Orbit Evaluation Technique for Planetary Orbiters

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A system analysis technique has been developed for evaluating the extent to which candidate spacecraft orbits meet mission scientific objectives under specified engineering constraints. The analytical technique is an extension of the "value function" method that has been used in selecting the encounter aim points for Ranger and Mariner missions. In order to illustrate its use, it is applied to a hypothetical 1971 Mars orbiting mission. Scientific experiments assumed for such a mission include TV, infrared spectrometry, infrared radiometry, celestial mechanics, and Earth occultation. Candidate orbits considered for the hypothetical mission ranged from 60 to 80° in inclination; 12–24 hr in orbit period; 1000–2500 km in periapsis altitude; and −30°–+30° in initial periapsis illumination angle. For the example mission considered, the results indicate that the 12 hr, 80° inclination orbit with an initial periapsis illumination angle of +30° has the highest over-all value. The largest contribution to this high over-all value derives from the imaging experiment, which is sensitive to both the inclination and initial periapsis illumination angle.

Introduction

THE three central scientific objectives of unmanned planetary exploration are to 1) determine the nature and evolution of the solar system, 2) determine the existence of extraterrestrial life, and 3) develop an understanding of the dynamic processes that shape man's terrestrial environment.[1] NASA has flown Mariner spacecraft by Mars and Venus. Current plans call for orbiting Mariner spacecraft around Mars and possibly Venus. These spacecraft will carry scientific instruments for performing orbital experiments and will have a system performance capability consistent with the specific mission design.

For each orbital experiment that utilizes a specific instrument that is flown on a spacecraft with specific performance capabilities there is an orbit that maximizes the attainment of the experiment's objectives over the mission's duration. A single orbit, however, cannot maximize the outcome of each of the experiments comprising a given set of experiments. This paper is concerned with the development and application of a generalized technique for evaluating the degree to which a given orbit satisfies the objectives of a set of experiments associated with a given spacecraft and mission. The technique is an extension of the "value function" method.[2,3] A value function V for a science experiment reflects the degree to which the objectives of the experiment are met, by performing measurements in a given orbit, in terms of instrument characteristics and orbit parameters. A value function for an engineering consideration reflects the degree of performance capability that must be provided to a spacecraft in order to accommodate a given orbit and science payload. For each scientific experiment and engineering consideration, there is a weighting factor W that reflects the importance of that experiment or consideration relative to all the other experiments and considerations. Then $\Sigma_i W_i V_i$ represents an over-all value of a given orbit for a given spacecraft and scientific experiments. Using a digital computer, the time history of the over-all value and the value of the various scientific experiments and engineering considerations can be computed and evaluated for a large number of candidate orbits.

The concept development and evaluation of orbits for 1973–1979 Mars opportunities are detailed in Ref. 4. A brief discussion of the technique, the computer program, the development of typical value functions, and the application to a hypothetical 1971 Mars orbiting mission are given below. It should be noted that although the technique provides useful preliminary data, it does not replace detail final analysis of specific orbits by the science experimenters and mission/system designers.

Orbit Evaluation Technique

The generalized technique developed for assessing the value of candidate orbits for unmanned spacecraft orbiting missions consists of the following nine steps:

1) identification of science experiments and spacecraft engineering considerations that are influenced by the spacecraft orbital characteristics, 2) determination of objectives for the identified science experiments and value criteria relating experiment measurements to those objectives, 3) determination of the impact of the orbital characteristics on the performance capability reflected in the identified engineering considerations, 4) development of value functions for the identified science experiments and engineering considerations, 5) estimation of weighting factors and formulation of an over-all value function, 6) specification of physical and functional characteristics for the identified science instruments and the flight hardware reflected in engineering considerations,

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7) description of candidate orbits in terms of mission/orbital parameters, 8) calculation of the over-all value function, as well as the individual scientific and engineering value functions, as a function of mission time, 9) assessment of the over-all and individual values.

The generalized technique uses a digital computer program. The flow diagram and elements of the program are shown in Fig. 1. The "inputs, control and data storage program" element accepts all inputs, e.g., instrument physical characteristics and the orbital parameters of the orbit to be evaluated, and controls the execution of the computations. The "orbit generator" and "auxiliary trajectory" program elements provide the required flight mechanics data, e.g., geometry, velocities. The "value function program" element contains routines for the 12 scientific experiments and 10 engineering considerations shown. The "output program" element provides for printing and plotting data at several levels of computational detail.

