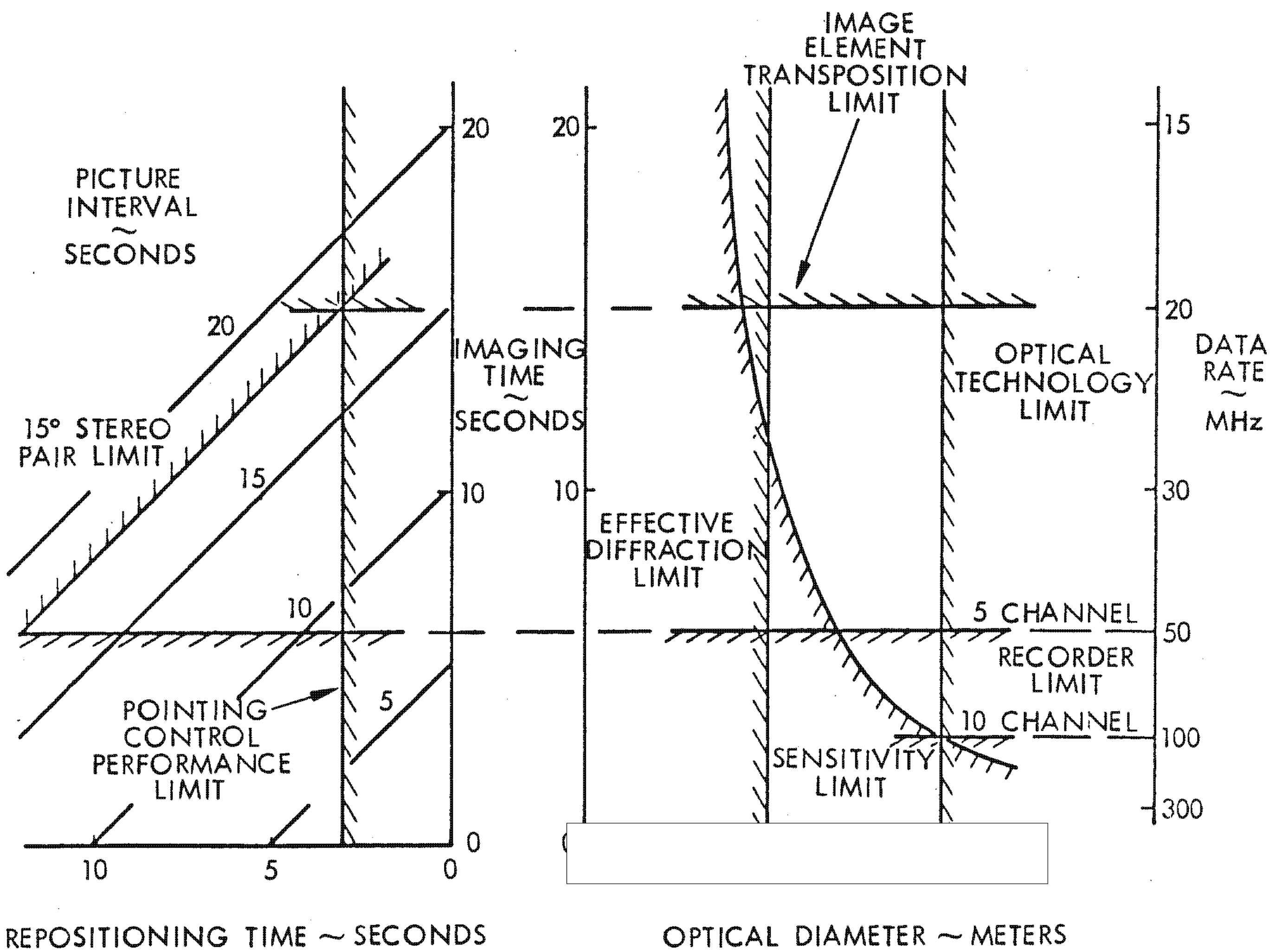


FIGURE 6-2 VIDICON READOUT TIME TRADE

6.2 BROOM-SCAN SYSTEM

Imaging time is a substantial contributor to picture interval in a broom-scan system, resulting in a direct trade relationship between the sensor subsystem and repositioning performance of the pointing control subsystem. In Figure 6-3 this is depicted by bringing together the transducer characteristics from Figure 5-2 and a graph relating picture interval to imaging time and repositioning time. Bounds on picture interval and repositioning time are the same as were introduced for the frame-imaging system.



ALTITUDE = 525 Km, RESOLUTION = .6 M, FRAME SIZE = 6 Km

FIGURE 6-3 BROOM SCAN SYSTEM DESIGN TRADE CONSTRAINTS

The upper and lower limits on optical diameter are identical to those previously discussed for a frame-imaging system. A maximum allowable imaging time is also indicated. The criterion used to establish this upper bound is related specifically to the quasi-linear array. Because of the way the detector elements are arrayed, the dynamic errors associated with the imaging process (attitude rates, pointing errors, etc.) will cause distributed sampling errors. Such errors will introduce sample-transposition errors rather than smear. A discussion of these errors is included in Volume II, Section 6. The upper bound on imaging time has been established by allowing a maximum transposition error of ±1 element along a line of object space transverse to the flight line. Because of the distributed nature of this error, it is difficult to analytically evaluate in a meaningful manner. The established tolerance is considered to be sufficiently stringent to ensure the acquisition of high-quality imagery.

The final element introduced into Figure 6-3 is data rate, also a direct function of the imaging time. The communication link capability (Section 5.4) limits data rate in a direct-readout system. For the store-and-forward concept, input rate capability of the tape recorder (Section 5.3) determines allowable data rate. Communication subsystem trades for the store-and-forward mode are the same as for a frame transducer (Figure 6-1(c)).

7.0 BASELINE SYSTEM SELECTION

The initial step in the development of the baseline system examples is the identification of the alternative methods for implementing the major functions that must be performed. Alternatives for image acquisition, data transport, and pointing control were identified in Section 3.0. From the potential combinations of these alternatives, the selected baseline system concepts are identified in Table 3-I. It is neither possible nor desirable to pursue all possible concepts. However, a number of representative baseline concepts are subjected to more thorough evaluation by definition of the baseline system examples described in Sections 9.0 through 14.0

The second step in the baseline system selection process is the generation of a set of performance options based upon mission requirements analysis. The three key parameters identified in Section 4.0 that have considerable influence on the baseline system design are orbital altitude, resolution, and frame size.

The third step in the process is to generate baseline systems by matching performance options with compatible baseline concepts. The final step is the selection of the command and navigation alternatives for each of the baseline system examples.

7.1 BASELINE SYSTEM EXAMPLES

The baseline systems are generated by matching each selected baseline concept to a combination of altitude, resolution, and frame size. A selected combination of altitude and resolution is referred to as a performance option. Frame size is not specifically included, as a constant 6-Km frame size was assumed for all options.

Five performance options were selected from altitudes of 250, 315, and 525 kilometers and from resolutions of 0.6 and 0.9 meter. Table 7-I is a matrix of the possible baseline systems and shows the performance options selected for each baseline concept.

TABLE 7-I BASELINE SYSTEM SELECTED

	ALTITUDE (Km)	250	315	315	525	525
	RESOLUTION (Km)	.6	.6	.9	.6	.9
SYSTEM CONCEPT	PERFORMANCE OPTION	1	2	3	4	5
A	• BROOM SCAN ARRAY • STORE & FORWARD VIA RELAY • MIRROR AND ROLL			●		
B	• BROOM SCAN ARRAY • STORE & FORWARD TO GROUND • SPACECRAFT ORIENTATION				●	
C	• FRAME - VIDICON • DIRECT READOUT • SPACECRAFT ORIENTATION					●
D	• BROOM SCAN ARRAY • DIRECT READOUT • MIRROR AND ROLL	●				
E	• FRAME - RECONOTRON • STORE & FORWARD VIA RELAY • SPACECRAFT ORIENTATION		●			
F	• FRAME - THERMOPLASTIC • STORE & FORWARD TO GROUND • SPACECRAFT ORIENTATION				●	

The two store-and-forward-to-ground concepts (B and F) were selected at the high altitude so that time available for data transmission to the ground is maximized.

The systems utilizing relay satellites, on the other hand, can operate at the lower altitudes because mutual visibility limitations are eased by proper positioning of the relay. Mirror pointing rather than spacecraft pointing was selected for the 250-Km option and one of the 315-Km options because of the desirability of a streamlined, low-drag configuration in the denser atmosphere.

7.2 COMMAND AND NAVIGATION ALTERNATIVES

For the command function there are three significant alternatives from which choices must be made to complete definition of the baseline systems. These are a stored program, a direct up-date method, and closed loop command.

Stored Program — The program is replaced or updated on a 12- or 24-hour cycle. The program specifies what aiming points to acquire in terms of time and attitude commands. No provision is made for modifying the program until the next replacement time occurs, once it has been sent to and stored onboard the observation satellite. This concept matches the store-and-forward-to-ground data return alternative. No relay satellites are required, as the program would be replaced or updated when the ground control station has direct electromagnetic contact with the observation satellite.

Direct Update — This alternative is similar to the stored program, with the addition of direct updating of the program stored on the observation satellite via a command link through a relay satellite. It is possible that this update link could share the use of a communication satellite serving other users.

Closed-Loop Command — This concept provides the capability for command transmission to the observation satellite based on the imagery received at the ground station. It requires the direct-readout data return alternative. The command link shares the relay satellite being used for direct readout of the acquired imagery. This closed-loop capability applies to commands only. Closed-loop control from a ground station was not considered because of the time lags inherent in the long-distance electromagnetic path.

Four alternatives have been identified for the navigation function. These are ground tracking, landmark observation, tracking through relay satellite, and onboard accelerometers.

Ground Tracking — The observation satellite is tracked from one or more ground tracking sites.

Landmark Observation — Updating the knowledge of the observation satellite's position and attitude is accomplished by observing known-location landmarks in returned imagery. Satellite navigation error is derived from discrepancies between commanded line-of-sight orientation and actual line-of-sight orientation as derived from returned imagery containing known landmarks. This data is used to update conventional ground tracking.

Tracking Through Relay Satellite — Tracking of the observation satellite is accomplished by ranging along the communication link from the ground through a relay satellite to the observation satellite. Both range and range rate are measured. This requires a relay satellite that is visible to the ground terminal a large portion of the time. Since the known-location relay satellite is in line-of-sight contact with the observation satellite much of the time, ranging between the two satellites permits tracking the observation satellite over large portions of its orbit using only one ground tracking station.

Onboard Accelerometer — An onboard accelerometer is used to measure drag decelerations on the observation satellite. This data, telemetered to ground control, provides a substantial reduction in drag uncertainty. This data is used to augment the normal ground tracking.

Table 7-II shows the command and navigation concept elements selected for each of the six baseline systems. Ground tracking alone is sufficient for the 525-Km altitude selections. At the lower altitudes ground tracking is augmented by one of the alternative methods.

TABLE 7-II COMMAND AND NAVIGATION SELECTION

SYSTEM CONCEPT		ALTITUDE (Km)	RESOLUTION (m)	NAVIGATION	COMMAND
A	. BROOM SCAN ARRAY . STORE & FORWARD VIA RELAY . MIRROR & ROLL	315	.9	ONBOARD ACCELEROMETER WITH GROUND TRACKING	DIRECT UPDATE OF STORED PROGRAM
B	. BROOM SCAN ARRAY . STORE & FORWARD TO GROUND . SPACECRAFT ORIENTATION	525	.6	GROUND TRACKING	STORED PROGRAM
C	. FRAME — VIDICON . DIRECT READOUT . SPACECRAFT ORIENTATION	525	.9	GROUND TRACKING	CLOSED LOOP COMMAND WITH STORED PROGRAM
D	. BROOM SCAN ARRAY . DIRECT READOUT . MIRROR & ROLL	250	.6	LANDMARK OBSERVATION WITH GROUND TRACKING	DIRECT UPDATE OF STORED PROGRAM
E	. FRAME — RECONOTRON . STORE & FORWARD VIA RELAY . SPACECRAFT ORIENTATION	315	.6	TRACKING THROUGH RELAY WITH GROUND TRACKING	DIRECT UPDATE OF STORED PROGRAM
F	. FRAME — THERMOPLASTIC . STORE & FORWARD TO GROUND . SPACECRAFT ORIENTATION	525	.6	GROUND TRACKING	STORED PROGRAM

8.0 BASELINE SYSTEM DESCRIPTION (GENERAL)

The baseline systems selected in Section 7.0 are described in some detail in the following six sections. System concepts and operational characteristics are identified as are the major system elements and functional subsystems. Weight, power, and reliability budgets are presented for each of the selected systems and the major subsystem performance specifications are listed.

Although the baseline systems have different characteristics in general, they share some common system constraints and are based on similar ground rules. This section lists these constraints and ground rules.

The configuration and characteristics of each of the six systems are based upon the selected baseline concepts and performance criteria, but are not necessarily optimum in design. Much additional effort is required to optimize the design of each of the systems. However, the systems are believed feasible and the specifications as tabulated are attainable.

8.1 BASELINE SYSTEM SUMMARY

Pertinent characteristics of the six selected baseline systems are summarized in Table 8-I. The differences between the systems are primarily related to the concept chosen for data transport. One variable that is not readily apparent is the elapsed time from image acquisition until transmission to the ground terminal. Direct readout, by definition, approaches zero elapsed time. Store and forward via relay requires approximately one hour. Store and forward to ground may take as long as 14 hours before the data can be read out. These characteristics are reflected to some degree by the data transmission rates. The store-and-forward-via-relay systems have much smaller data rates than do those systems using the other two data-transport concepts.

Because of the short transmission range and the absence of a relay in the store-and-forward-to-ground systems, the RF transmission subsystems are relatively simple. At the other extreme are the direct-readout systems which require high data rates, large antennas, and high-powered RF transmitters.

TABLE 8-1 SUMMARY OF BASELINE SYSTEMS

PARAMETER	BASELINE SYSTEM A	B	C	D	E	F
Image Transducer	Broom Scan Array	Broom Scan Array	Vidicon	Broom Scan Array	Reconotron	Thermoplastic
Data Storage	Magnetic Tape	Magnetic Tape	None	None	Reconotron	Thermoplastic Film
Data Transport	Store & Forward via Relay	Store & Forward to Ground	Direct Readout	Direct Readout	Store & Forward via Relay	Store & Forward to Ground
Pointing Control	Mirror & Roll	Spacecraft Orientation	Spacecraft Orientation	Mirror & Roll	Spacecraft Orientation	Spacecraft Orientation
Orbit Altitude	315 Km	525 Km	525 Km	250 Km	315 Km	525 Km
Resolution	0.9 Meter	0.6 Meter	0.9 Meter	0.6 Meter	0.6 Meter	0.6 Meter
Ground Frame Size	6 x 6 Km	6 x 6 Km	6 x 6 Km	6 x 6 Km	6 x 6 Km	6 x 6 Km
Picture Interval	8 Seconds	18 Seconds	9 Seconds	11 Seconds	6.4 Seconds	6.4 Seconds
Data Rate	8.0 MHz	37.5 MHz	26.6 MHz	33.7 MHz	8.0 MHz	37.5 MHz
Optical Diameter						
RF Power	40 Watts	1 Watt	200 Watts	200 Watts	40 Watts	1 Watt
Antenna Size						
Orbitkeeping Fuel	310 Lbs	None	None	1250 Lbs	450 Lbs	None
Weight Est.	5345 Lbs	4850 Lbs	5290 Lbs	4595 Lbs	4025 Lbs	4795 Lbs
Equiv. MTTF	4.7 Years	5.0 Years	5.4 Years	5.4 Years	5.8 Years	5.8 Years
Relay Satellite	Stationary Orbit	None	Polar Orbit	Polar Orbit	Stationary Orbit	None
System Cost Estimate						

Redundancy is incorporated into the design of each observation spacecraft to bring the equivalent mean time to failure to approximately [redacted] Design considerations include sufficient margin in expendables to preclude depletion in less than one year.

The system cost summary is based upon a two-year research and development phase and a [redacted] operational phase. The program schedule is shown in Figure 8-1. Government systems management and engineering costs are not included in the listed system cost.

8.2 GENERAL SYSTEM GROUND RULES

Definition of a point design requires a list of requirements and design assumptions. The list to be presented in this section is generally applicable to all six baseline systems. Exceptions and additions to the ground rules will be stated in the discussion of each selected system.

8.2.1 Operational Requirements

Requirements affecting system operation are:

(1) Each observation satellite has two optical sensor subsystems.

(2) The low-resolution sensor has a field of view five times that of the high-resolution sensor but has only one-fifth the ground resolution. Therefore, the number of data cycles per frame is the same for both sensor subsystems.

(3) The daily image capacity of each of the selected baseline systems is 75 images. This capacity is made up of 55 high-resolution and 20 low-resolution images; 36 of the high-resolution images are assumed to be stereo pairs.

(4) The angular separation between stereo pairs is 15° except where noted.

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8.2.2 Operational Design Assumptions

System operational design assumptions are as follows:

(1) High- and low-resolution sensors have equal imaging times and therefore have equal data rates.

(2) The transducers for the two sensors are separate but identical.

(3) The two sensors do not operate simultaneously.

(4) Orbitkeeping is not used for the observation satellites at the 525-kilometer altitude.

(5) [redacted]

8.2.3 Communications Link Design Assumptions

Assumptions made to permit sizing the communications equipment are as follows:

(1) A single ground terminal for both data reception and satellite command and control is located [redacted]

(2) Imagery data transmissions are all in non-encrypted analog format.

(3) All commands for observation and relay satellites are encrypted, both when transmitted from the ground to the satellites and when returned for verification.

(4) 60-foot diameter antennas are used at the ground terminal. The receiving system effective noise temperature is 290°K.

(5) The signal-to-noise ratio requirement for the data link is 30 db at the detector input.

(6) The relay satellite receivers all have 6 db noise figures.

(7) Communications equipment and frequency selection are based upon 1970 state-of-the-art hardware.

8.2.4 Reliability and Cost Assumptions

Assumptions made for the system costing exercise are as follows:

(1) The design life of the observation satellites is [redacted]

(2) The design life of the relay satellites is [redacted]

(3) Sufficient redundancy is provided throughout the system where required to achieve design life goals.

(4) For the lower-altitude systems where orbitkeeping is required propellant weights are increased by 25 percent so that system life will not be limited by expendables.

(5) Program go-ahead occurs after the feasibility of each system element and subsystem has been proven.

(6) The research and development phase covers a two-year period.

(7) Flight hardware fabrication and acceptance testing requires one year [redacted]

(8) The total system operational phase covers [redacted]

(9) [redacted] observation satellites are launched.

(10) [redacted] relay satellites are launched for those systems requiring relays.

Figure 8-1 depicts the assumed Operations Plan for all six baseline systems, except relay satellites are not required for systems B4 and F4.

8.3 MAJOR SUBSYSTEMS

Each of the six baseline systems consists of 9 to 12 major subsystems. A block diagram of each of the subsystems depicting appropriate interfaces is included in the technology support material of Volume II. The major subsystem performance specifications for each of the selected baseline systems are included in the system descriptions in the following six chapters of this document. A list of the major subsystems follows.

Optical — Mirrors, lenses, shutters, supporting structure and alignment mechanisms.

Transducer — Sensor elements, sensor scanning, readout, and associated data conditioning circuitry. There are basically four types of transducer subsystem, only one of which is used in each baseline system:

(1) Quasi-linear phototransistor array (Broom);

(2) Return-beam vidicon;

(3) Reconotron;

(4) Screened thermoplastic film.

Magnetic Tape Storage — Magnetic tape, tape transport mechanism, and record and playback and control electronics.

Communications — Modulators, frequency converters, transmitters, receivers, and antennas.

Ground Terminal — Data receiver, spacecraft tracking, telemetry receiver, and command generation.

Ground Reconstruction — Data storage and processing to produce hard copy prints.

[redacted]

Command and Control Computer — Digital computer that generates internal control signals from stored commands and onboard measurements.

Propulsion — Tankage, thrusters, and propellant for orbitkeeping and [redacted] desaturation.

Primary Power — Solar array, storage batteries, charging circuits, and power conditioning.

Relay Satellite — Data repeater, and supporting subsystems.

Command and Telemetry — Encoders, decoders, command security equipment, and associated circuitry.

9.0 BASELINE SYSTEM A

System A consists of a single observation satellite in a near-polar circular orbit at a 315-kilometer altitude, a single relay satellite in a stationary orbit, a single ground receiving terminal located [redacted] and a ground reconstruction facility in the vicinity of the ground terminal. The mission objective is to acquire and relay to the ground reconstruction facility 55 images per day at a resolution of .9 meter and a frame size of 6 x 6 kilometers, and 20 images per day at a resolution of 4.5 meters and frame size of 30 x 30 kilometers.

The operational concept of System A is summarized as follows:

Data Acquisition — The optical transducer is a broom-scan array consisting of phototransistor elements.

Data Onboard Storage — The acquired imagery data is stored on magnetic tape.

Data Transport — The imagery data is read out of the magnetic tape and transmitted, on command, through the relay satellite to the ground terminal.

Pointing — The optical axis is directed to the target by positioning a full-aperture mirror in the fore-aft direction and by rolling the spacecraft.

Command — The onboard stored program can be periodically updated as required through the relay satellite.

Navigation — Position information is acquired periodically by tracking from ground stations. Drag information from an onboard accelerometer is transmitted to the ground and used in orbit prediction.

A simplified system block diagram is shown in Figure 9-1. The performance specifications for each of the subsystems are presented in Section 9.6.

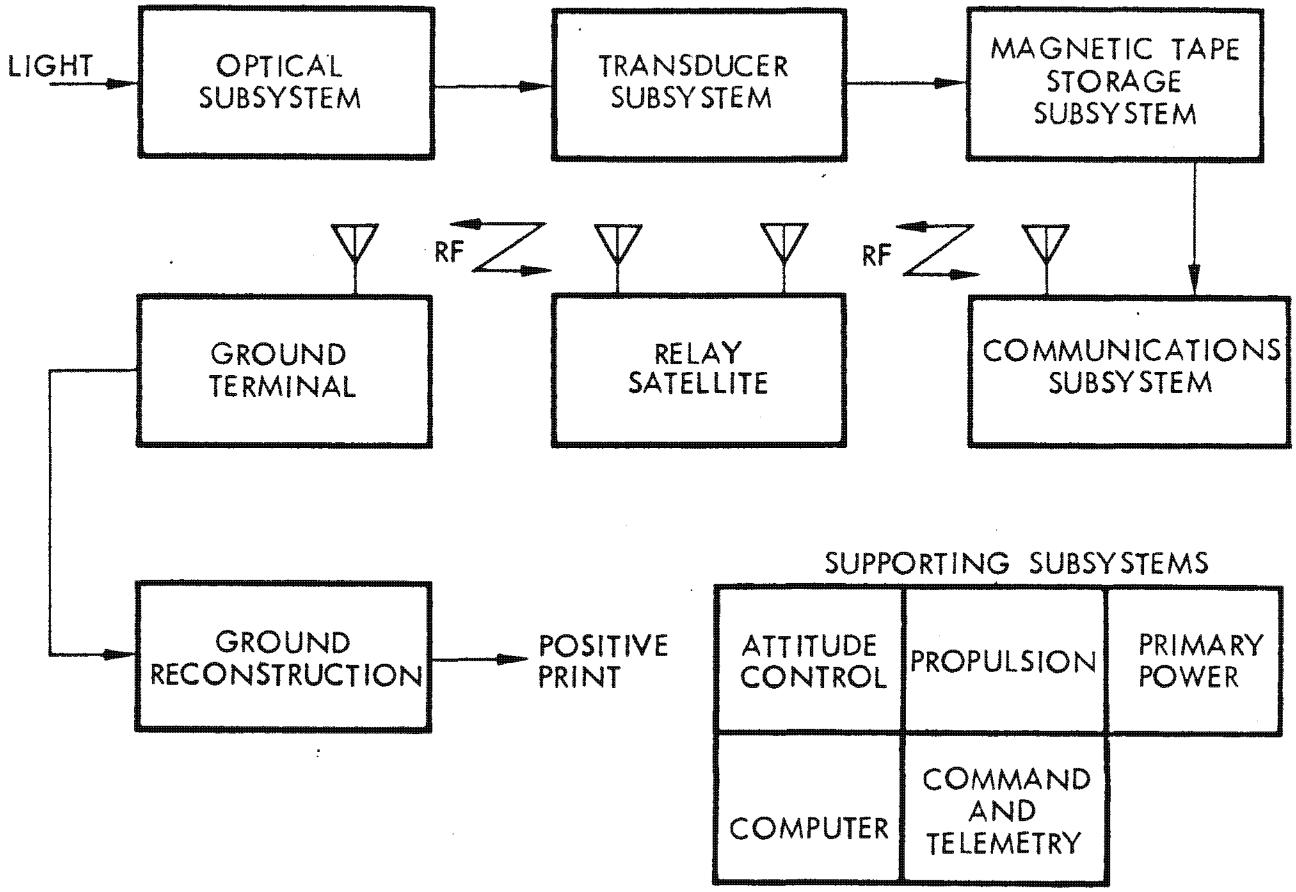

FIGURE 9-1 SIMPLIFIED SYSTEM BLOCK DIAGRAM
BASELINE SYSTEM A

9.1 SYSTEM DESIGN PARAMETERS

The available design freedom for this concept is depicted in Figure 9-2. The constraints imposed by mission requirements and subsystem capability are shown by boundaries, as discussed in Section 6.

A data acquisition rate of 30 MHz is well within the bandwidth capability of a four-channel magnetic tape recorder. The slowed scan (4.5-second imaging time) dictated by this choice results in considerable latitude in optical diameter, as shown in Graph b. The design choice indicated is well above the sensitivity limit and permits operation with a minus-blue filter at a scene brightness of only 400 ft-lamberts. The selected optical diameter of 1.2 meters is well within the assumed state of the art for a full-aperture folding mirror.

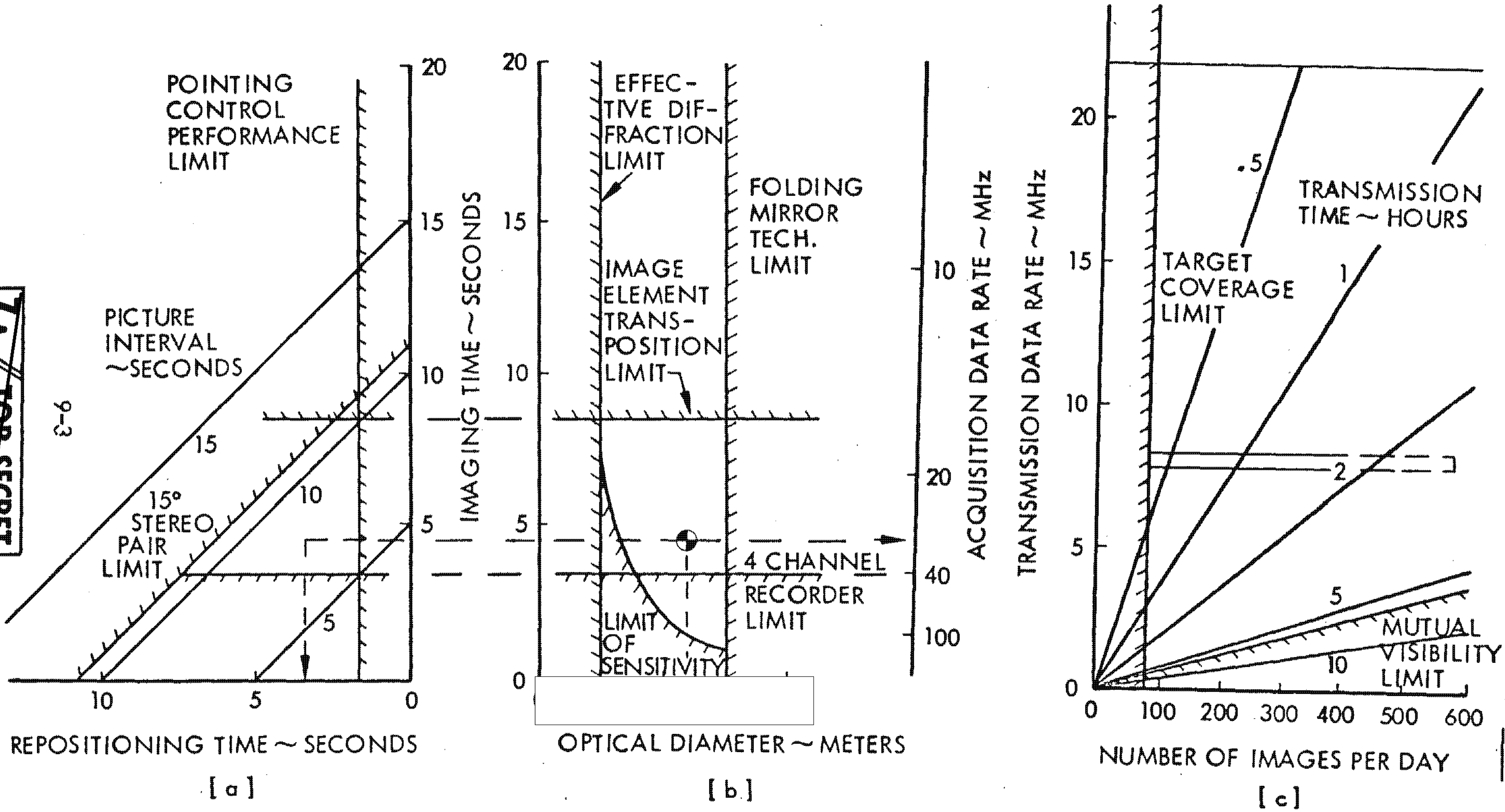

FIGURE 9-2 DESIGN TRADE CONSTRAINTS - BASELINE SYSTEM A

In Graph a the design choice of 3.5 seconds for repositioning time leaves significant margins for pointing control performance and picture interval.

The choice of 8 MHz as the transmission data rate allows the required 75 images to be read out in 21 minutes. Because the store-and-forward-via-relay concept leads to an available transmission time of six hours, substantial capacity growth is available, as suggested by the bar in Graph c. Since the tape recorder can be read out on each orbit revolution, the system capacity is limited by acquisition rate (picture interval) rather than storage or transmission capability.

Design parameters for System A are summarized in Table 9-I.

TABLE 9-I SYSTEM A DESIGN PARAMETERS

ORBIT ALTITUDE	315 Kilometers
RESOLUTION	0.9 meters
FRAME SIZE	6 Kilometers
IMAGING TIME	4.5 Seconds
DATA ACQUISITION RATE	30 MHz
REPOSITIONING TIME	3.5 Seconds
PICTURE INTERVAL	8 Seconds
OPTICAL DIAMETER	
DATA TRANSMISSION RATE	8 MHz

9.2 SYSTEM OPERATION

The system is designed to acquire and relay to ground via relay satellite 75 images per day. These images are relayed to the ground within a few hours of acquisition. In most instances the data can be read out within one hour. The normal mode of operation allows data transmission on practically every orbit, as the relay satellite, in its orbit at 100° West Longitude, is continuously visible to the ground station. The observation satellite is visible to the relay satellite approximately one-half of each revolution.

The super high frequency (SHF) communications antenna is pointed toward the relay satellite with a torquer controlled by the attitude and position reference information and by the error signal developed by the auto-track circuitry. The antenna emits a pilot tone to assist the receive antenna of the relay satellite in tracking.

The two optical sensor systems use identical phototransistor transducers and use full-aperture, flat, folding mirrors for beam bending. The low-resolution system has its primary axis boresighted to the high-resolution system axis.

The two mirror-drive servo systems are controlled by a common drive signal with the assumption that the high- and low-resolution systems need not be used simultaneously. A study of the requirements indicates that this assumption is reasonable. The system is designed so that data need not be transmitted when imagery is being taken. All spacecraft appendages, except mirrors, are held fixed during image-taking periods to avoid motion disturbances.

9.3 ORBIT AND NAVIGATION

At 315-Km altitude the orbitkeeping requirement is not severe. To maintain the orbit between altitudes of 310 and 325 Km, velocity control of approximately 25 feet per second is necessary every 3 weeks. In a year's time this requires propellant equal to only 5 percent of the initial satellite weight. To provide a margin, the spacecraft fuel capacity is increased by 25 percent.

Orbit position accuracy of one kilometer is required. The variable is in-track time with one kilometer representing about .13 second. This accuracy can be maintained if the orbit ephemeris is updated by a single tracking site ranging directly through the observation satellite and by obtaining integrated drag data from an onboard accelerometer several times per day. If the accelerometer should fail it is necessary to utilize a second, and perhaps a third, tracking station to maintain the same position and time accuracy.

It appears that an orbit fix can be calculated within nine hours of a velocity control maneuver to an accuracy that again permits high-resolution imagery of targets on command.

A summary of the orbit and navigation considerations is listed in Table 9-II.

TABLE 9-II SYSTEM A ORBIT AND NAVIGATION DATA

ORBIT: ALTITUDE	315 Km
INCLINATION	96.7 Degrees (Sun Synchronous)
GROUND VISIBILITY TO STATION AT [redacted] LATITUDE	5 Minutes Minimum in a 12-hour period at a 5° elevation angle.
COVERAGE PERIOD ABOVE 30° LATITUDE	7 Days at a view angle of ±45°
ORBIT DECAY	Approximately 0.7 Km per day.
ORBIT VELOCITY CONTROL	Every 20 Days (310 - 325 Km)
POSITION ACCURACY	±1.0 Km
UPDATE ORBIT EPHEMERIS	Single tracking site; onboard accelerometer data being returned 2 to 3 times per day.
ACCURATE ORBIT DETERMINATION	Within 5 to 6 revolutions following velocity maneuvers.

9.4 SYSTEM RF LINKS

The data stored on the magnetic tape in the observation satellite is transmitted through an SHF transmission system with an effective radiated power (ERP) of approximately 57 dbw in the center of the main beam. The 75 stored images can be transmitted through the relay satellite to the ground terminal in 21 minutes at a baseband rate of 8 MHz. Data transmission is initiated by ground command through the relay satellite. A backup mode of data transmission is available in case of failure of the primary system. This alternate method utilizes the command and tracking S-band link directly with the ground, and a secondary low-power transmitter. Mutual visibility between the observation satellite and the ground terminal is available about 12 minutes per day so that only a portion of the daily storage capacity, or 40 images, can be transmitted to the ground each day in this secondary mode of operation.

The command, telemetry, and tracking link between the relay satellite and ground terminal is operated at S-band. The relay satellite is always visible to the ground terminal so does not require additional tracking facilities. The observation satellite, on the other hand, would require additional tracking sites if the onboard accelerometer was not supplying accurate drag data.

Figure 9-3 is a simplified representation of the system RF links.

9.5 OBSERVATION SPACECRAFT CHARACTERISTICS

The satellite is configured so that the solar array is body mounted to reduce the drag and therefore the weight of the orbitkeeping propellant. The five-foot diameter communications antenna extends from the aft end so that additional solar cells can be mounted on the forward end (southern end on the sunlit side of the Earth) to provide additional power while taking imagery in the northern latitudes during the winter months.

The fuel tanks are located near the roll-axis so that fuel depletion will not appreciably affect the pointing control characteristics. The orbitkeeping thrusters are aligned parallel to the vehicle longitudinal axis and the [redacted]

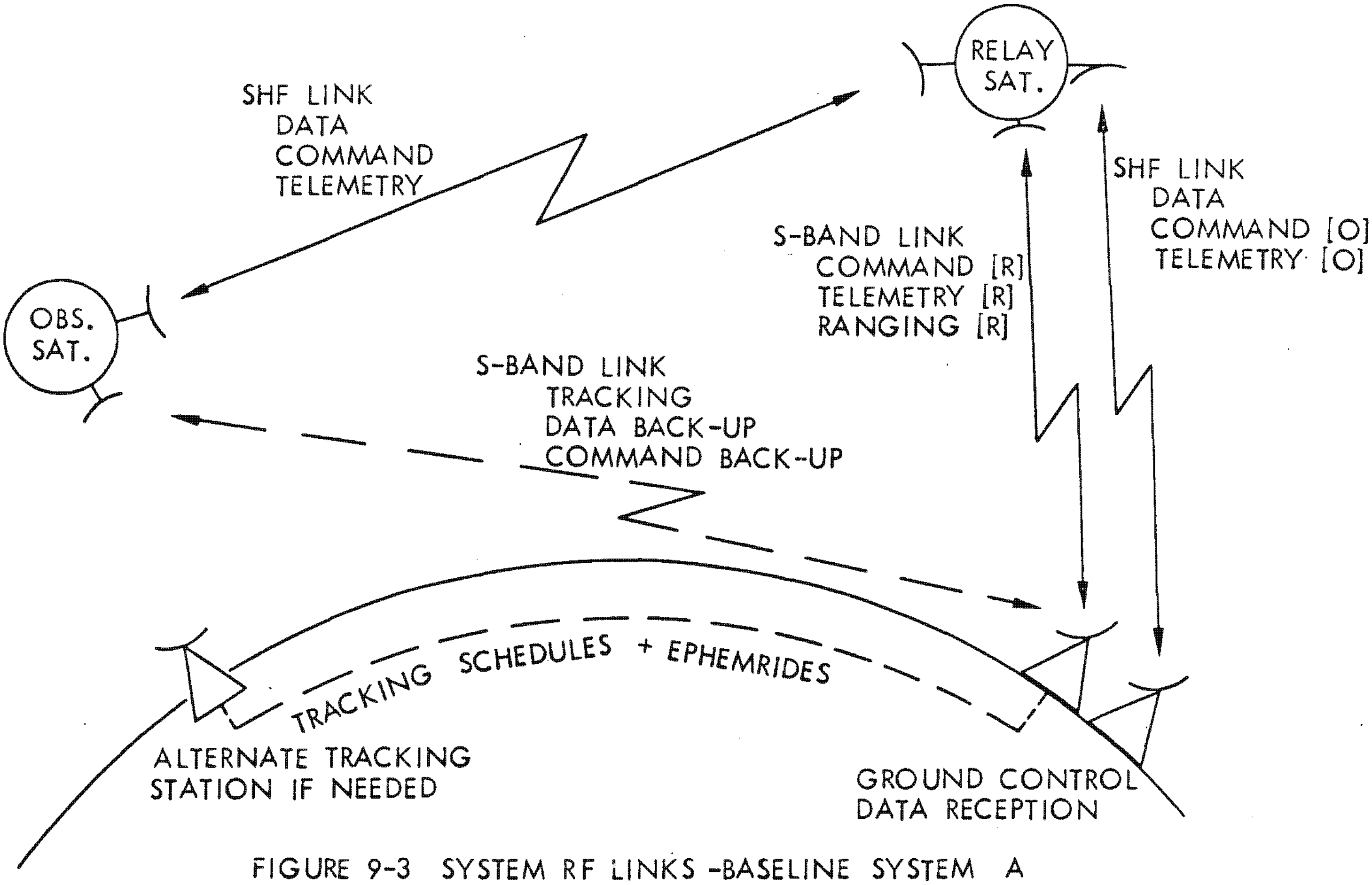

FIGURE 9-3 SYSTEM RF LINKS -BASELINE SYSTEM A

desaturation thrusters are mounted radially at the base end of the spacecraft. These thrusters share common tankage and propellant systems with the orbit-keeping thrusters.

The high-gain communications antenna has hemispherical coverage capability and is pointed to within one-half of a degree of the desired aiming point, using reference signals from the command and control computer. Fine pointing is accomplished by an auto-track mode using signals received from the relay satellite.

The satellite may be launched by either a Titan IIIB/Agena (N_2O_4) or an Atlas SLV-3B/Agena (N_2O_4) with a lengthened OAO shroud. The spacecraft structure is designed so that the launch loads are taken through the aft end with the communications antenna folded during launch operations.

The solar array design has not been optimized, and it may be that a trade study would indicate the desirability of less array area and a set of rotatable paddles. With oriented paddles the stationkeeping propellant weight would increase proportionately to compensate for the increased drag area.