Orbit Generator

The orbit generator computes the spacecraft state as a function of time. A variable time step is used in the orbit computations to provide a finer grid in the vicinity of periapsis. Nodal regression and apsidal precession because of the oblateness of Mars have been considered, and the orbital elements are updated each revolution. Orbit parameters are normally computed every 15 days. This reduces the number of computations and saves machine time. The six subroutines of the orbit generator are: CONE, which computes the positions of the Earth, center of Mars, and a landed capsule located on the surface, relative to the orbiting spacecraft; OCCULT, which determines whether the Earth, sun, and Canopus are physically occulted by Mars (The subroutine computes wherein the orbit occultations occur and their duration.); STAR,which is a special purpose routine that determines if there is stray light within the Canopus sensor stray light field of view; GTRACK, which computes the longitude, latitude, and illumination angle at the subspacecraft point (Altitude, relative surface velocity, and orientation of the spacecraft are also computed); LANDER, which computes landing site location and the relative geometry between a landed capsule and the Orbiter, Earth, and sun; and COMM, which computes range, range rate, and range acceleration for the lander-Earth, lander-spacecraft, and spacecraft-Earth communication links.

Auxiliary Trajectory Programs

Auxiliary trajectory programs are used to perform orbit computations that require considerable data that are unique

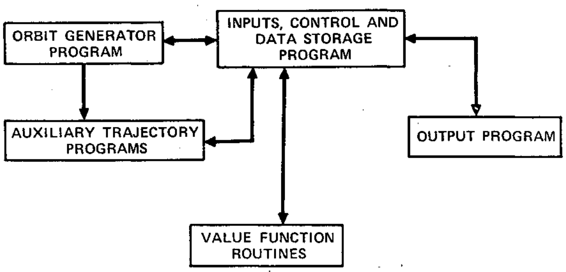

Fig. 1 Computer program flow diagram. Valve function routines for scientific experiments: imaging, infrared spectrometer, ultraviolet spectrometer, occultation, infrared scanner, celestial mechanics, fields and particles, ionosphere sounder, RF noise detector, gamma ray detector, cosmic dust, massive meteor detector. Engineering considerations: solar array sizing, battery sizing, attitude control, fixed antenna coverage at insertion, data return, orbit insertion ΔV, Earth occultation at insertion, automatic orbital sequencing sensors, orbit determination, capsule delivery (from orbit).

to a single value function, or require the results of more than one orbit revolution to arrive at a solution. Four auxiliary trajectory programs are used:

1) POGO, which examines data computed by the occultation routine and predicts the first and last days of sun and Earth occultation. (It also computes the occultation duration history throughout the entire mission.)

2) IMAGE, which is used to evaluate the orbit parameters required for the imaging experiment value function. (This program accounts for coverage of surface features and for frame overlap.)

3) ATCAP, which computes a capsule descent trajectory at any time within a specified orbital revolution. (It also computes the orbit parameters for the capsule deorbit maneuver by considering deorbit ΔV and orientation, Earth occultation at deorbit, capsule-spacecraft descent relay link, capsule descent time, capsule angle of attack at entry, capsule landing site illumination, and capsule-Earth postlanding direct link.)

4) RELAY, which computes the number of orbits required to obtain harmonic synchronization. (This synchronization occurs when the lander-to-spacecraft geometry repeats within specified tolerances. With the required number of orbits determined, the program computes the time history of the look angles from the spacecraft-to-lander and from the lander-to-spacecraft.)

Typical Value Functions

Value functions can be developed for any orbit-dependent scientific experiment or engineering design consideration. Each value function is a quantitative relationship that reflects the ability of a candidate orbit to accomplish one or more of the following: 1) maximize the acquisition and quality of scientific data; 2) satisfy the various spacecraft requirements; 3) optimize system design/operation. The first step in a detailed development of any value function is the identification of the major criteria that influence value. The next step is determination of the functional form, variables, and parameters that define the criteria for value. The final step in the value function development process is the normalization of the functional relationship. Normalization provides a common base for all candidate orbits and constrains the individual value function to the interval 0 to 1.0.

In many cases, a value function has more than one criterion of value. This situation is best handled by specifying a subvalue function for each consideration. The top level value function is then defined by combining the subvalue functions. The subvalue functions may be combined by addition, multiplication, or a combination of both, depending on the interrelationship between each. If not related, i.e., each possessing a specific independent objective, they are averaged. In this case, each subvalue function has equal impact on the top level value function. When the importance of each subvalue function is different, they may be weighted where the weighting factor represents the fractional contribution associated with a particular subvalue function. The sum of the weighting factors equals 1.0 to preserve normalization.