Table 9-III summarizes the major characteristics of the observation spacecraft.

TABLE 9-III OBSERVATION SPACECRAFT CHARACTERISTICS - SYSTEM A

LENGTH	Overall Excluding Antenna	37 30	Feet Feet
DIAMETER (Maximum Lateral Dimension)		7	Feet
WEIGHT (Initial Injection)		5345	Pounds
OPTICAL DIAMETER	High Resolution Low Resolution	[redacted]	Meters Meters
OPTICAL TRANSDUCER	Photo-Transistor Array		Elements
DATA STORAGE CAPACITY	Cycles Images	2×10^{10} 150	
COMMUNICATIONS	Information Bandwidth Carrier Frequency RF Power Antenna Size Effective Radiated Power Transmission Time/Day	8 [redacted] 40 5 59 21	MHz Watts Feet dbw Minutes
ATTITUDE CONTROL		[redacted]	
VELOCITY CONTROL	Bipropellant Thrusters		
SOLAR ARRAY OUTPUT (Normal Incidence)		2,600	Watts

9.5.1 Weight and Power Summary

Table 9-IV lists the weight and power required by each of the major subsystems. The two optical sensors have identical transducer subsystems of approximately 50 pounds each. In addition, 400 pounds of the weight included for the high-resolution optical subsystem total is devoted to the full-aperture mirror and associated tilting mechanism. The magnetic tape storage total includes redundant recorders and multiplexers. Primary power weight includes the solar array, power conditioning equipment, and the nickel-cadmium storage batteries.

The listed power values represent maximum watts drawn by each subsystem, but are not to be added for system totals. Simultaneous operation of all subsystems does not occur. No power is allotted for optical-subsystem thermal control.

TABLE 9-IV OBSERVATION SPACECRAFT WEIGHT AND POWER SUMMARY - SYSTEM A

SUBSYSTEM	WEIGHT (Lbs)	POWER (Watts)
HIGH-RESOLUTION OPTICS	2200	---
LOW-RESOLUTION OPTICS	220	---
TRANSDUCER (2)	100	150 ea.
MAGNETIC TAPE STORAGE (2)	230	200
COMMUNICATIONS	100	200
COMMAND AND TELEMETRY	50	60
COMMAND & CONTROL COMPUTER	50	50
ATTITUDE CONTROL	280	105
PROPULSION (INCLUDING FUEL)	360	---
PRIMARY POWER	300	---
STRUCTURE/THERMAL	970	---
CONTINGENCY	485	100
TOTAL	5345	---

9.6 MAJOR SUBSYSTEM PERFORMANCE SPECIFICATIONS

The major subsystems are identified in the following twelve paragraphs, with some of the more pertinent specifications listed. The first nine subsystems are integral components of the observation satellite. The tenth is the relay satellite, and the last two are ground elements. The ground elements include command and control functions as well as the primary function of data processing.

9.6.1 Optical

	High Resolution	Low Resolution
Type	3-Mirror Reflective	Refractive
Diameter	[redacted]	
Focal Length (Equivalent)		
Pointing Method	Full Aperture Mirror — Both	
Angular Field of View	1.1°	5.4°
Weight	2200 Lbs	220 Lbs

9.6.2 Transducer

Type	Quasi-Linear Phototransistor Array
Detector Element Size	16.6 x 16.6 Microns
Array Length	[redacted]
Number of Elements	
Array Scan Time	
4 Multiplexer Channels of	7.5 MHz Each

9.6.3 Magnetic Tape Storage

Input Bandwidth	30 MHz
Record Tape Speed	34.3 Inches/Second
Playback Tape Speed	8.6 Inches/Second
Playback Bandwidth	8 MHz
Head Wheel Speed: Record	16,050 rpm
Playback	4,012
Tape Start-Stop Time	0.2 Second
Tape Length	2200 Feet
Power Requirements: Record	200 Watts
Playback	80

9.6.4 Communications

Baseband Width	8 MHz
Carrier Frequency	[redacted]
Frequency Modulation Index	4
Transmitter RF Power	40 Watts
Trnnsmitter Antenna	45 db Gain, 0.9° Beam
Antenna Pointing	±0.3 Degree

9.6.5 Command and Telemetry

Modified Space-to-Ground-Link Subsystem (SGLS)

Command Decryption

Telemetry RF Bandwidth	4 MHz
Telemetry Transmitter	2 Watts
Magnetic Tape Telemetry Storage	10^8 Bits

9.6.6 Command & Control Computer

Magnetic Core Memory	8000 Words
Memory Cycle Time	2 Microseconds
Clock Accuracy (24 Hours)	1 Part in 10^7
Inputs/Outputs	150/300
External Commands	100

9.6.7 Attitude Control

[redacted]

Attitude Sensing with Inertial Rate Gyros, Horizon Sensors, Star Trackers

[redacted]

Antenna Pointing Reference ±0.5°

Uses Computer for Storage and Computation

9.6.8 Propulsion

Maintain Orbital Altitude of 315 ± 10 Km	
Launch Weight	5345 Lbs
Mission Duration	One Year
Effective Drag Area	50 Ft^2
Restarts	> 100
Liquid Bipropellant	I_{sp} = 290 Seconds
Propellant Weight	310 Lbs

9.6.9 Primary Power

Body-Mounted Silicon Solar Arrays	230 Sq. Ft.
Nickel-Cadmium Storage Batteries	~ 100 Lbs
Peak Power Demands (< 10% of time)	Sunlight - 725 Watts Shadow --- 470 "
Steady-State Demands	Sunlight - 265 Watts Shadow --- 240 "
Main Solar Array Maintained in Earth-Sun Plane Within ±15°	
Maximum Developed Power	2600 Watts

9.6.10 Relay Satellite

Weight		~ 500 Lbs
Communications:	Bandwidth	8 MHz
	Power	2 Watts
	Frequency	[redacted]
Despun	Receive Antenna	5-Foot Phased Array
Despun	Transmit Antenna	1.0-Foot Parabola
Solar Array Maximum Output		100 Watts
Stationary Orbit Over 100° West Longitude		
Command and Control through Modified SGLS		
Attitude Control		Spin plus Reaction
Velocity Control		Monopropellant Thrusters

9.6.11 Ground Terminal

Located [redacted]
Receive all data from relay satellite on [redacted]
Generate and transmit all commands.
Two 60-foot antennas.
SOA Low-Noise Receivers for 80-MHz RF bandwidth.
Correlate tracking data from all stations.

9.6.12 Ground Reconstruction

Input Rate of 8-MHz Video Baseband
21-Minute Running Time Before Reload
Sensor Sensitivity Gain Correction.
Synchronized by Burst Pilot Tone Phase-Lock
Image Shape Rectification
Output is Silver-Halide Photographic Positive

9.7 SYSTEM RELIABILITY SUMMARY

Table 9-V summarizes the reliability estimation for system A for [redacted] operation. Sufficient redundancy is included in the observation satellite design to provide an equivalent mean time to failure (MTTF) of [redacted] The subsystems that contain added redundancy are indicated by an asterisk. The methodology for estimating reliability is included in Section 17.0 of Vol. II.

9.8 SYSTEM COST ESTIMATE

Table 9-VI shows a breakdown of estimated total system cost for research and development, production, and system operation excluding government system-management and engineering cost. The estimate is based upon the parametric costing data developed in Volume II, Section 18.

TABLE 9-V SYSTEM A RELIABILITY

SYSTEM ELEMENT	ESTIMATED RELIABILITY (Includes Redundancy)
OBSERVATION SPACECRAFT:	Equiv. MTTF = [redacted]
OPTICAL	.962
TRANSDUCER	.964
MAGNETIC TAPE STORAGE	.994*
COMMUNICATIONS	.987*
COMMAND AND TELEMETRY	.989*
COMMAND & CONTROL COMPUTER	.979*
ATTITUDE CONTROL	.969
PROPULSION	.979*
PRIMARY POWER	.988*
STRUCTURAL & THERMAL CONTROL	.987
RELAY SPACECRAFT	Equiv. MTTF = [redacted]
LAUNCH VEHICLE	.90
*Added Redundancy	

TABLE 9-VI SYSTEM A COST ESTIMATE

SYSTEM ELEMENT	COST IN MILLIONS
SPACECRAFT AND AGE R&D	[redacted]
GROUND TERMINAL R&D	
LAUNCH VEHICLE R&D	
SPACECRAFT AND AGE PRODUCTION	
OBSERVATION [redacted]	
RELAY	
AGE	
LAUNCH VEHICLES	
LAUNCH SUPPORT	
GROUND TERMINAL HARDWARE	
SYSTEM OPERATION	
TOTAL	

Ground terminal costs include the microwave receiving and transmitting terminal, the command and control equipment, the ground reconstruction facility and equipment, and associated ground checkout equipment. System operation up to, and including, the generation of silver-halide positivies was estimated at [redacted] dollars per year.

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10.0 BASELINE SYSTEM B

This system consists of a single observation satellite in a near-polar circular orbit at a 525-kilometer altitude, a single ground receiving terminal located [redacted] and a ground reconstruction facility in the vicinity of the ground receiving terminal. The mission objective is to acquire and relay to the ground reconstruction facility 55 images per day at a resolution of 0.6 meters and a frame size of 6 x 6 kilometers, and 20 images per day at a resolution of 3 meters and a frame size of 30 x 30 kilometers.

The operational concept of System B is summarized as follows:

Data Acquisition — The optical transducer is a broom-scan array consisting of phototransistor elements.

Data Onboard Storage — The acquired imagery data is stored on magnetic tape.

Data Transport — The imagery data is read out of the magnetic tape on command and transmitted directly to the ground terminal.

Pointing — The optical axis is directed to the target by positioning the entire spacecraft.

Command — The onboard stored program is updated twice a day by direct transmission from the ground terminal.

Navigation — Position information is acquired periodically by tracking from ground stations. Orbit prediction is accomplished utilizing the tracking data.

A simplified system block diagram is shown in Figure 10-1. The performance specifications for each of the subsystems are presented in Section 10.6.

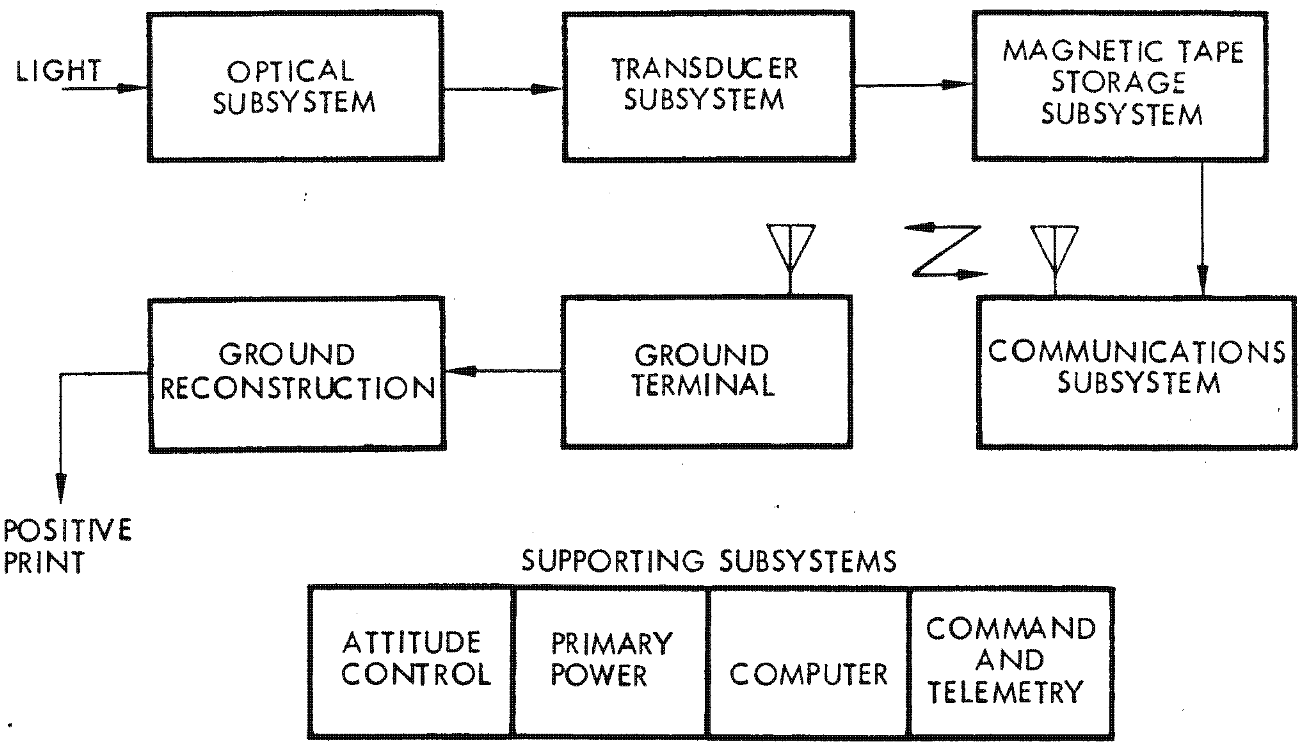

FIGURE 10-1 SIMPLIFIED SYSTEM BLOCK DIAGRAM
BASELINE SYSTEM B

10.1 SYSTEM DESIGN PARAMETERS

Data transmission time is the limiting constraint for store and forward to ground. Graph c of Figure 10-2 shows that 37.5 MHz is the minimum data transmission rate that meets the requirement of 75 images per day. The five-channel recorder suggested by this constraint also provides an adequate acquisition data rate capability, as shown in Graph b.

A reasonable system design is possible without speed change on the recorder. Baseline System B design indicated in Graph b uses a lower data rate for acquisition to minimize optical diameter. This choice leads to a large imaging time. As a result, picture interval is selected at the 15° stereo limit to provide a design margin for the pointing control subsystem (Graph a).

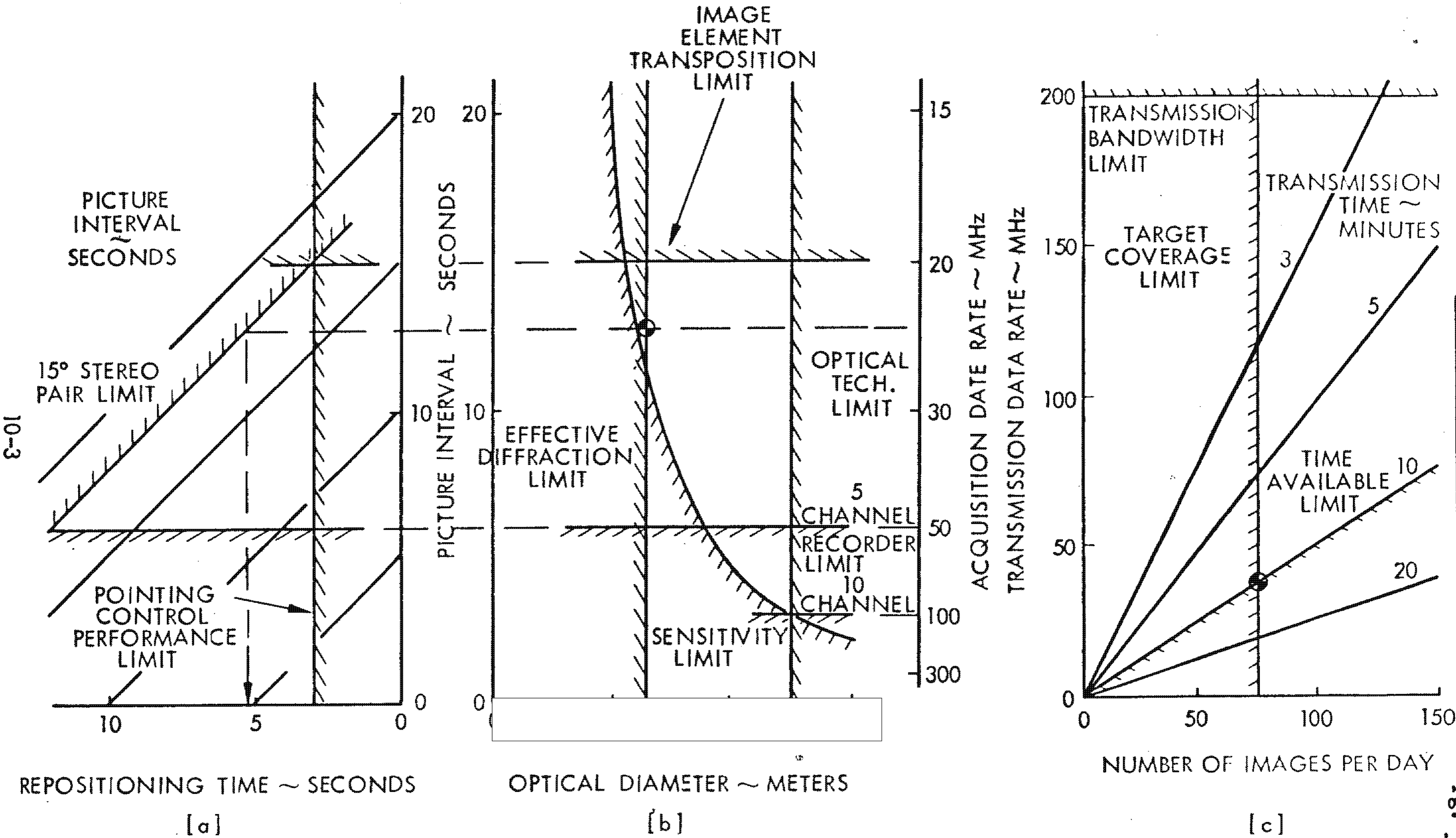

FIGURE 10-2 DESIGN TRADE CONSTRAINTS-BASELINE SYSTEM B

Use of larger optics with this system concept would allow greater margins for transducer sensitivity and target coverage performance as measured by picture interval.

Design parameters for System B are summarized in Table 10-I.

TABLE 10-I SYSTEM B DESIGN PARAMETERS

ORBIT ALTITUDE	525 Kilometers
RESOLUTION	0.6 Meter
FRAME SIZE	6 Kilometers
IMAGING TIME	12.8 Seconds
DATA ACQUISITION RATE	23.4 MHz
REPOSITIONING TIME	5.2 Seconds
PICTURE INTERVAL	18 Seconds
OPTICAL DIAMETER	[redacted]
DATA TRANSMISSION RATE	37.5 MHz

10.2 SYSTEM OPERATION

The satellite will be in view of the ground terminal for 2 or 3 consecutive passes twice a day. The minimum mutual visibility on a half-daily basis is 10 minutes with an elevation look angle from the ground terminal of 5 degrees. The system is designed to transmit 75 images to the ground terminal in 10 minutes, so that all collected imagery can be read out within approximately 12 hours.

The attitude control subsystem consists of [redacted] Attitude reference information is generated onboard the satellite from outputs of the inertial measurement unit, the star tracker, and the sun sensor. Line-of-sight orientation, image acquisition time, antenna pointing, and other commands are generated on the ground and transmitted to the satellite where the information is stored in the computer. Commands are updated on a twice-daily basis. The spacecraft maneuvers in attitude at programmed rates to provide fore-aft and crosstrack positioning of the optical axis for the optical pointing and scan rates required. The observation spacecraft maintains an Earth-oriented attitude (the optical axis pointed toward the Earth) except for several brief periods each day to re-establish inertial reference. This is accomplished by orienting the spacecraft so that its attitude sensors can acquire two fixed references such as the Sun and Canopus.

The SHF communication antenna is pointed toward the ground terminal by a combination of spacecraft maneuverability and antenna gimballing. The antenna is gimballed through a quarter of a sphere with the segment center 45° outboard from the optical axis along one side of the spacecraft. To permit hemispherical coverage the spacecraft must be maneuvered in roll about the optical axis.

10.3 ORBIT AND NAVIGATION

At an orbit altitude of 525 kilometers, daily doppler tracking by one site is sufficient for surveillance operations. At this altitude complete target coverage is assured for more than [redacted] without orbitkeeping. Based on a 1972 launch date and a ballistic coefficient of approximately 10 pounds per [redacted]

Orbit determination by a single tracking site is sufficient to ensure position prediction accuracy to one kilometer. As no orbitkeeping maneuvers that might detract from prediction accuracy are required, there should be no periods when target location uncertainties preclude image acquisition.