The product of the subvalue functions could be used if it were necessary to achieve all three objectives for a meaningful experiment.

To illustrate the development of individual scientific and engineering value functions, a brief description of an Imaging Experiment and a Battery Sizing value function follows. The technique used in combining individual value functions into a single over-all function is also described. These two value functions are developed in general terms.

Imaging Experiment

The objectives of the assumed imaging experiment are narrow-angle topographic photography, and wide-angle mapping. The imaging value function V_{IE} is formulated as

$$V_{IE} = W_{TE}V_{TE} + W_{ME}V_{ME} \tag{1}$$

where V_{TE} and V_{ME} are value functions for the topographic and mapping experiments, respectively. The weighting factors W_{TE} and W_{ME} allow for biasing the imaging experiment toward either objective. For the example mission presented later in this paper, W_{ME} and W_{TE} are 0.6 and 0.4, respectively. V_{IE} is evaluated on the basis of a complete orbital revolution.

The value for narrow-angle (telephoto) topographic imaging at any instant during an orbital revolution is determined on the basis of three considerations; 1) resolution, including photometric and imaging system characteristics (instantaneous subvalue function $\dot{v}_{RPT}$), 2) observation of specific areas or features of interest (instantaneous subvalue function $\dot{v}_{SFL}$), and 3) surface contrast due to seasonal variations, i.e., the wave of darkening (instantaneous subvalue function $\dot{v}_{SFC}$).

The topographic imaging value function V_{TE} is structured to allow variation in the relative importance of these three considerations

$$V_{TE} = \frac{1}{2\Delta\eta} \int_{\eta_{\max}-\Delta\eta}^{\eta_{\max}+\Delta\eta} (w_{RPT}\dot{v}_{RPT} + w_{SFL}\dot{v}_{SFL} + w_{SFC}\dot{v}_{SFC})d\eta \tag{2}$$

where $\eta_{\max}$ is the true anomaly at which the integrand is maximum, and $\Delta\eta$ is a small half-interval of photo acquisition about this point. Thus, the formulation of this value function yields an average value of the photos which could be acquired on a single orbital revolution. For a typical Mars orbiting mission, $w_{RPT} = 0.5$, $w_{SFL} = 0.30$, and $w_{SFC} = 0.20$.

The subvalue function for resolution is determined here from the relation $\dot{v}_{RPT} = l_{\text{norm}}/l$ where l_{norm} is a normalization constant, and l is the arithmetic mean of the surface dimension of positive and negative ramps at a given slope which produce a specified signal-to-noise ratio, e.g., 3. This latter quantity is determined using techniques developed by JPL for the Ranger spacecraft.[2] The detectable surface dimension l depends on; 1) noise characteristics of the imaging system chain, 2) sensitivity of the imaging system's primary sensor, 3) optical characteristics of the imaging system, 4) currently postulated photometric function for the Martian surface, 5) image smear due to planetary and vehicular motions, 6) lighting and viewing geometry, and 7) orbital altitude above the target.

For the present, $\dot{v}_{RPT}$ is based on $\pm 10°$ ramps and has been normalized to give unity value at 850 km altitude (at the optimum illumination angle with no smear).

The value of the surface feature or area under observation is accounted for by $\dot{v}_{SFL}$. The Mars surface is broken into areas of interest with each area assigned a value ranging from 0.50 to 1.0. Values are allocated on the basis of current scientific interest, with a minimum value of 0.50.

Photometric observations of Mars indicate a significant variation in surface contrast with time and Martian latitude. Observations during periods of peak darkening are desirable and are accounted for by the subvalue function $\dot{v}_{SFC}$. The variation in relative contrast with Mars latitude as a function of time throughout the Mars year has been determined from the results of photometric observations by J. H. Focas.[5] Three levels of contrast are used, high, medium, and low. Values of $\dot{v}_{SFC}$ are allocated to each level as follows, 1.0 is high, 0.7 is medium, and 0.5 is low.