A summary of the orbit and navigation considerations is presented in Table 10-II.

TABLE 10-II SYSTEM B ORBIT AND NAVIGATION DATA

ORBIT: ALTITUDE INCLINATION	525 Km 96.7° (Sun Synchronous)
GROUND VISIBILITY TO STATION AT 39° LATITUDE	10 Minutes Minimum in a 12-Hour Period at 5° Elevation Angle.
COVERAGE PERIOD ABOVE 30° LATITUDE	4 to 5 Days at a View Angle of ±45°
ORBIT DECAY	Approximately 0.1 Km per Day.
POSITION ACCURACY	±1.0 Km
UPDATE ORBIT EPHEMERIS	Single Tracking Site on Daily Basis

10.4 SYSTEM RF LINKS

The imagery data stored on magnetic tape in the observation satellite is transmitted through an SHF transmission system with an effective radiated power of approximately 27 dbw in the center of the main beam. The 75 stored images can be transmitted directly to the ground terminal in 10 minutes at a baseband rate of 37.5 MHz. Data transmission is initiated by ground command following receipt of link closure confirmation.

The high-gain communications antenna provides hemispherical coverage by a combination of antenna pointing and spacecraft rolling. This results in 10 minutes of continuous transmission as the satellite passes directly over the ground terminal. The antenna beam is 6° wide so that antenna pointing within ±2° is adequate to assure quality transmission. The diameter of the half-power beam patch on the ground is only 30 nautical miles, providing a relatively high degree of privacy to the data transmission.

The command, telemetry, and tracking link between the observation satellite and the ground terminal is operated at S-band. The S-band link and associated equipment are compatible with the Air Force Space-to-Ground Link Subsystem

(SGLS). Commands are transmitted to, and telemetry data is received from, the satellite during the previously mentioned mutual visibility periods. Onboard storage is provided to accommodate telemetry data generated during non-visibility periods.

10.5 OBSERVATION SPACECRAFT CHARACTERISTICS

The spacecraft configuration is basically a cylinder 22 feet long and 8 feet in diameter. The base contains the equipment mounting deck (electronics, reaction control fuel tanks, [redacted] and other non-optical equipment) and thermal radiators. Mounted above the base is the optical sensor system which occupies approximately ten feet in length excluding the sunshade. When extended, the sunshade adds another 10 feet to the length of the vehicle.

The communication antenna and two solar paddles are stowed along the cylindrical structure during boost and are deployed to positions in the plane of the equipment mounting deck following separation from the launch vehicle. The sunshade is retracted for boost to permit the spacecraft to fit within the standard OAO shroud. The low-resolution optical sensor system is mounted parallel with, and external to, the high-resolution system.

The two solar-array paddles are rotated about a common axis and are oriented normal to the Earth-sun plane while in the sunlight. During the shadow periods the paddles are oriented so as to minimize the spacecraft drag area.

The satellite may be launched by either a Titan IIIB Agena (N_2O_4) or an Atlas SLV-3B/Agena (N_2O_4).

Table 10-III summarizes the major characteristics of the observation spacecraft.

TABLE 10-III OBSERVATION SPACECRAFT CHARACTERISTICS - SYSTEM B

LENGTH	OVERALL	22 Feet
DIAMETER	EXCLUDING SOLAR PADDLES	8 Feet
WEIGHT	INITIAL INJECTION	4850 Lbs
OPTICAL DIAMETER	HIGH RESOLUTION LOW RESOLUTION	[redacted]
OPTICAL TRANSDUCER	PHOTO TRANSISTOR ARRAY	
DATA STORAGE CAPACITY	CYCLES IMAGES	4.5×10^{10} 150
COMMUNICATIONS	INFORMATION BANDWIDTH CARRIER FREQUENCY RF POWER ANTENNA SIZE EFFECTIVE RADIATED POWER TRANSMISSION TIME/DAY	37.5 MHz [redacted] 1 Watt 1 Foot 27 dbw 10 Minutes
ATTITUDE CONTROL		[redacted]
SOLAR ARRAY	ARRAY AREA ARRAY POWER	100 Feet^2 1000 Watts

10.5.1 Weight and Power Summary

Table 10-IV lists the weight and power required by each of the major subsystems. The two optical sensors have identical transducer subsystems of approximately 50 pounds each. The magnetic tape storage total includes redundant recorders and multiplexers. Primary-power weight includes the solar paddles, power conditioning equipment, and the nickel-cadmium storage batteries.

The listed power values represent maximum watts drawn by each subsystem, but are not to be added for system totals. Simultaneous operation of all subsystems does not occur. No power is allotted for optical-subsystem thermal control.

10.6 MAJOR SUBSYSTEM PERFORMANCE SPECIFICATIONS

The major subsystems are identified in the following ten paragraphs with some of the more pertinent specifications listed. The first eight subsystems are integral components of the observation satellite and the last two are ground

TABLE 10-IV OBSERVATION SPACECRAFT WEIGHT & POWER SUMMARY - SYSTEM B

SUBSYSTEM	WEIGHT (Lbs)	POWER (Watts)
HIGH-RESOLUTION OPTICS	2100	----
LOW-RESOLUTION OPTICS	250	----
TRANSDUCER (2)	100	250 Ea.
MAGNETIC TAPE STORAGE (2)	280	250
COMMUNICATIONS	50	50
COMMAND AND TELEMETRY	50	60
COMMAND AND CONTROL COMPUTER	50	50
ATTITUDE CONTROL	350	105
PRIMARY POWER	300	----
STRUCTURE/THERMAL	880	----
CONTINGENCY	440	100
TOTAL	4850	----

elements. The ground elements include the command and control functions as well as the primary function of data processing.

10.6.1 Optical

	High Resolution	Low Resolution
Type	[redacted]	
Diameter		
Focal Length (Equivalent)		
Pointing Method	Spacecraft	Spacecraft
Angular Field of View	0.65°	3.3°
Weight	2100 Lbs.	250 Lbs

10.6.2 Transducer

Type	[redacted]
Detector Element Size	
Array Length	
Number of Elements	
Array Scan Time	
5 Multiplexer Channels of	

10.6.3 Magnetic Tape Storage

Input Bandwidth		23.4 MHz
Record Tape Speed		26.7 Inches/Second
Playback Tape Speed		42.7 Inches/Second
Playback Bandwidth		37.5 MHz
Head-Wheel Speed:	Record	10,000 rpm
	Playback	16,000 rpm
Tape Start-Stop Time		0.2 Second
Tape Length		4800 Feet
Power Requirements:	Record	180 Watts
	Playback	250 Watts

10.6.4 Communications

Baseband Width	37.5 MHz
Carrier Frequency	[redacted]
Frequency Modulation Index	1
Transmitter RF Power	1 Watt
Transmitter Antenna	29 db Gain, 6° Beam
Antenna Pointing	±2.0 Degrees

10.6.5 Command and Telemetry

Modified Space-to-Ground-Link Subsystem (SGLS)

Command Decryption

Telemetry RF Bandwidth	4 MHz
Telemetry Transmitter	0.15 Watt
Magnetic Tape Telemetry Storage	10^8 Bits

10.6.6 Command and Control Computer

Magnetic Core Memory	8000 Words
Memory Cycle Time	2 Microseconds
Clock Accuracy (24 Hours)	1 Part in 10^7
Inputs/Outputs	150/300
External Commands	100

10.6.7 Attitude Control

[redacted]

Optical Pointing Accuracy	±0.1°
Antenna Pointing Accuracy	±2.0°

Uses Computer for Storage and Computation

10.6.8 Primary Power

Paddle-Mounted Silicon Solar Arrays	100 Sq. Ft.
Nickel-Cadmium Storage Batteries	~ 100 Lbs
Peak Power Demands (~ 10% of Time)	Sunlight - 650 Watts Shadow --- 500 "
Steady State Demands	Sunlight - 250 Watts Shadow --- 250 "

Solar-Array Paddles Oriented Toward Sun Within ±15°

Maximum Developed Power	1000 Watts

10.6.9 Ground Terminal

Located [redacted]

Receive all data from relay satellite on [redacted] carrier.

Generate and transmit all commands.

Two 60-Foot Antennas

SOA Low Noise Receivers for 150-MHz RF Bandwidth

Correlate Tracking Data From All Stations

10.6.10 Ground Reconstruction

Input Data Rate	37.5 MHz
Running Time Before Reload	10 Minutes

Sensor Sensitivity Gain Correction

Synchronized by Burst Pilot Tone Phase Lock

Image Shape Rectification

Output is Silver-Halide Photographic Positive

10.7 SYSTEM RELIABILITY SUMMARY

Table 10-V summarizes the reliability estimation for System B for [redacted] operation. Sufficient redundancy is included in the observation satellite design to provide an equivalent mean time to failure (MTTF) of [redacted]

The subsystems that contain added redundancy are indicated by an asterisk. The methodology for estimating reliability is included in Section 17.0 of Volume II.

TABLE 10-V SYSTEM B RELIABILITY

SYSTEM ELEMENT	ESTIMATED RELIABILITY (Includes Redundancy)
OBSERVATION SPACECRAFT	Equiv. MTTF - [redacted]
OPTICAL	.978
TRANSDUCER	.960
MAGNETIC TAPE STORAGE	.984*
COMMUNICATIONS	.988*
COMMAND AND TELEMETRY	.986*
COMMAND & CONTROL COMPUTER	.973*
ATTITUDE CONTROL	.960
PRIMARY POWER	.990*
STRUCTURAL & THERMAL CONTROL	.983
LAUNCH VEHICLE	.90
* Added Redundancy	

10.8 SYSTEM COST ESTIMATE

Table 10-VI shows a breakdown of estimated total system cost for research and development, production, and system operation excluding government system management and engineering costs. The estimate is based upon the parametric costing data developed in Volume II, Section 18.

Ground terminal costs include the microwave receiving and transmitting terminal, the command and control equipment, the ground reconstruction facility and equipment, and associated ground checkout equipment. System operation up to, and including, the generation of silver-halide positives was estimated at [redacted] dollars per year.

TABLE 10-VI SYSTEM B COST ESTIMATE

SYSTEM ELEMENT	COST IN MILLIONS
SPACECRAFT AND AGE R&D	[redacted]
GROUND TERMINAL R&D	
LAUNCH VEHICLE R&D	
SPACECRAFT AND AGE PRODUCTION	
SPACECRAFT [redacted]	
AGE	
LAUNCH VEHICLES	
LAUNCH SUPPORT	
GROUND TERMINAL HARDWARE	
SYSTEM OPERATION	
TOTAL	

11.0 BASELINE SYSTEM C

This system consists of a single observation satellite in a near polar circular orbit at 525 kilometers altitude, a single relay satellite in a 24-hour polar orbit, a single ground receiving terminal located [redacted] and a ground reconstruction facility in the vicinity of the ground terminal. The mission objective is to acquire and relay to the ground reconstruction facility 55 images per day at a resolution of 0.9 meter and a frame size of 6 x 6 kilometers, and 20 images per day at a resolution of 4.5 meters and a frame size of 30 x 30 kilometers.

The operational concept of System C is summarized as follows:

Data Acquisition — The optical transducer is a return-beam vidicon.

Data Transport — The imagery data is read out of the vidicon and transmitted through the relay satellite to the ground terminal.

Pointing — The optical axis is directed to the target by positioning the entire spacecraft.

Command — The onboard stored program is supplemented by closed-loop command.

Navigation — Position information is acquired periodically by tracking from ground stations. Orbit prediction is accomplished utilizing the tracking data.

A simplified system block diagram is shown in Figure 11-1. The performance specifications for each of the subsystems are presented in Section 11.6.

11.1 SYSTEM DESIGN PARAMETERS

Design of this direct-readout system is dictated by the characteristics of the vidicon image transducer. As shown in Graph a of Figure 11-2, the maximum permissible exposure time must be used to achieve a sensitivity margin. Under most lighting conditions, a shorter exposure time can be used.

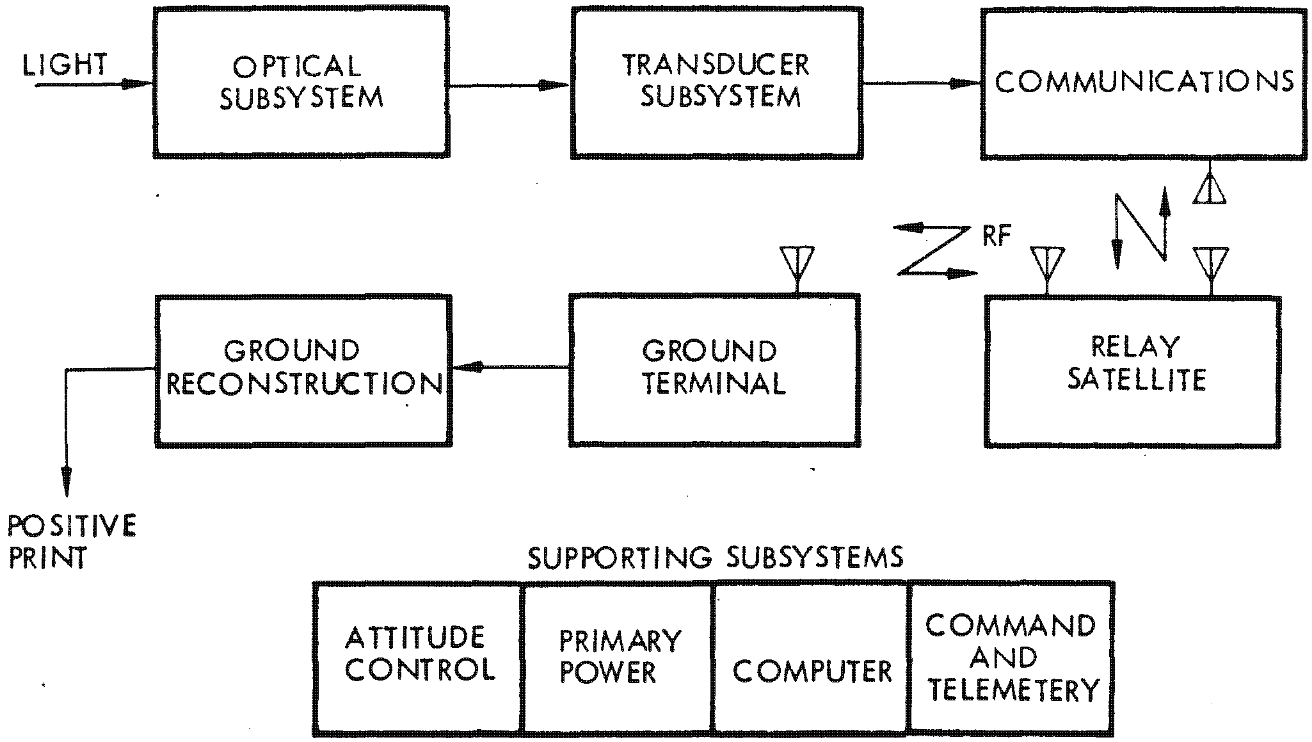

FIGURE 11-1 SIMPLIFIED SYSTEM BLOCK DIAGRAM BASELINE SYSTEM C

Because picture interval is constrained by the vidicon cycle time, more than adequate time is available for repositioning. The maximum data rate limitation imposed by the vidicon is about the same as the communication subsystem capability. The readout time selected in Figure 11-2 b results in a picture interval only half as great as the allowable value. Reducing transmission bandwidth by using a longer readout time is quite reasonable, although some penalty in vidicon performance results from longer storage time.

Design Parameters for System C are summarized in Table 11-I.

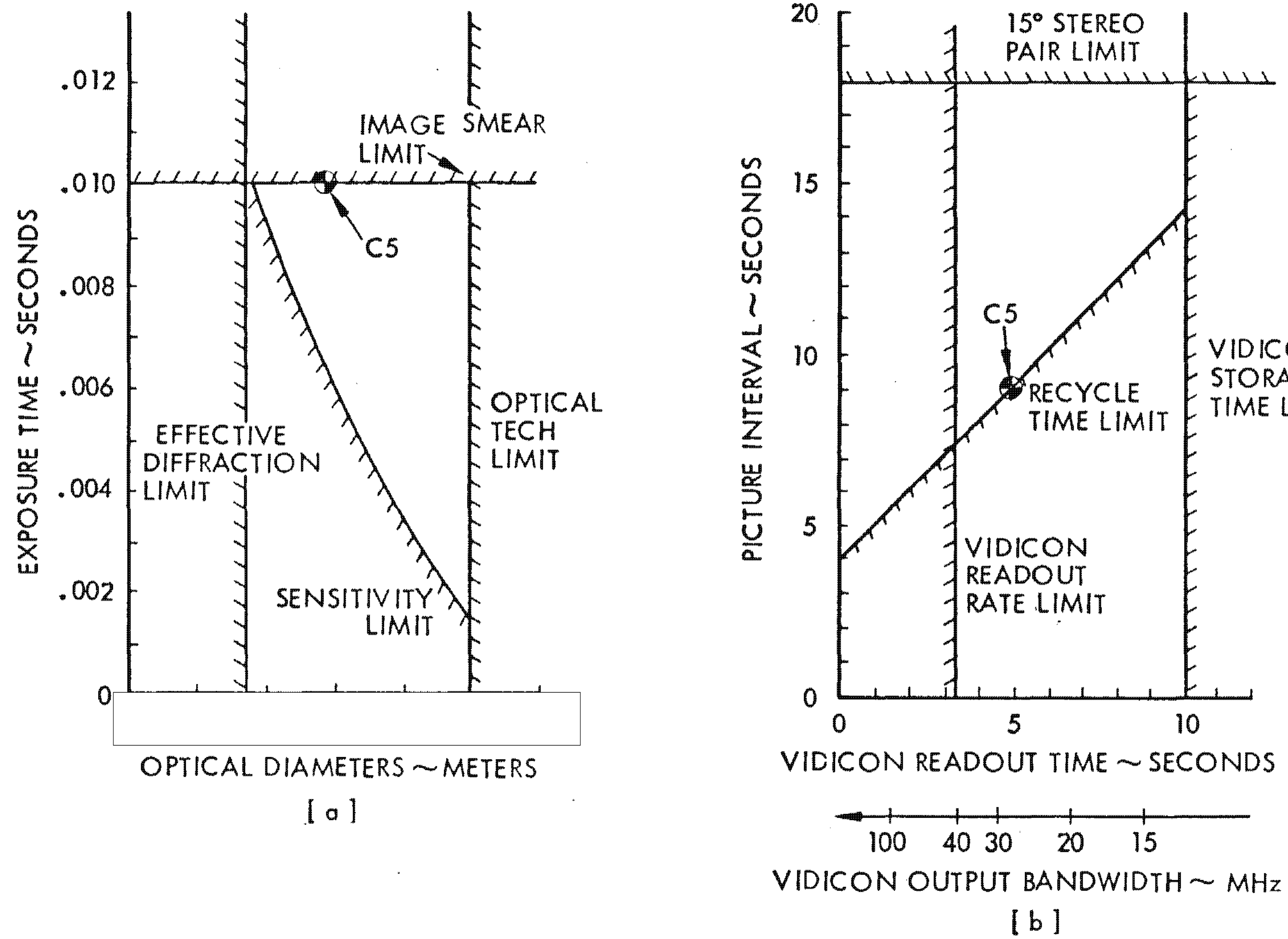

FIGURE 11-2 DESIGN TRADE CONSTRAINTS - BASELINE SYSTEM C5

ZA // TOP SECRET

TABLE 11-I SYSTEM C DESIGN PARAMETERS

ORBIT ALTITUDE	525 Kilometers
RESOLUTION	0.9 Meter
FRAME SIZE	6 Kilometers
IMAGING TIME	1 Second
REPOSITIONING TIME	8 Seconds
PICTURE INTERVAL	9 Seconds
OPTICAL DIAMETER	
DATA TRANSMISSION RATE	26.6 MHz

11.2 SYSTEM OPERATION

The system is designed to acquire and relay to ground a maximum of 75 images per day. These images are relayed to the ground as they are acquired. Imagery is taken only when mutual visibility exists between the two satellites and between the relay satellite and the ground terminal. The orbit of the relay satellite is elliptical ($\varepsilon = 0.5$) with apogee located above the North pole. As a consequence, the satellite spends most of its period over the northern hemisphere. With an orbital period of 24 solar hours the relay is always at apogee when it is local noon at 90° East Longitude. As a consequence, its ground track moves westward on the Earth surface about one degree per day. Earth oblateness causes apsidal rotation of 4.3 degrees per year. This rotation is neglected, however, as its effect is negligible. Figure 11-3 shows the pertinent system geometry.