Two considerations are involved in the mapping experiment; 1) area coverage (subvalue function v_{MAC}), and 2) resolution (subvalue function v_{RPM}). The value function for wide-angle mapping is given as

$$V_{ME} = w_{MAC}v_{MAC} + w_{RPM}v_{RPM} \tag{3}$$

where v_{RPM} reflects the average resolution achieved over the total area mapped at any time. Instantaneous values $\dot{v}_{RPM}$ are calculated over the total area mapped in each orbital pass and then averaged to determine v_{RPM}. $\dot{v}_{RPM}$ is determined identically to $\dot{v}_{RPT}$ except that wide-angle camera characteristics are used.

In considering area coverage, v_{MAC} is formulated as the percent of the required area that has been mapped up to a given time during the mission. Acceptable mapping is assumed to occur at those points along the orbit track where; 1) $\dot{v}_{RPM}$ is greater than a specified minimum value (0.1 is used in this analysis), and 2) the latitude falls within the latitude bands desired to be mapped (-60 to $+20°$ latitude). For the example presented later in this paper, it was assumed that $w_{MAC} = 0.8$ and $w_{RPM} = 0.2$.

Battery Sizing

The principal effect of orbit selection on battery sizing arises from the requirement to supply power during off-sun periods when solar cells are ineffective. Conditions requiring battery power are sun occultation by Mars during the orbital phase of the mission and off-sun time during two spacecraft maneuvers; orbit insertion, and orbit trim. Of the two maneuvers, only orbit insertion is pertinent to this value function. It is assumed that orbit trim can be scheduled and accomplished in a manner that will not affect the battery size.

In the case of a fixed design battery capacity, the primary consideration is one of battery adequacy. The battery sizing value function will have a value of unity if it can sustain the nominal load profile over the off-sun durations encountered. If the battery cannot sustain the nominal load, the value will be either zero or some value less than one, depending on the degree to which the allowable depth of discharge is exceeded.

The battery sizing value function V_{BZ} is formulated as

$$\begin{aligned} V_{BZ} &= 1.0,\ d \leqslant d_a \\ V_{BZ} &= 1 - [(d - d_a)/(1 - d_a)],\ d_a < d < 1.0 \\ V_{BZ} &= 0,\ d \geqslant 1.0 \end{aligned} \tag{4}$$

where d is the battery depth of discharge resulting from any particular off-sun occurrence and d_a is the allowable depth of discharge at that particular time.

The allowable depth of battery discharge is determined on the basis of the cycling history and the type of battery em-

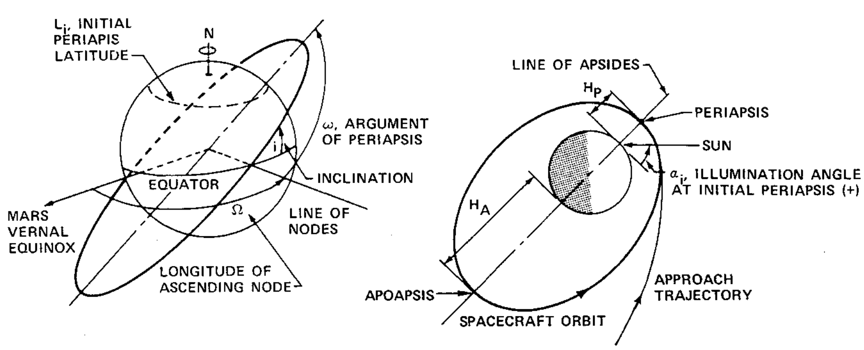

Fig. 2 Definition of orbital parameters.

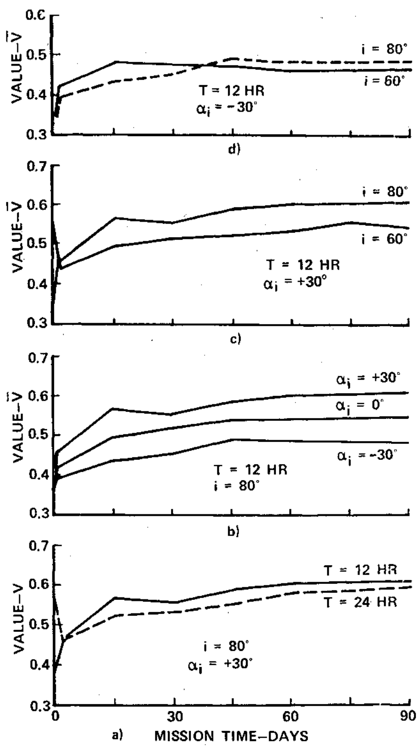

Fig. 3 Effects of orbital elements on $\bar{V}$.

ployed. Silver-cadmium and nickel-cadmium batteries are candidate battery types.

Over-All Value Function

Individual value functions permit evaluation of candidate orbits on the basis of individual considerations only. An over-all value function is used to reflect the combined influence of all scientific experiments and engineering considerations. The basic formulation of the over-all value function $\bar{V}$ at any time t during the orbital mission duration is

$$\bar{V} = \Sigma_i W_{Ei} V_{Ei} + \Sigma_i W_{Si} V_{Si} \tag{5}$$

where W_{Si}, V_{Si} = weighting factor and value function for the ith scientific experiment at time t, and W_{Ei}, V_{Ei} = weighting factor and value function for the ith engineering consideration at time t.

The weighting factors reflect the relative contribution of each scientific and engineering consideration to the over-all value of a given mission. They are determined most readily by first examining the scientific and engineering considerations separately. In each case, the individual considerations are ordered according to their importance and assigned relative values on the basis of 100 points total. The weighting factors are then determined from the following expressions:

$$\begin{aligned} W_{Si} &= N(RVS)_i/(N+1)100 \\ W_{Ei} &= (RVE)_i/(N+1)100 \end{aligned} \tag{6}$$

where $(RVS)_i$ = relative value of the ith scientific consideration, $(RVE)_i$ = relative value of the ith engineering consideration, and N = relative value of science to engineering.

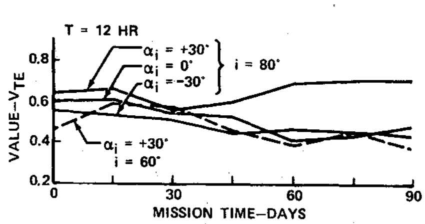

Fig. 4 Topographic experiment value function comparison.

The factor N is used to bias the orbit selection toward either engineering or science. The sum of all scientific and engineering weighting factors is equal to 1.0, consistent with the normalized value function approach.

The weighting factors may vary with mission opportunity because of the evolution of mission objectives and increased emphasis on science relative to engineering with each succeeding mission. They may also vary as a function of time in orbit for a given mission. Such time dependence may be because of either the need for only a limited number of observations to complete a scientific investigation, or the early accomplishment of a specific objective such as the orbit insertion maneuver.

For this study, the weighting factors were held constant throughout the mission. As value functions are deleted from the over-all value function, it is necessary to renormalize the remaining weighting factors so that their sum still remains 1.0. Otherwise, the over-all value is discontinuous in time. This is accomplished by introducing a renormalization parameter $K_r = 1.0 - W_d$ where K_r = renormalization parameter at time t, and W_d = summation of weighting factors associated with value functions deleted through time t. The over-all value function is then determined by

$$\bar{V} = (1/K_r)[\Sigma_i W_{Si} V_{Si} + \Sigma_i W_{Ei} V_{Ei}] \tag{7}$$

All value functions for the hypothetical 1971 mission considered below, except that for orbit insertion ΔV, are evaluated throughout the mission. Consequently, $K_r = 1.0$ for the initial orbital revolution, and $K_r = 1.0 - W_{IV}$ for all subsequent revolutions.

Application to a Hypothetical 1971 Mars Orbiting Mission

Application of the orbit evaluation technique requires specification of mission objectives and nominal mission/system requirements. These provide the basis for identification of candidate orbits and specification of major weighting factors. For the evaluation presented here, the following objectives were assumed for the hypothetical 1971 Mars orbiting mission: 1) observation of the dynamic characteristics of the planet for a period of 90 days; 2) obtaining of data on atmospheric composition, density, pressure, and temperature, and surface composition, temperature, and topography; 3) covering 70% of the planet area, with emphasis on the southern hemisphere; and 4) maximizing data return to Earth.

Consistent with the aforementioned item 1, each candidate orbit was evaluated for the first 90 days only. All candidate orbits were required, however, to have a minimum lifetime of 20 yr from planetary quarantine considerations.