With a sun-synchronous surveillance orbit, the local time at the satellite varies only about an hour as the satellite goes from 20° N Latitude to 70° N Latitude. This local time profile is repeated on each successive orbit. Therefore a single relay satellite in a 24-hour, polar, elliptical orbit will provide relay capability between a U.S. station and a sun-synchronous observation spacecraft while overflying all of Russia and China. The minimum visibility will be approximately 10 hours per day, even under the worst condition.

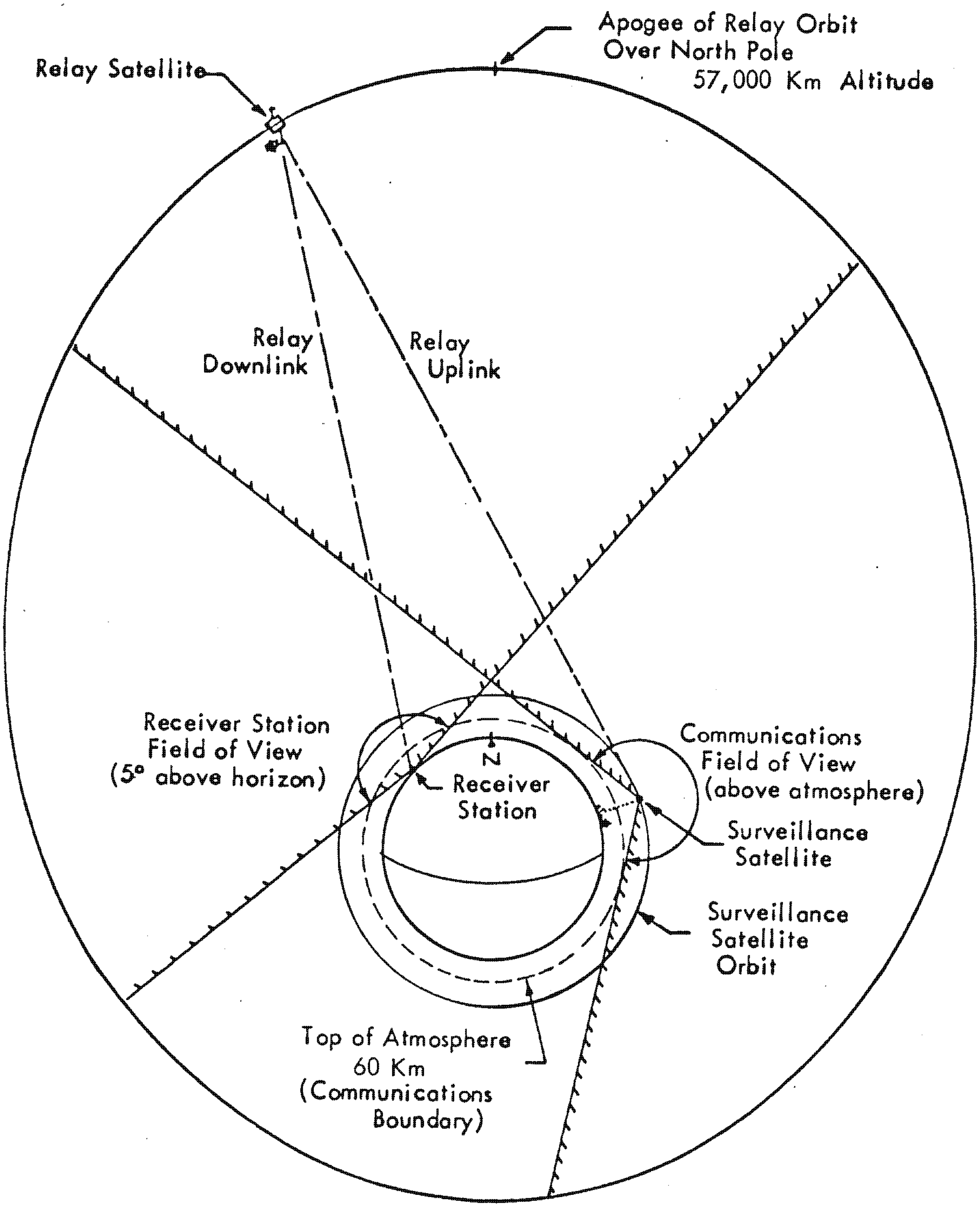

FIGURE 11-3 SATELLITE ORBIT GEOMETRY

By altering the time of arrival of the relay satellite at apogee, the locii of local noon can be shifted so that the local time of overflight can be synchronized with the observation system.

The capability for direct readout of 99% of required imagery, with coverage expanded to include the southern hemisphere, can be achieved by two approaches. If truly worldwide coverage were required, with one observation satellite, it could be accomplished with multiple relay satellite links. It could also be accomplished by recording and storing the southern hemisphere imagery data and transmitting it at a later time, either directly to ground or through the relay satellite. The latter approach is more desirable from a system complexity and cost standpoint. This would require the addition of a data recording and storage subsystem.

In this system the low-resolution optical axis is oriented 30° ahead of the high-resolution axis, permitting the low-resolution image of a particular area to be transmitted and monitored at the reconstruction facility prior to taking a high-resolution image in the same area. The time separation ranges from approximately 40 seconds (with the high-resolution optics pointed at the nadir) to approximately 90 seconds when the low-resolution optics are pointed toward the Earth's horizon.

Images can be acquired simultaneously with the two optical sensor systems, but the data is read out of the two vidicons in series. This is possible because an image can be retained on the photoconductive surface for at least 10 seconds without appreciable degradation. The system is designed so that the high-resolution transducer is read out during the initial 5-second period. The low-resolution transducer is read out during the remaining 5-second period. This series operation permits simultaneous acquisition of imagery without an increase in RF bandwidth.

[redacted] Attitude reference information is generated onboard the satellite from outputs of the inertial measurement unit, the star tracker, and the sun sensors. Antenna pointing, satellite position, and target selection information are generated on the ground, transmitted to the satellite, and stored in the command and control computer. The information is updated as necessary to maintain the proper tolerances. The spacecraft maneuvers in attitude at programmed rates to provide fore-aft and crosstrack positioning of the optical axis for target acquisition.

The observation spacecraft maintains an Earth-oriented attitude except for several brief periods each day to re-establish inertial reference. This is accomplished by orienting the spacecraft so that its attitude sensors can acquire two fixed references such as the Sun and Canopus. This attitude is maintained for several minutes after which the spacecraft is again ready to commence image acquisition.

11.3 ORBIT AND NAVIGATION

At an orbit altitude of 525 kilometers, daily doppler tracking by one site is sufficient for surveillance operations. Complete target coverage is assumed for more than one year without orbitkeeping. Based on a 1972 launch date and a ballistic coefficient of approximately 10 pounds per square foot, the 525- [redacted]

Orbit determination by a single tracking site is sufficient to ensure position prediction accuracy to one kilometer. As no orbitkeeping maneuvers that might detract from prediction accuracy are required, there should be no periods when image acquisition cannot be accomplished because of target location uncertainties.

A summary of the orbit and navigation considerations is listed in Table 11-II.

TABLE 11-II SYSTEM C ORBIT AND NAVIGATION DATA

ORBIT: ALTITUDE	525 Km
INCLINATION	96.7° (Sun Synchronous)
GROUND VISIBILITY TO STATION AT [redacted] LATITUDE	10 Minutes Minimum in a 12-Hour Period at 5° Elevation Angle.
COVERAGE PERIOD	4 to 5 Days at a View Angle of ±45° Above 30° Latitude
ORBIT DECAY	Approximately 0.1 Km per Day
POSITION ACCURACY	±1.0 Km
UPDATE ORBIT EPHEMERIS	Single Tracking Site on Daily Basis

11.4 SYSTEM RF LINKS

The imagery data is read out directly through the relay satellite as the data is acquired. The return-beam vidicon is scanned at a rate consistent with 5-second readout. At 0.9-meter resolution and a ground frame size of 6 kilometers, a communications subsystem baseband width of 26.6 MHz is required. The imagery data is transmitted through an SHF transmission system, at [redacted] with an effective radiated power of approximately 68 dbw in the center of the main beam. The 75 images are transmitted through the relay satellite to the ground terminal in 11 minutes total transmission time.

The high-gain communication antenna provides hemispherical coverage capability and is pointed toward the relay satellite using reference signals from the computer. Fine pointing is accomplished by the auto-track system which develops antenna drive signals from received transmissions of the relay satellite. When mutual visibility exists, the communication link between the two satellites is closed and data can be transmitted as acquired. The observation satellite communications transmitters will be turned on by ground command only as needed because of the high power requirements.

The command and telemetry link between the observation satellite and the ground terminal is operated in the SHF band through the relay satellite in the normal mode. An additional command and telemetry link is provided by S-band directly with the ground terminal. This link is also used for tracking and ranging. In addition, the ground terminal maintains an S-bank link with the relay satellite for command and control. A simplified representation of the system RF links is shown in Figure 9-3.

11.5 OBSERVATION SPACECRAFT CHARACTERISTICS

The spacecraft configuration is basically a cylinder 20 feet long and 9 feet in diameter. The base contains the equipment mounting deck and thermal radiators. Mounted above the base is the optical sensor system which occupies approximately eight feet of length excluding the sunshade. When extended, the sunshade adds another 10 feet to the length of the vehicle.

The communication antenna and two solar paddles are stowed along the cylindrical structure during boost and are deployed to positions in the plane of the equipment mounting deck following separation from the launch vehicle. The sunshade is retracted for boost to permit the spacecraft to fit within the standard OAO shroud. The low-resolution optical sensor system is mounted with its axis 30° from, and external to, the high-resolution system.

The two solar-array paddles are rotated about a common axis and are oriented normal to the Earth-Sun plane while in the sunlight. During the shadow periods the paddles are oriented so as to minimize the spacecraft drag area.

The satellite may be launched by either a Titan IIIB Agena (N_2O_4) or an Atlas SLV-3B/Agena (N_2O_4) with a standard OAO shroud.

Table 11-III summarizes the major characteristics of the observation spacecraft.

TABLE 11-III OBSERVATION SPACECRAFT CHARACTERISTICS - SYSTEM C

LENGTH	OVERALL	20 Feet
DIAMETER	EXCLUDING SOLAR PADDLES	9 Feet
WEIGHT	INITIAL INJECTION	5290 Lbs
OPTICAL DIAMETER	HIGH RESOLUTION LOW RESOLUTION	[redacted]
OPTICAL TRANSDUCER	RETURN-BEAM VIDICON	3 In. x 3 In.
COMMUNICATIONS	INFORMATION BANDWIDTH CARRIER FREQUENCY RF POWER ANTENNA SIZE EFFECTIVE RADIATED POWER TRANSMISSION TIME/DAY	26.6 MHz [redacted] 200 Watts 6 Feet 68 dbw 6.5 Minutes
ATTITUDE CONTROL		[redacted]
SOLAR ARRAY	ARRAY AREA ARRAY POWER	100 $Feet^2$ 1000 Watts

11.5.1 Weight and Power Summary

Table 11-IV lists the weight and power required by each of the major subsystems. The two optical sensors have identical transducer subsystems of

TABLE 11-IV OBSERVATION SPACECRAFT WEIGHT & POWER SUMMARY - SYSTEM C

SUBSYSTEM	WEIGHT (Lbs)	POWER (Watts)
HIGH-RESOLUTION OPTICS	2500	----
LOW-RESOLUTION OPTICS	220	----
TRANSDUCER (2)	240	250 ea.
COMMUNICATIONS	130	1000
COMMAND AND TELEMETRY	50	60
COMMAND AND CONTROL COMPUTER	50	50
ATTITUDE CONTROL	360	140
PRIMARY POWER	300	----
STRUCTURE/THERMAL	950	
CONTINGENCY	475	100
TOTAL	5290	----

approximately 120 pounds each. Primary power weight includes the solar paddles, power conditioning equipment, and the nickel-cadmium storage batteries.

The listed power values represent maximum watts drawn by each subsystem, but are not to be added for system totals. Simultaneous operation of all subsystems does not occur. No power is allotted for optical-subsystem thermal control.

11.6 MAJOR SUBSYSTEM PERFORMANCE SPECIFICATIONS

The major subsystems are identified in the following ten paragraphs with some of the more pertinent specifications listed. The first seven subsystems are integral components of the observation satellite, the eighth is the relay satellite, and the last two are ground elements. The ground elements include the command and control functions as well as the primary function of data processing.

11.6.1 Optical

	High Resolution	Low Resolution
Type	Corrected Cassegrain	Refractive
Diameter		
Focal Length (Equivalent)		
Pointing Method	Spacecraft	Spacecraft
Angular Field of View	0.65°	3.3°
Weight	2500 Lbs	220 Lbs

11.6.2 Transducer

Type	Return-Beam Vidicon
Resolution	90 Line Pairs per Millimeter
Format Size	3 x 3 Inches
Exposure Time	0.01 Second
Readout Time	5 Seconds

11.6.3 Communications

Baseband Width	26.6 MHz
Carrier Frequency	[redacted]
Frequency Modulation Index	2.5
Transmitter RF Power	200 Watts
Transmitter Antenna	47 db Gain, 0.8° Beam
Antenna Pointing	±0.2 Degree

11.6.4 Command and Telemetry

Modified Space-to-Ground-Link Subsystems (SGLS)

Command Decryption

Telemetry RF Bandwidth	4 MHz
Telemetry Transmitter	2 Watts
Magnetic Tape Telemetry Storage	10^8 Bits

11.6.5 Command and Control Computer

Magnetic Core Memory	8000 Words
Memory Cycle Time	2 Microseconds
Clock Accuracy (24 Hours)	1 Part in 10^7
Inputs/Outputs	150/300
External Commands	100

11.6.6 Attitude Control

[redacted]

Antenna Pointing Reference	±0.5°

Uses Command and Control Computer for Storage and Computation

11.6.7 Primary Power

Paddle-Mounted Silicon Solar Arrays	100 Sq. Ft.
Nickel-Cadmium Storage Batteries	~ 100 Lbs.
Peak Power Demands (~ 10% of Time)	Sunlight - 1750 Watts Shadow --- 249 Watts
Steady-State Demands	Sunlight - 250 Watts Shadow --- 190 Watts

Solar-Array Paddles Oriented Toward Sun within ±15°

Maximum Developed Power by Array	1000 Watts

11.6.8 Relay Satellite

Weight	~ 500 Lbs.
Communications: Bandwidth (baseband)	26.7 MHz
Power	20 Watts
Frequency	[redacted]
Despun { Receive Antenna	
Despun { Transmit Antenna	
Solar-Array Maximum Output	100 Watts
24-Hour Polar Orbit ($\varepsilon = 0.5$) Max. Range	62,000 Km

Command and Control Through Modified SGLS

Attitude Control	Spin plus Reaction Control
Velocity Control	Monopropellant Thrusters

11.6.9 Ground Terminal

Located [redacted]

Receive all data from relay satellite on 12-GHz carrier.

Generate and transmit all commands.

Two 60-Foot Antennas

SOA Low-Noise Receivers for 200-MHz RF Bandwidth

Correlate Tracking Data from All Stations

11.6.10 Ground Reconstruction

Input Data Rate	26.6 MHz
Running Time Before Reload	6.5 Minutes

Synchronized by Burst Pilot Tone Phase-Lock

Image Shape Rectification

Output is Silver-Halide Photographic Positive

11.7 SYSTEM RELIABILITY SUMMARY

Table 11-V summarizes the reliability estimation for System C for [redacted] operation. Sufficient redundancy is included in the observation satellite design to provide an equivalent mean time to failure (MTTF) of [redacted] The subsystems that contain added redundancy are indicated by an asterisk.

The methodology for estimating reliability is included in Section 17.0 of Volume II.

TABLE 11-V SYSTEM C RELIABILITY

SYSTEM ELEMENT	ESTIMATED RELIABILITY (Includes Redundancy)
OBSERVATION SPACECRAFT:	Equiv. MTTF = [redacted]
OPTICAL	.974
TRANSDUCER	.973
COMMUNICATIONS	.983*
COMMAND AND TELEMETRY	.986*
COMMAND & CONTROL COMPUTER	.973*
ATTITUDE CONTROL	.960
PRIMARY POWER	.990*
STRUCTURAL & THERMAL CONTROL	.983
RELAY SPACECRAFT	Equiv. MTTF = [redacted]
LAUNCH VEHICLE	.90

* Added Redundancy

11.8 SYSTEM COST ESTIMATE

Table 11-VI shows a breakdown of estimated total system cost for research and development, production, and system operation excluding government system management and engineering costs. The estimate is based upon the parametric costing data developed in Volume II, Section 18.

TABLE 11-VI SYSTEM C COST ESTIMATE

SYSTEM ELEMENT	COST IN MILLIONS
SPACECRAFT AND AGE R&D	
GROUND TERMINAL R&D	
LAUNCH VEHICLE R&D	
SPACECRAFT AND AGE PRODUCTION	
SPACECRAFT	
RELAY SPACECRAFT	
AGE	
LAUNCH VEHICLES	
LAUNCH SUPPORT	
GROUND TERMINAL HARDWARE	
SYSTEM OPERATION	
TOTAL	

Ground-terminal costs include the microwave receiving and transmitting terminal, the command and control equipment, the ground reconstruction facility and equipment, and associated ground checkout equipment. System operation up to, and including, the generation of silver-halide positives was estimated at dollars per year.

12.0 BASELINE SYSTEM D

This system consists of a single observation satellite in a near-polar circular orbit at 250 kilometers altitude, a single relay satellite in a 24-hour polar orbit, a single ground receiving terminal located [redacted] and a ground reconstruction facility in the vicinity of the ground terminal. The mission objective is to acquire and relay to the ground reconstruction facility 55 images per day at a resolution of 0.6 meter and a frame size of 6 x 6 kilometers, and 20 images per day at a resolution of 3 meters and a frame size of 30 x 30 kilometers.

The operational concept of System D is summarized as follows:

Data Acquisition — The optical transducer is a broom-scan array consisting of phototransistor elements.

Data Transport — The imagery data is read out of the transducer directly and transmitted through the relay satellite to the ground terminal.

Pointing — The optical axis is directed to the target by positioning a full-aperture mirror in the fore-aft direction and by rolling the spacecraft.

Command — The onboard stored program is periodically updated as required through the relay satellite.

Navigation — Position information is acquired periodically by tracking from ground stations. Orbit prediction is accomplished by adding landmark observation data acquired from interpretation of imagery data.

A simplified system block diagram is shown in Figure 12-1. The performance specifications for each of the subsystems are presented in Section 12.6.

12.1 SYSTEM DESIGN PARAMETERS

At the selected altitude of 250 kilometers, the use of slowed scan with the quasilinear-array transducer is limited by two factors. Both picture interval,

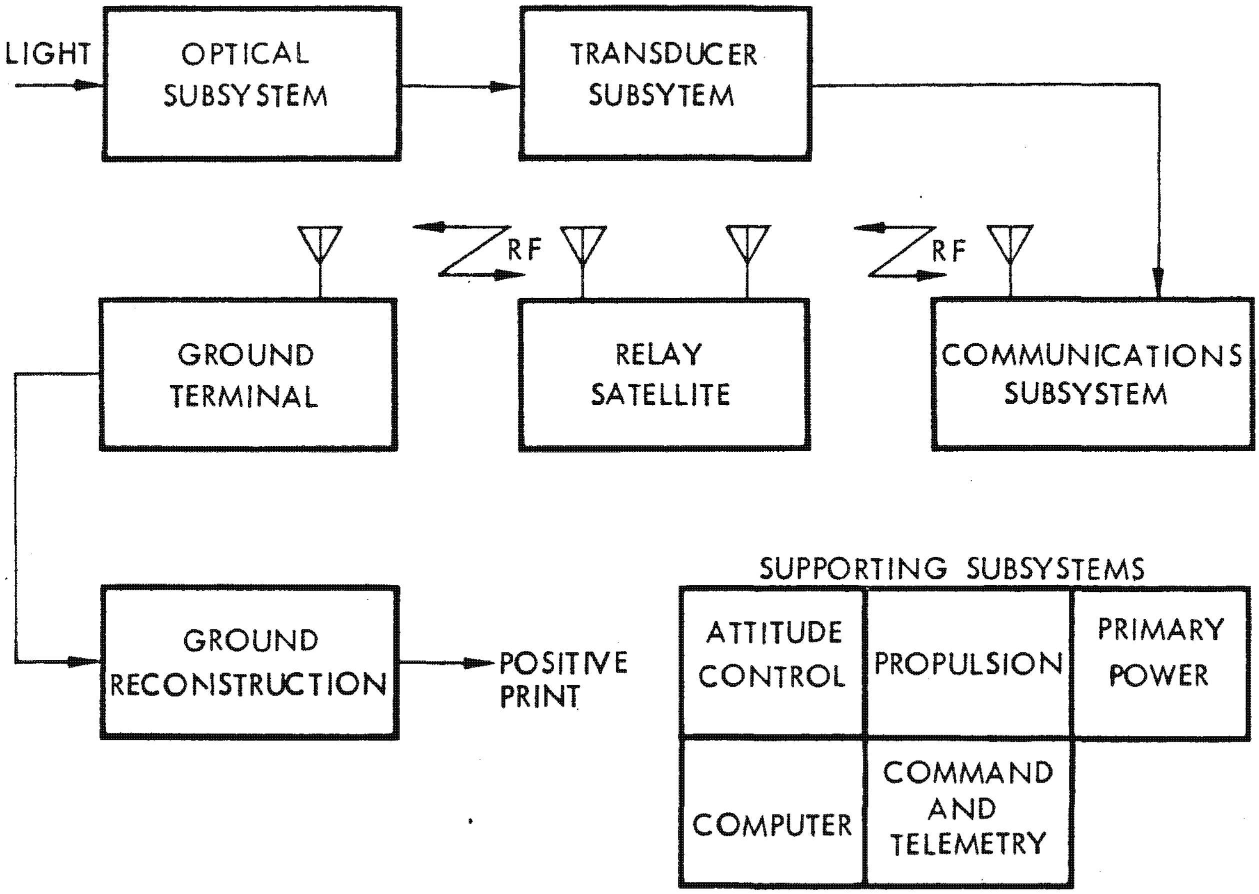

FIGURE 12-1 SIMPLIFIED SYSTEM BLOCK DIAGRAM
BASELINE SYSTEM D

as constrained by stereo-pair acquisition, and imaging time, as constrained by image element transposition, place limits on imaging time.

The System-D design point, shown in Figure 12-2, minimizes the transmission data rate. The long imaging time allows a 1-meter optical system to acquire images with a scene brightness of 400 ft-lamberts and a minus-blue filter.

This particular design leads to a picture interval too long for 15° stereo separation. A system design meeting this objective is attainable by going to a higher data transmission rate.

Design parameters for System D are summarized in Table 12-I.

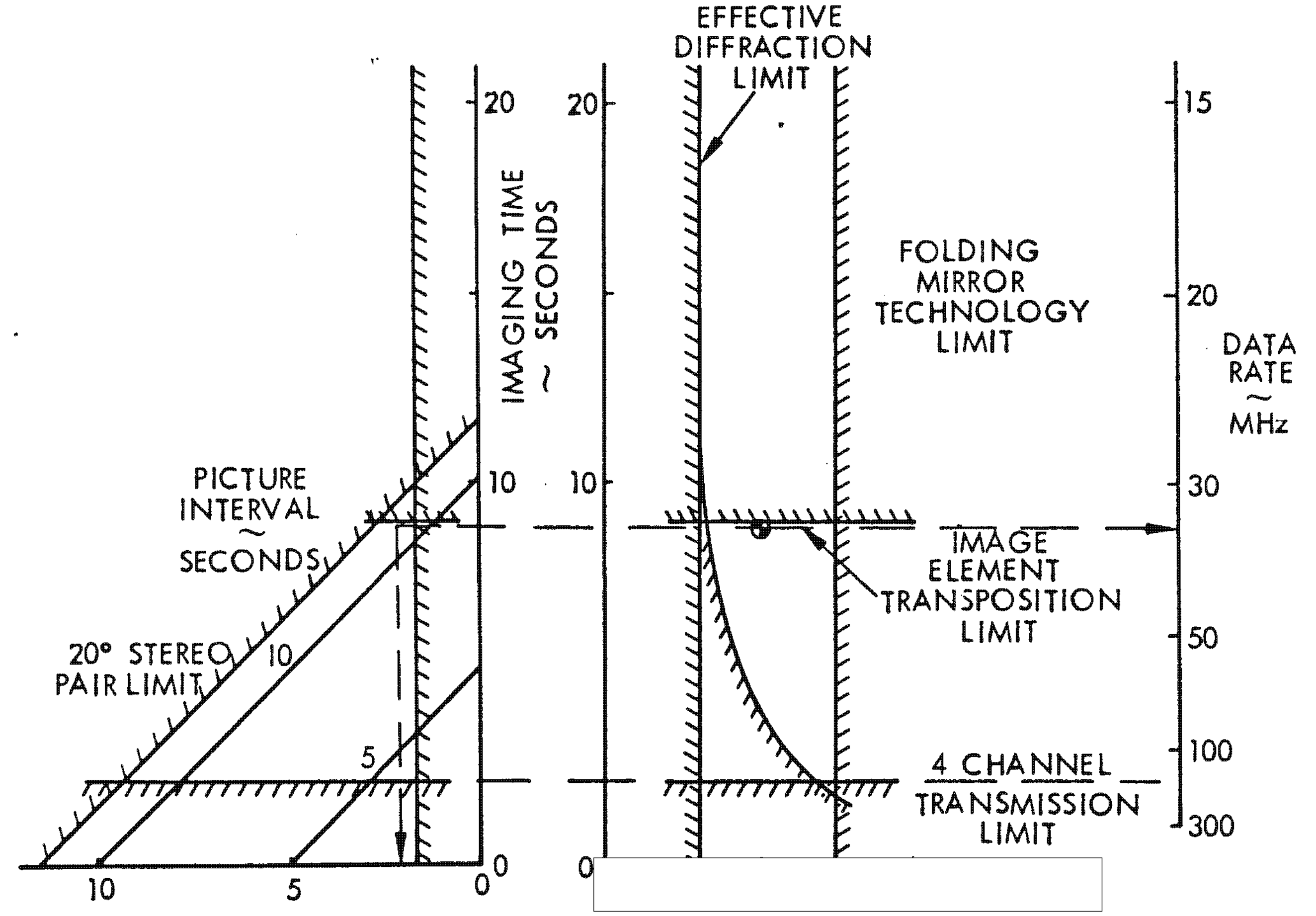

FIGURE 12-2 DESIGN TRADE CONSTRAINTS - BASELINE SYSTEM D

TABLE 12-I SYSTEM D DESIGN PARAMETERS

ORBIT ALTITUDE	250 Kilometers
RESOLUTION	0.6 Meter
FRAME SIZE	6 Kilometers
IMAGING TIME	8.9 Seconds
DATA RATE	33.7 MHz
REPOSITIONING TIME	2.1 Seconds
PICTURE INTERVAL	11.0 Seconds
OPTICAL DIAMETER	

12.2 SYSTEM OPERATION

The system is designed to acquire and relay to ground 75 images per day via relay satellite. These images are relayed to the ground as they are acquired. Imagery is taken only when mutual visibility exists between the relay satellite and the ground terminal. With the relay satellite in a highly eccentric 24-hour polar orbit, operation is restricted primarily to the northern hemisphere. Refer to Section 11.2 for applicable discussion of visibility relationships and possible methods of increasing the coverage area.

Operation of the optical and attitude control subsystems is very similar to that for System A. Refer to Section 9.2 for additional detail.

12.3 ORBIT AND NAVIGATION

To achieve a maximum coverage period of 8 days it is necessary to perform orbitkeeping every five days. To provide this coverage above 30° North Latitude the altitude must be maintained between 230 and 250 kilometers. With an average ballistic coefficient of 35 lbs/ft^2 for the observation satellite, incomplete coverage will exist if this 5-day schedule is not maintained. Velocity adjustment of approximately 30 feet per second is required for each maneuver to maintain a nominal altitude of 240 kilometers. In a [redacted] this requires 1000 pounds of fuel. The quantity of fuel carried is increased by 250 pounds to provide operational margin.

An orbit prediction interval of approximately two hours is required to hold the intrack position error within one kilometer. This accuracy can be maintained if the orbit ephemeris is updated by multiple tracking sites or if landmark observation can be accomplished on each revolution. It is quite likely that a combination of the two methods is the most effective approach. Landmark observation involves the determination of the satellite's precise location from interpretation of the collected imagery.

An orbit fix can be determined within five to six revolutions following a velocity control maneuver with accuracy to permit high resolution imagery of targets on command. A summary of the orbit and navigation considerations is listed in Table 12-II.

TABLE 12-II SYSTEM D ORBIT AND NAVIGATION DATA

ORBIT: ALTITUDE INCLINATION	250 Km 96.7 Degrees (Sun Synchronous)
GROUND VISIBILITY TO STATION AT 39° LATITUDE	4 Minutes Minimum in a 12-Hour Period at a 5° Elevation Angle
COVERAGE PERIOD ABOVE 30° LATITUDE	8 Days at a View Angle of ±45°
ORBIT DECAY	Approximately 4 Km per Day
ORBIT VELOCITY CONTROL	Every 5 Days (230 - 250 Km)
POSITION ACCURACY	±1.0 Km
UPDATE ORBIT EPHEMERIS	Multiple tracking sites, with Landmark Observation data being returned nearly every revolution.
ACCURATE ORBIT DETERMINATION	Within 5 to 6 revolutions following velocity maneuvers.

12.4 SYSTEM RF LINKS

In this system the imagery data is read out directly through the relay satellite. The quasi-linear phototransistor array is scanned for readout at a rate consistent with the slowed scanning of the target. A communications baseband width of 33.7 MHz is required for 0.6-meter ground resolution and a ground frame size of 6 kilometers. The imagery data is transmitted through an SHF transmission system, at [redacted] The effective radiated power is approximately 68 dbw in the center of the main beam. The 75 images are transmitted through the relay satellite to the ground terminal in 11 minutes total transmission time.

The communication antenna provides hemispherical coverage capability and is pointed toward the relay satellite using reference signals from the command and control computer. Fine pointing is accomplished by the auto-track system which develops antenna drive signals from transmissions received from the relay satellite. When mutual visibility exists the communications link between the two satellites is closed and data can be transmitted as acquired.

The command and telemetry link between the observation satellite and the ground terminal is operated in the SHF band through the relay satellite in the normal mode. An additional command and telemetry link is provided at S-band directly with the ground terminal. This link is also used for tracking and ranging. In addition, the ground terminal maintains an S-band link with the relay satellite for command and control. See Figure 9.3 for a simplified representation of the system RF links.

12.5 OBSERVATION SPACECRAFT CHARACTERISTICS

The spacecraft configuration is basically a cylinder 20 feet long and 6 feet in diameter. The base contains the equipment mounting deck and the thermal radiators. The optical sensor system is mounted above the base and occupies approximately 15 feet of length including the full-aperture mirror.

The communication antenna and the two solar paddles are stowed along the cylindrical structure during boost and are deployed in the plane of the equipment mounting deck following separation from the launch vehicle. Equipment mounting and solar array orientation are as discussed in Section 10.5.

The satellite can be launched by either a Titan IIIB Agena (N_2O_4) or an Atlas SLV-3B Agena (N_2O_4) with a standard OAO shroud. Table 12-III summarizes the major characteristics of the observation spacecraft.

12.5.1 Weight and Power Summary

Table 12-IV lists the weight and power required by each of the major subsystems. The optical subsystem weight includes allowances for the full-aperture mirror and associated tilting mechanisms. Comments concerning the listed power requirements can be found in Section 9.5.1.

TABLE 12-III OBSERVATION SPACECRAFT CHARACTERISTICS - SYSTEM D

LENGTH	Overall	26 Feet
	Excluding Antenna	20 Feet
DIAMETER	Excluding Solar Paddles	6 Feet
WEIGHT	Initial Injection	4595 Pounds
OPTICAL DIAMETER	High Resolution	[redacted]
	Low Resolution	
OPTICAL TRANSDUCER	Photo-Transistor Array	
COMMUNICATIONS	Information Bandwidth	33.7 MHz
	Carrier Frequency	[redacted]
	RF Power	200 Watts
	Antenna Size	6 Feet
	Effective Radiated Power	68 dbw
	Transmission Time/Day	11 Minutes
ATTITUDE CONTROL	[redacted]	
VELOCITY CONTROL	Bipropellant Thrusters	
SOLAR ARRAY	Array Area	[redacted]
	Array Power	

TABLE 12-IV OBSERVATION SPACECRAFT WEIGHT & POWER SUMMARY-SYSTEM D

SUBSYSTEM	WEIGHT (Lbs)	POWER (Watts)
HIGH-RESOLUTION OPTICS	950	---
LOW-RESOLUTION OPTICS	150	---
TRANSDUCER (2)	100	250 Ea.
COMMUNICATIONS	130	1000
COMMAND & TELEMETRY	50	60
COMMAND AND CONTROL	50	50
ATTITUDE CONTROL	270	105
PROPULSION (Including Fuel)	1350	---
PRIMARY POWER	300	---
STRUCTURE/THERMAL	830	---
CONTINGENCY	415	100
TOTAL	4595	

12.6 MAJOR SUBSYSTEM PERFORMANCE SPECIFICATIONS

The major subsystems are identified in the following eleven paragraphs with some of the more pertinent specifications listed. The first eight subsystems are integral components of the observation satellite, the ninth is the relay satellite, and the last two are ground elements. The ground elements include the command and control functions as well as the primary function of data processing.

12.6.1 Optical

	High Resolution	Low Resolution
Type	3-Mirror Reflective	Refractive
Diameter	[redacted]	
Focal Length (Equivalent)		
Pointing Method	Mirror & Roll	Mirror & Roll
Angular Field of View	1.4°	6.9°
Weight	950 Pounds	150 Pounds

12.6.2 Transducer

Type	[redacted]
Detector Element Size	
Array Length	
Number of Elements	
Array Scan Time	
Four Multiplexer Channels of	8.5 MHz Each

12.6.3 Communications

Baseband Width	33.7 MHz
Carrier Frequency	[redacted]
Frequency Modulation Index	4
Transmitter RF Power (Total)	200 Watts
Transmitter Antenna	47 db Gain, 8° Beam
Antenna Pointing	±0.2 degree

12.6.4 Command and Telemetry

Modified Space-to-Ground-Link Subsystem (SGLS)

Command Decryption

Telemetry RF Bandwidth	4 MHz
Telemetry Transmitter	2 Watts
Magnetic Tape Telemetry Storage	10^8 Bits

12.6.5 Command and Control Computer

Magnetic Core Memory	8000 Words
Memory Cycle Time	2 Microseconds
Clock Accuracy (24 Hours)	1 part in 10^7
Inputs/Outputs	150/300
External Commands	100

12.6.6 Attitude Control

Antenna Pointing Reference	±0.5°

Uses Computer for Storage and Computation

12.6.7 Propulsion

Maintain Orbital Altitude of 240 ± 10 Km

Launch Weight	4595 Pounds
Mission Duration	One Year
Effective Drag Area	50 Ft^2
At Least 100 Restarts	
Liquid Bipropellant	I_{sp} = 290 Seconds
Propellant Weight	1250 Pounds

12.6.8 Primary Power

Paddle-Mounted Silicon Solar Arrays 100 Sq. Ft.

Nickel-Cadmium Storage Batteries 100 Pounds

Peak Power Demands (10% of Time) Sunlight - 1715 Watts
Shadow - 240 Watts

Steady State Demands Sunlight - 215 Watts
Shadow - 190 Watts

Solar Array Maintained in Earth-Sun plane withing ±15°

Maximum Developed Power by Array 1000 Watts

12.6.9 Relay Satellite

Weight 50 Pounds

Communications: Bandwidth (baseband) 34 MHz
Power [redacted]
Frequency
Despun { Receive Antenna
Transmit Antenna

Solar-Array Maximum Output 100 Watts

24-Hour Polar Orbit (ε =0.5) Maximum Range 62,000 Km

Command and Control Through Modified SGLS

Attitude Control Spin plus Reaction

Velocity Control Monopropellant Thrusters

12.6.10 Ground Terminal

Located [redacted]

Receive all data from relay satellite on [redacted] carrier

Generate and transmit all commands.

Two 60-foot antennas.

SOA Low-Noise Receivers for 80-MHz RF Bandwidth each of 4 Channels

Correlate Tracking Data from all stations.

12.6.11 Ground Reconstruction

Input Rate of 8-MHz Video Baseband for each of 4 Channels

Eleven-Minute Running Time before Reload

Sensor Sensitivity Gain Correction

Synchronized by Burst Pilot Tone Phase-Lock

Image Shape Rectification

Output is Silver-Halide Photographic Positive

12.7 SYSTEM RELIABILITY SUMMARY

Table 12-V summarizes the reliability estimation for System D for [redacted] operation. Sufficient redundancy is included in the observation satellite design to provide an equivalent mean time to failure (MTTF) of [redacted] The subsystems that contain added redundancy are indicated by an asterisk. The methodology for estimating reliability is included in Section 17.0 of Volume II.

TABLE 12-V SYSTEM D RELIABILITY

SYSTEM ELEMENT	ESTIMATED RELIABILITY (Includes Redundancy)
OBSERVATION SPACECRAFT:	Equiv. MTTF = [redacted]
OPTICAL	.980
TRANSDUCER	.968
COMMUNICATIONS	.983*
COMMAND AND TELEMETRY	.986*
COMMAND & CONTROL COMPUTER	.973*
ATTITUDE CONTROL	.960
PROPULSION	.982*
PRIMARY POWER	.990*
STRUCTURAL & THERMAL CONTROL	.983
RELAY SPACECRAFT	Equiv. MTTF = [redacted]
LAUNCH VEHICLE	.90
* Added Redundancy	

12.8 SYSTEM COST ESTIMATE

Table 12-VI shows a breakdown of estimated total system cost for research and development, production, and system operation excluding government system management and engineering costs. The estimate is based upon the parametric costing data developed in Volume II, Section 18.

Ground terminal costs include the microwave receiving and transmitting terminal, the command and control equipment, the ground reconstruction facility and equipment, and associated ground checkout equipment. System operation up to, and including, the generation of silver-halide positives was estimated at [redacted] dollars per year.

TABLE 12-VI SYSTEM D COST ESTIMATE

SYSTEM ELEMENT	COST IN MILLIONS
SPACECRAFT AND AGE R&D	[redacted]
GROUND TERMINAL R&D	
LAUNCH VEHICLE R&D	
SPACECRAFT AND AGE PRODUCTION	
OBSERVATION: [redacted]	
RELAY:	
AGE:	
LAUNCH VEHICLES	
LAUNCH SUPPORT	
GROUND TERMINAL HARDWARE	
SYSTEM OPERATION	
TOTAL	

13.0 BASELINE SYSTEM E

System E consists of a single observation satellite in a near-polar circular orbit at a 315-kilometer altitude, a single relay satellite in a stationary orbit, a single ground receiving terminal located [redacted] and a ground reconstruction facility in the vicinity of the ground terminal. The mission objective is to acquire and relay to the ground reconstruction facility 55 images per day at a resolution of 0.6 meter and a frame size of 6 x 6 kilometers, and 20 images per day at a resolution of 3 meters and frame size of 30 x 30 kilometers.

The operational concept of System E is summarized as follows:

Data Acquisition — The optical transducer is a CBS Reconotron IV.

Data Onboard Storage — The acquired imagery data is stored in the transducer.

Data Transport — The imagery data is read out of the transducer and transmitted on command through the relay satellite to the ground terminal.

Pointing — The optical axis is directed to the target by positioning the entire spacecraft.

Command — The onboard stored program can be periodically updated as required through the relay satellite.

Navigation — Position information is acquired periodically by tracking from ground stations. Orbit prediction is accomplished by obtaining observation satellite ranging signals through the relay satellite in addition to direct ranging.

A simplified system block diagram is shown in Figure 13-1. The performance specifications for each of the subsystems are presented in Section 13.6.

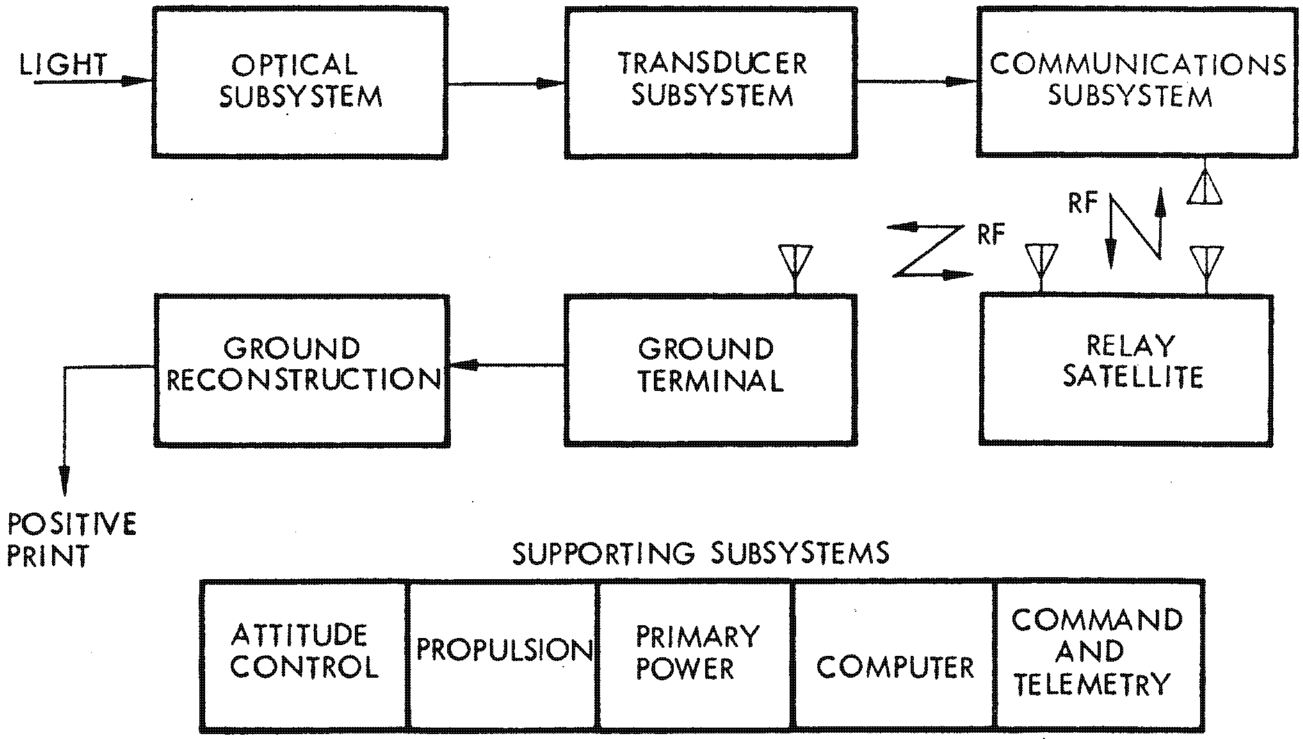

FIGURE 13-1 SIMPLIFIED SYSTEM BLOCK DIAGRAM
BASELINE SYSTEM E

13.1 SYSTEM DESIGN PARAMETERS

The design trade constraint diagrams in Figure 13-2 show the wide latitude in parameter selection for this concept. Graph a shows that an optical diameter at or near the effective diffraction limit is satisfactory. The repositioning time selected in Graph b provides margins for pointing control performance and picture interval.

The store-and-forward-via-relay data transport mode provides much greater capacity than the daily requirement indicated in Graph c. Because the images stored in the Reconotron can be read out on each orbit, the system capacity is limited by acquisition capability. While the value selected in Graph b is more than adequate for the surveillance mission, reducing picture interval by high-performance pointing control would lead to a very high capacity system.

Design parameters for System E are summarized in Table 13-I.

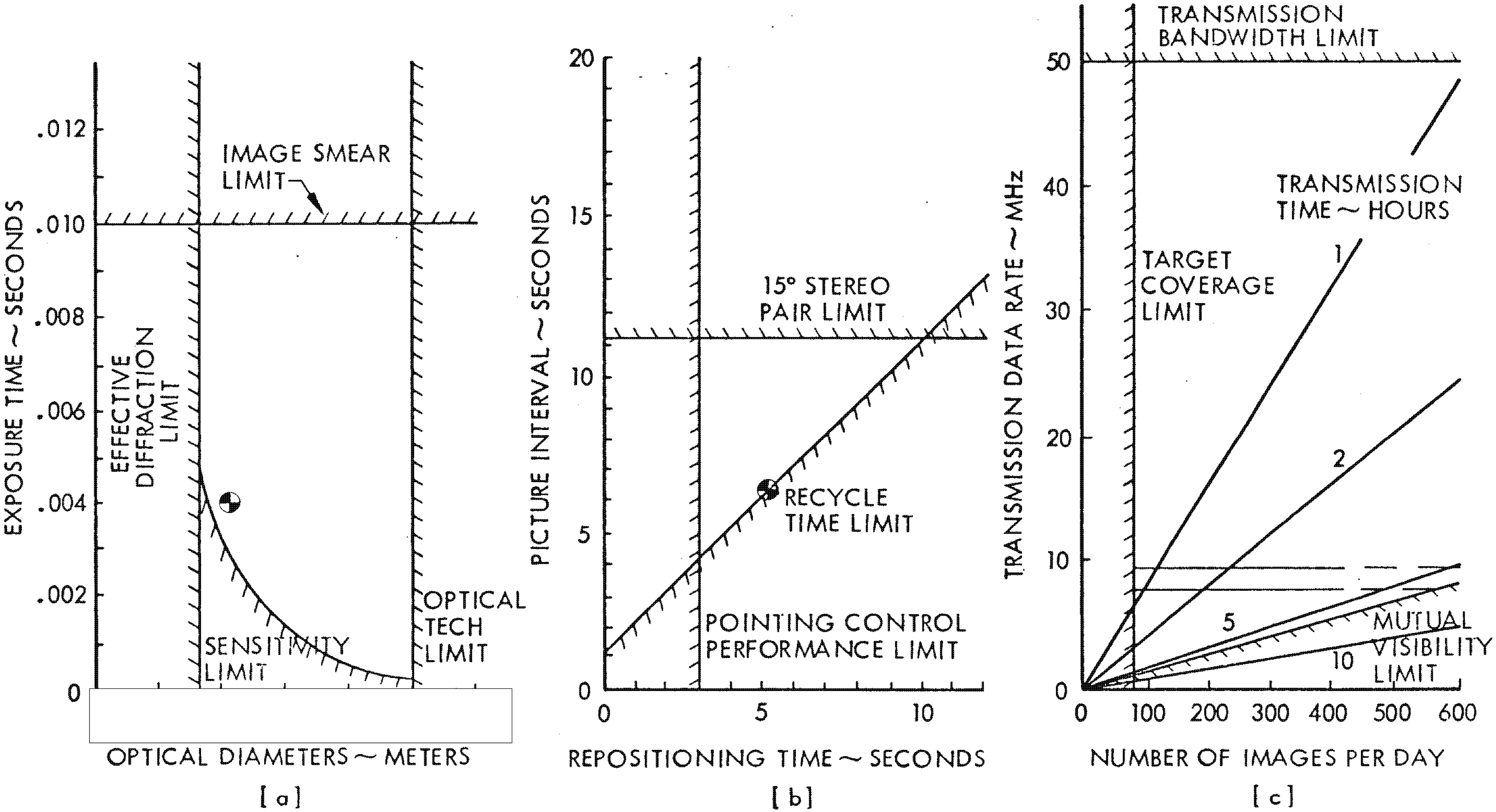

FIGURE 13-2 DESIGN TRADE CONSTRAINTS - BASELINE SYSTEM E2

TABLE 13-I SYSTEM E DESIGN PARAMETERS

ORBIT ALTITUDE	315 Kilometers
RESOLUTION	0.6 Meter
FRAME SIZE	6 Kilometers
IMAGING TIME	1 Second
DATA TRANSMISSION RATE	8 MHz
REPOSITIONING TIME	5.4 Seconds
PICTURE INTERVAL	6.4 Seconds
OPTICAL DIAMETER	[redacted]

13.2 SYSTEM OPERATION

The system is designed to acquire and relay to ground, via relay satellite, 75 images per day. These images are relayed to the ground within a few hours of acquisition. In most instances the data can be read out within one hour. The normal mode of operation allows data transmission on practically every orbit, as the relay satellite, in its orbit at 100° West Longitude, is continuously visible to the ground station. The observation satellite is visible to the relay satellite approximately one-half of each revolution.

The observation satellite maintains a streamlined attitude to reduce the effects of atmospheric drag, except during the periods when image acquisition is taking place. Image acquisition will require the spacecraft to be oriented with its longitudinal axis (optical axis) pointing towards Earth only about one hour each day. Attitude repositioning and control is provided by [redacted]

13.3 ORBIT AND NAVIGATION

To maintain the observation satellite orbit altitude between 310 and 325 kilometers for one year requires 350 pounds of propellant. The spacecraft fuel capacity has been set at 450 pounds to provide an adequate margin.

Orbit position accuracy of one kilometer is required. The variable is in-track time with one kilometer representing about .13 second. This accuracy can be maintained if the orbit ephemeris is updated by a single tracking site ranging directly through the observation satellite and by obtaining ranging information from the observation satellite through the relay satellite.

An accurate orbit fix can be calculated within nine hours of a velocity control maneuver. This would permit the resumption of acquisition of high-resolution imagery of targets on command.

A summary of the orbit and navigation considerations is presented in Table 13-II.

TABLE 13-II SYSTEM E ORBIT AND NAVIGATION DATA

ORBIT: ALTITUDE INCLINATION	315 Km 96.7 Degrees (Sun Synchronous)
GROUND VISIBILITY TO STATION AT 39° LATITUDE	5 Minutes Minimum in a 12-Hour Period a 5° Elevation Angle
COVERAGE PERIOD ABOVE 30° LATITUDE	7 Days at a View Angle of ±45°
ORBIT DECAY	Approximately 1 Km per Day
ORBIT VELOCITY CONTROL	Every 15 Days (310 - 325 Km)
POSITION ACCURACY	±1.0 Km
UPDATE ORBIT EPHEMERIS	Single tracking site, with additional ranging through relay satellite.
ACCURATE ORBIT DETERMINATION	Within 5 to 6 revolutions following velocity maneuvers.

13.4 SYSTEM RF LINKS

The data stored in the electrostatic storage of the Reconotron in the observation satellite is transmitted through an SHF transmission system with an effective radiated power of approximately 57 dbw in the center of the main beam. The 75 stored images can be transmitted through the relay satellite to the ground terminal in 47 minutes at a baseband rate of 8 MHz. Data transmission is initiated on ground command through the relay satellite. A backup mode of data transmission is available in case of failure of the primary link. This alternate method utilizes the command and tracking S-band link directly with the ground. Mutual visibility between the observation satellite and the ground terminal is limited to about 12 minutes per day. Therefore, only a portion of the daily storage capacity (20 images) can be transmitted to the ground each day in this secondary mode of operation. However, the desired imagery can be read out at a later time because the Reconotron has a long-time storage capability.

The primary command and telemetry link between the observation satellite and the ground terminal is operated in the SHF-band through the relay satellite. An additional command and telemetry link is provided at S-band directly with the ground terminal. This link is also used for tracking and ranging. In addition, the ground terminal maintains an S-band link with the relay satellite for command and control. See Figure 9-3 for a simplified representation of the system RF links.

13.5 OBSERVATION SPACECRAFT CHARACTERISTICS

The spacecraft configuration is basically a cylinder 16 feet long and 6 feet in diameter. The base contains the equipment mounting deck and the terminal radiators. The optical sensor system is mounted above the base and occupies approximately 6 feet of length excluding the sunshade. When extended, the sunshade adds another 8 feet to the length of the vehicle.

The communication antenna and the two solar paddles are stowed along the cylindrical structure during boost and are deployed in the plane of the equipment mounting deck following separation from the launch vehicle. Equipment mounting and solar-array orientation are as discussed in Section 10.5.

The satellite may be launched by either a Titan IIIB Agena (N_2O_4) or an Atlas SLV-3B Agena (N_2O_4) with a standard OAO shroud.

Table 13-III summarizes the major characteristics of the observation spacecraft.

TABLE 13-III OBSERVATION SPACECRAFT CHARACTERISTICS-SYSTEM E

LENGTH	Overall Excluding Antenna	23 16	Feet Feet
DIAMETER (Maximum Lateral Dimension)		6	Feet
WEIGHT (Initial Injection)		4025	Pounds
OPTICAL DIAMETER	High Resolution Low Resolution		
OPTICAL TRANSDUCER	CBS Reconotron IV		
DATA STORAGE CAPACITY	Cycles Images	4.5×10^{10} 150	
COMMUNICATIONS	Information Bandwidth Carrier Frequency RF Power Antenna Size Effective Radiated Power Transmission Time/Day	8 40 5 59 47	MHz Watts Feet dbw Minutes
ATTITUDE CONTROL			
VELOCITY CONTROL		Bipropellant Thrusters	
SOLAR ARRAY	Array Area Array Power		

13.5.1 Weight and Power Summary

Table 13-IV lists the weight and power required by each of the major subsystems. The two optical sensor systems have identical transducer subsystems of approximately 175 pounds each. Primary power weight includes the solar paddles, power conditioning equipment, and the nickel-cadmium storage batteries.

The listed power values represent maximum watts drawn by each subsystem, but are not to be added for system totals. Simultaneous operation of all subsystems does not occur. No power is allotted for optical-subsystem thermal control.

TABLE 13-IV OBSERVATION SPACECRAFT WEIGHT AND POWER SUMMARY SYSTEM-SYSTEM E

SUBSYSTEM	WEIGHT (Lbs)	POWER (Watts)
HIGH RESOLUTION OPTICS	1150	---
LOW RESOLUTION OPTICS	150	---
TRANSDUCER (2)	350	200
COMMUNICATIONS	100	200
COMMAND AND TELEMETRY	50	60
COMMAND AND CONTROL COMPUTER	50	50
ATTITUDE CONTROL	280	105
PROPULSION (INCLUDING FUEL)	500	---
PRIMARY POWER	300	---
STRUCTURE/THERMAL	730	---
CONTINGENCY	365	100
TOTAL	4025	---

13.6 MAJOR SUBSYSTEM PERFORMANCE SPECIFICATIONS

The major subsystems are identified in the following eleven paragraphs with some of the more pertinent specifications listed. The first eight subsystems are integral components of the observation satellite, the ninth is the relay satellite, and the last two are ground elements. The ground elements include the command and control functions as well as the primary functions of data processing.

13.6.1 Optical	High Resolution	Low Resolution
Type	Corrected Cassegrain	Refractive
Diameter		
Focal Length		
Pointing Method	Spacecraft	Spacecraft
Angular Field of View	1.1°	5.4°
Weight	1150 Pounds	150 Pounds

13.6.2 Transducer

Type	CBS Reconotron IV
Resolution	100 Line Pairs Per Millimeter
Dielectric Tape Width	4 Inches
Exposure Time	4 Milliseconds

13.6.3 Communications

Baseband Width	8 MHz
Carrier Frequency	[redacted]
Frequency Modulation Index	4
Transmitter RF Power	40 Watts
Transmitter Antenna	45 db Gain, 0.9° Beam
Antenna Pointing	±0.2 Degree

13.6.4 Command and Telemetry

Modified Space-to-Ground-Link Subsystem (SGLS)

Command Decryption

Telemetry RF Bandwidth	4 MHz
Telemetry Transmitter	2 Watts
Magnetic Tape Telemetry Storage	10^8 Bits

13.6.5 Command and Control Computer

Magnetic Core Memory	8000 Words
Memory Cycle Time	2 Microseconds
Clock Accuracy (24 Hours)	1 part in 10^7
Inputs/Outputs	150/300
External Commands	100

13.6.6 Attitude Control

[redacted]

Antenna Pointing Reference	±0.5°

Uses Command and Control Computer for Storage and Computation

13.6.7 Propulsion

Maintain Orbital Altitude of	315 ± 10 Km
Launch Weight	4025 Pounds
Mission Duration	One Year
Effective Drag Area	68 $Ft.^2$
At Least 100 Restarts	
Liquid Bipropellant	I_{sp} = 290 Seconds
Propellant Weight	450 Pounds

13.6.8 Primary Power

Paddle-Mounted Silicon Solar Arrays	100 Sq. Ft.
Nickel-Cadmium Storage Batteries	100 Pounds
Peak Power Demands (~ 10% of Time)	Sunlight - 815 Watts Shadow - 590 Watts
Steady-State Demands	Sunlight - 265 Watts Shadow - 240 Watts

Solar-Array Paddles oriented toward Sun within ±15°

Maximum Power Developed by Array	1000 Watts

13.6.9 Relay Satellite

Weight	500 Pounds
Communications: Bandwidth	8 MHz
Power	2 Watts
Frequency	
Despun { Receive Antenna	5-Foot Phased Array
Despun { Transmit Antenna	1-Foot Parabola
Solar Array Maximum Output	100 Watts
Stationary Orbit over 100° West Longitude	
Command and Control through Modified SGLS	
Attitude Control	Spin plus Reaction Control
Velocity Control	Monopropellant Thrusters

13.6.10 Ground Terminal

Located near [redacted]

Receive all data from relay satellite on [redacted] carrier.

Generate and transmit all commands.

Two 60-foot antennas.

SOA Low-Noise Receivers for 80-MHz RF Bandwidth

Correlate Tracking Data from all Stations

13.6.11 Ground Reconstruction

Input Data Rate	8 MHz
Running Time Before Reload	47 Minutes

Synchronized by Burst Pilot Tone Phase Lock

Image Shape Rectification

Output is Silver-Halide Photographic Positive

13.7 SYSTEM RELIABILITY SUMMARY

Table 13-V summarizes the reliability estimation for System E for [redacted] operation. Sufficient redundancy is included in the observation satellite design to provide an equivalent mean time to failure (MTTF) of [redacted] The subsystems that contain added redundancy are indicated by an asterisk. The methodology for estimating reliability is included in Section 17.0 of Volume II.

13.8 SYSTEM COST ESTIMATE

Table 13-VI shows a breakdown of estimated total system cost for research and development, production, and system operation excluding government system management and engineering costs. The estimate is based upon the parametric costing data developed in Volume II, Section 18.

Ground terminal costs include the microwave receiving and transmitting terminal, the command and control equipment, the ground reconstruction facility and equipment, and associated ground checkout equipment. System operation up to, and including, the generation of silver-halide positives was estimated at [redacted] dollars per year.

TABLE 13-V SYSTEM E RELIABILITY

SYSTEM ELEMENT	ESTIMATED RELIABILITY (Includes Redundancy)
OBSERVATION SPACECRAFT	Equivalent MTTF = [redacted]
OPTICAL	.985
TRANSDUCER	.983
COMMUNICATIONS	.987*
COMMAND AND TELEMETRY	.990*
COMMAND AND CONTROL COMPUTER	.980*
ATTITUDE CONTROL	.975
PROPULSION	.982*
PRIMARY POWER	.990*
STRUCTURAL AND THERMAL CONTROL	.987
RELAY SPACECRAFT	Equivalent MTTF = [redacted]
LAUNCH VEHICLE	.90
*Added Redundancy	

TABLE 13-VI SYSTEM E COST ESTIMATE

SYSTEM ELEMENT		COST IN MILLIONS
SPACECRAFT AND AGE R&D		[redacted]
GROUND TERMINAL R&D		
LAUNCH VEHICLE R&D		
SPACECRAFT AND AGE PRODUCTION		
SPACECRAFT	[redacted]	
RELAY SPACECRAFT		
AGE		
LAUNCH VEHICLES		
LAUNCH SUPPORT		
GROUND TERMINAL HARDWARE		
SYSTEM OPERATION		
TOTAL		

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14.0 BASELINE SYSTEM F

This system consists of a single observation satellite in a near-polar circular orbit at a 525-kilometer altitude, a single ground receiving terminal located [redacted] and a ground reconstruction facility in the vicinity of the ground receiving terminal. The mission objective is to acquire and relay to the ground reconstruction facility 55 images per day at a resolution of 0.6 meters and a frame size of 6 x 6 kilometers, and 20 images per day at a resolution of 3 meters and a frame size of 30 x 30 kilometers.

The operational concept of System F is summarized as follows:

<u>Data Acquisition</u> — The optical transducer is screened thermoplastic film.

<u>Data Onboard Storage</u> — The acquired imagery data is stored on the individual thermoplastic plates.

<u>Data Transport</u> — The imagery data is read out of the storage on command and transmitted directly to the ground terminal.

<u>Pointing</u> — The optical axis is directed to the target by positioning the entire spacecraft.

<u>Command</u> — The onboard stored program is updated twice a day by direct transmission from the ground terminal.

<u>Navigation</u> — Position information is acquired periodically by tracking from ground stations. Orbit prediction is accomplished utilizing the tracking data.

A simplified system block diagram is shown in Figure 14-1. The performance specifications for each of the subsystems are presented in Section 14.6.

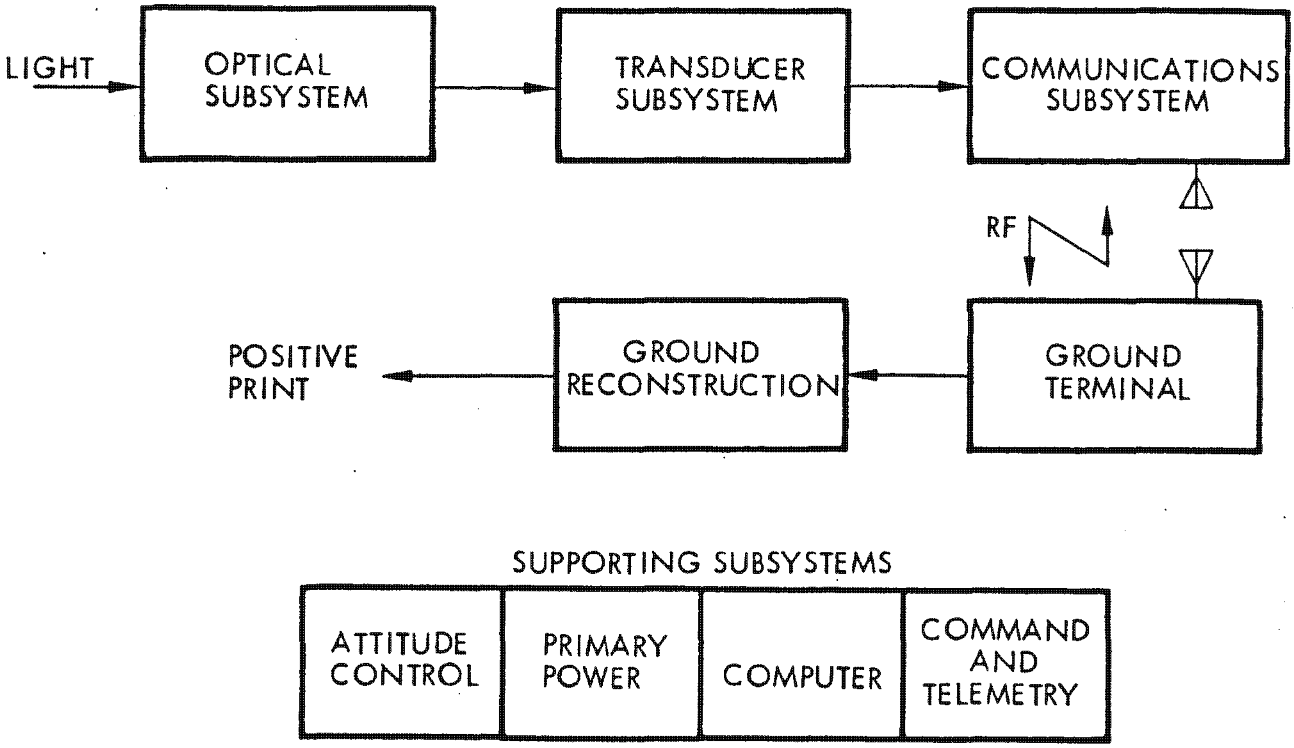

FIGURE 14-1 SIMPLIFIED SYSTEM BLOCK DIAGRAM
BASELINE SYSTEM F

14.1 SYSTEM DESIGN PARAMETERS

Data transmission time is the limiting constraint for store and forward to ground. Graph c of Figure 14-2 shows that 37.5 MHz is the minimum data transmission rate that meets the requirement of 75 images in 12 hours. Although this value is used in System F, substantial growth is available by increasing transmission data rate.

The very high sensitivity of the thermoplastic transducer makes it possible to use minimum size optics. The very short exposure time, indicated in Graph a, results in relatively large tolerance to pointing angle rate errors.

The large picture interval allowable at the 525-Km altitude permits a long repositioning time. The design value chosen in Graph b provides a much greater acquisition rate than required, as well as allowing a performance margin for pointing control.

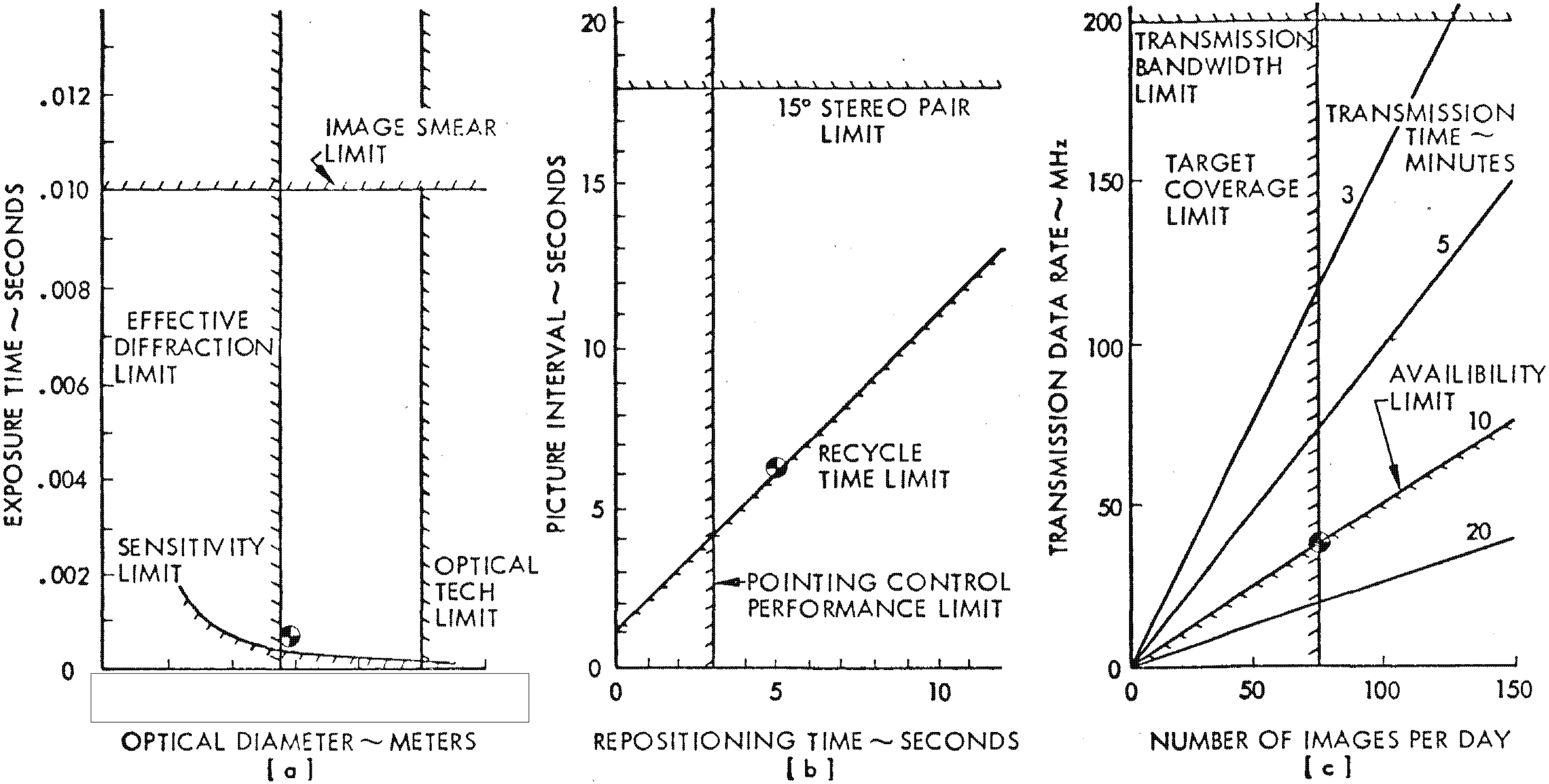

FIGURE 14-2 DESIGN TRADE CONSTRAINTS – BASELINE SYSTEM F

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Design parameters for System F are summarized in Table 14-I.

TABLE 14-I SYSTEM F DESIGN PARAMETERS

ORBIT ALTITUDE	525 Kilometers
RESOLUTION	0.6 Meters
FRAME SIZE	6 Kilometers
EXPOSURE TIME	0.6 Milliseconds
IMAGING TIME	1.0 Seconds
REPOSITIONING TIME	5.4 Seconds
PICTURE INTERVAL	6.4 Seconds
OPTICAL DIAMETER	[redacted]
DATA TRANSMISSION RATE	37.5 MHz

14.2 SYSTEM OPERATION

Operation of System F is very similar to that of System B. Refer to Section 10.2 for additional detail.

14.3 ORBIT AND NAVIGATION

At an orbit altitude of 525 kilometers, daily doppler tracking by one site is sufficient for surveillance operations. At this altitude complete target coverage is assured for more than one year without orbitkeeping. Based on a 1972 launch date and a ballistic coefficient of approximately 10 pounds per square foot, the 525 Km altitude will decay to 485 Km [redacted]

Orbit determination by a single tracking site is sufficient to ensure position prediction accuracy to one kilometer. As no orbitkeeping maneuvers that might detract from prediction accuracy are required, there should be no periods when target location uncertainties preclude image acquisition.

A summary of the orbit and navigation considerations is presented in Table 14-II.

TABLE 14-II SYSTEM F ORBIT AND NAVIGATION DATA

ORBIT: ALTITUDE INCLINATION	525 Km 96.7° (Sun Synchronous)
GROUND VISIBILITY TO STATION AT [redacted] LATITUDE	10 Minutes minimum in a 12-hour period at 5° elevation angle
COVERAGE PERIOD ABOVE 30° LATITUDE	4 to 5 Days at a view angle of ±45°
ORBIT DECAY	Approximately 0.1 Km per day
POSITION ACCURACY	±1.0 Km
UPDATE ORBIT EPHEMERIS	Single tracking site on daily basis

14.4 SYSTEM RF LINKS

The imagery data stored on the screened thermoplastic film is transmitted through an SHF transmission system with an effective radiated power of approximately 27 dbw in the center of the main beam. The 75 stored images can be transmitted directly to the ground terminal in 10 minutes at a baseband rate of 37.5 MHz. Data transmission is initiated by ground command following receipt of link closure confirmation.

The high-gain communication antenna provides hemispherical coverage capability by a combination of antenna pointing and spacecraft rolling. This results in 10 minutes of continuous transmission as the satellite passes directly over the ground terminal. The antenna beam is 6° wide so that antenna pointing within ±2° is adequate to assure quality transmission. The diameter of the half-power beam patch on the ground is only 30 nautical miles, providing a relatively high degree of privacy to the data transmission.

The command, telemetry, and tracking link between the observation satellite and the ground terminal is the same as for System B (Section 10).

14.5 OBSERVATION SPACECRAFT CHARACTERISTICS

The spacecraft configuration is basically that of a cylinder 20 feet long and 9 feet in diameter. The base contains the equipment mounting deck and thermal radiators. Mounted above the base is the optical sensor system which occupies approximately eight feet of length excluding the sunshade. When extended, the sunshade adds another 10 feet to the length of the vehicle.

The communication antenna and two solar paddles are stowed along the cylindrical structure during boost and are deployed to positions in the plane of the equipment mounting deck following separation from the launch vehicle. The sunshade is retracted for boost to permit the spacecraft to fit within the standard OAO shroud. The low-resolution optical sensor system is mounted parallel with, and external to, the high-resolution system.

The two solar-array paddles are rotated about a common axis and are oriented normal to the Earth-sun plane while in the sunlight. During the shadow periods the paddles are oriented so as to minimize the spacecraft drag area.

The satellite may be launched by either a Titan IIIB Agena (N_2O_4) or an Atlas SLV-3B/Agena (N_2O_4) with a standard OAO shroud.

Table 14-III summarizes the major characteristics of the observation spacecraft.

14.5.1 Weight and Power Summary

Table 14-IV lists the weight and power required by each of the major subsystems. The two optical sensor systems have identical transducers of 50 pounds each, and share a common readout section weighing approximately 300 pounds. Primary-power weight includes the solar paddles, power conditioning equipment, and the nickel-cadmium storage batteries.

TABLE 14-III OBSERVATION SPACECRAFT CHARACTERISTICS - SYSTEM F

LENGTH	Overall	20 Feet
DIAMETER	Excluding Solar Paddles	9 Feet
WEIGHT	Initial Injection	4795 Pounds
OPTICAL DIAMETER	High Resolution Low Resolution	
OPTICAL TRANSDUCER	Screened Thermoplastic	4 x 4 Inches
DATA STORAGE CAPACITY	Cycles Images	4.5×10^{10} 150
COMMUNICATIONS	Information Bandwidth Carrier Frequency RF Power Antenna Size Effective Radiated Power Transmission Time/Day	37.5 MHz 2.5 Watts One Foot 31 dbw 10 Minutes
ATTITUDE CONTROL		
SOLAR ARRAY	Array Area Array Power	100 Feet^2 1000 Watts

TABLE 14-IV OBSERVATION SPACECRAFT WEIGHT AND POWER SUMMARY-SYSTEM F

SUBSYSTEM	WEIGHT (Lbs)	POWER (Watts)
HIGH RESOLUTION OPTICS	2100	---
LOW RESOLUTION OPTICS	200	---
TRANSDUCER (2)	100	200
TRANSDUCER READOUT	300	400
COMMUNICATIONS	50	50
COMMAND AND TELEMETRY	50	60
COMMAND AND CONTROL COMPUTER	50	50
ATTITUDE CONTROL	290	110
PRIMARY POWER	300	---
STRUCTURE/THERMAL	870	---
CONTINGENCY	435	100
TOTAL	4795	

14.6 MAJOR SUBSYSTEM PERFORMANCE SPECIFICATIONS

The major subsystems are identified in the following nine paragraphs with some of the more pertinent specifications listed. The first seven subsystems are integral components of the observation satellite and the last two are ground elements. The ground elements include the command and control functions as well as the primary function of data processing.

14.6.1 Optical

	High Resolution	Low Resolution
Type	Corrected Cassegrain	Refractive
Diameter	[redacted]	
Focal Length (Equivalent)		
Pointing Method	Spacecraft	Spacecraft
Angular Field of View	0.65°	3.3°
Weight	2100 Pounds	200 Pounds
Method of IMC	Film Platen	Film Platen

14.6.2 Transducer

Type	Screened Thermoplastic Film
Resolution	100 Lines per Millimeter
Format Size	4 x 4 Inches
Exposure Time	0.3 Millisecond
Readout Time	8 Seconds

14.6.3 Communications

Baseband Width	37.5 MHz
Carrier Frequency	[redacted]
Frequency Modulation Index	1
Transmitter RF Power	2.5 Watts
Transmitter Antenna	29 db Gain, 6° Beam
Antenna Pointing	±2 Degrees

14.6.4 Command and Telemetry

Modified Space-to-Ground-Link Subsystems (SGLS)

Command Decryption

Telemetry RF Bandwidth	4 MHz
Telemetry Transmitter	0.15 Watt
Magnetic Tape Telemetry Storage	10^8 Bits

14.6.5 Command and Control Computer

Magnetic Core Memory	8000 Words
Memory Cycle Time	2 Microseconds
Clock Accuracy (24 hour)	1 part in 10^7
Inputs/Outputs	150/300
External Commands	100

14.6.6 Attitude Control

Antenna Pointing Accuracy	$\pm 2°$

Uses Computer for Storage and Computation

14.6.7 Primary Power

Paddle-Mounted Silicon Solar Arrays	100 Sq. Ft.
Nickel-Cadmium Storage Batteries	100 Pounds
Peak Power Demands (~10% of Time)	Sunlight - 670 Watts Shadow - 640 Watts
Steady-State Demands	Sunlight - 270 Watts Shadow - 240 Watts

Solar-Array Paddles oriented toward Sun within $\pm 15°$

Maximum Power Developed by Array	1000 Watts

14.6.8 Ground Terminal

Located [redacted]

Receive all data from relay satellite on 12-GHz carrier.

Generate and transmit all commands

Two 60-foot antennas.

SOA Low-Noise Receivers for 200-MHz RF Bandwidth

Correlate Tracking Data from All Stations

14.6.9 Ground Reconstruction

Input Data Rate	37.5 MHz
Running Time Before Reload	10 Minutes

Synchronized by Burst Pilot Tone Phase Lock

Image Shape Rectification

Output is Silver-Halide Photographic Positive.

14.7 SYSTEM RELIABILITY SUMMARY

Table 14-V summarizes the reliability estimation for System F for [redacted] operation. Sufficient redundancy is included in the observation satellite design to provide an equivalent mean time to failure (MTTF) of [redacted] The subsystems that contain added redundancy are indicated by an asterisk. The methodology for estimating reliability is included in Section 17.0 of Volume II.

14.8 SYSTEM COST ESTIMATE

Table 14-VI shows a breakdown of estimated total system cost for research and development, production, and system operation excluding government system management and engineering costs. The estimate is based upon the parametric costing data developed in Volume II, Section 18.

Ground terminal costs include the microwave receiving and transmitting terminal, the command and control equipment, the ground reconstruction facility and equipment, and associated ground checkout equipment. System operation up to, and including, the generation of silver-halide positives was estimated at [redacted] dollars per year.

TABLE 14-V SYSTEM F RELIABILITY

SYSTEM ELEMENT	ESTIMATED RELIABILITY (Includes Redundancy)
OBSERVATION SPACECRAFT	Equivalent MTTF =
OPTICAL	.980
TRANSDUCER	.967
COMMUNICATIONS	.988*
COMMAND AND TELEMETRY	.986*
COMMAND AND CONTROL COMPUTER	.973*
ATTITUDE CONTROL	.967
PRIMARY POWER	.990*
STRUCTURAL AND THERMAL CONTROL	.983
LAUNCH VEHICLE	.90

*Added Redundancy

TABLE 14-VI SYSTEM F COST ESTIMATE

SYSTEM ELEMENT	COST IN MILLIONS
SPACECRAFT AND AGE R&D	
GROUND TERMINAL R&D	
LAUNCH VEHICLE R&D	
SPACECRAFT AND AGE PRODUCTION	
SPACECRAFT	
AGE	
LAUNCH VEHICLES	
LAUNCH SUPPORT	
GROUND TERMINAL HARDWARE	
SYSTEM OPERATION	
TOTAL	