Table 1 Candidate 1971 Mars orbits

Launch date: Arrival date: T, hr/i, deg	May 21, 1971 November 25, 1971 α_i, deg	 H_p, km	Posigrade Initial periapsis near evening terminator H_A, km	 L_i, deg
12/80	+30	1000	17,530	−78.35
12/80	0	2500	16,030	−16.96
12/80	−30	2025	16,505	+48.52
12/60	+30	1970	16,560	−38.80
12/60	0	2370	16,160	−5.90
12/60	−30	2190	16,340	+27.90
24/80	+30	1000	32,410	−78.35
24/80	0	2500	30,910	−16.96
24/80	−30	2025	31,385	+48.52
24/60	+30	1970	31,440	−38.80
24/60	0	2370	31,040	−5.90
24/60	−30	2190	31,220	+27.90

Table 2 Weighting factors for the hypothetical 1971 mission

Value function	Relative value	Weighting factor ($N = 2$)
Science		
Imaging	50	0.333
Infrared spectrometer	15	0.100
Infrared radiometer	15	0.100
Occultation	10	0.067
Celestial mechanics	10	0.067
	100	0.667
Engineering		
Orbit insertion — ΔV	60	0.200
Data return	25	0.082
Attitude control	5	0.017
Battery sizing	5	0.017
Solar array sizing	5	0.017
	100	0.333

Ten scientific and engineering considerations were selected for evaluating the orbiting mission. The scientific considerations reflect the anticipated nominal scientific experiments; imaging, ir spectrometry, ir radiometry, Earth occultation, and celestial mechanics. The engineering considerations characterize the major engineering design aspects; orbit insertion propulsion, communication data return, attitude control, battery sizing, and solar array sizing.

Achievement of the scientific observations is accomplished by the five experiments. Science instrumentation is assumed to be of the type developed for the Mariner-Mars 1969 mission[6]; specifically, a single channel ir spectrometer, a nonscanning wide-angle ir radiometer, and a dual camera (narrow and wide angle) video imaging system. The Earth occultation experiment utilizes the spacecraft S-band telemetry link, and celestial mechanics data are acquired by Earth-based Doppler and range measurements.

The surface area to be mapped photographically is specified to lie between 60° south and 20° north latitude. This mapping will be accomplished with the wide-angle camera with a minimum of 15% overlap between adjacent photos.

It is assumed that only the single 210-ft-diam antenna at the Goldstone Station (DSN) is available for high-data-rate communication in 1971. To simplify mission sequencing and mapping data return (with limited data storage), the Mars orbit will be constrained such that its period is synchronous or subsynchronous with the rotational period of Earth.

Candidate Orbits

Candidate orbits may be specified by various combinations of parameters that reflect insertion conditions, orbit size, and orbit orientation. The following parameters are used here: 1) launch date and arrival date; 2) orbit period T; 3) periapsis altitude H_p; 4) inclination i; 5) initial periapsis solar illumination angle α_i.

Orbital parameters of interest are illustrated in Fig. 2. The orbits are uniquely defined; the initial orbit plane being constrained to contain the approach asymptote (in-plane insertion maneuver).

Orbit period, periapsis altitude, and inclination are constrained by the assumed mission requirements. The orbit period must be synchronous or subsynchronous with the 24-hr period of Earth. Periapsis altitude is constrained to insure a minimum 15% overlap in adjacent mapping photos. The inclination of the orbit cannot be less than 60° in order to map the surface at 60° south latitude.

Twelve candidate orbits (Table 1) were selected for the evaluation.

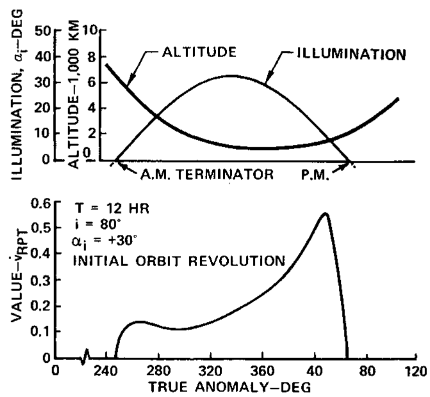

Fig. 5 History of $\dot{v}_{RPT}$ and critical orbit parameters.

Weighting Factors

The relative values and resultant weighting factors for each value function are shown in Table 2. The 1971 science value is biased toward imaging, with half the total science value being allocated to this experiment. The major engineering consideration is orbit insertion ΔV, since it is a major contributor to spacecraft weight.

The weighting factors assume science to be twice as important as engineering, i.e., $N = 2$, in evaluating the candidate orbits.

Results

The results of the evaluation reflect the high weighting factor placed on the imaging experiment in general, and on the wide-angle mapping aspect of that experiment in particular. Figure 3 presents the variation of the over-all value function with mission time as a function of orbit period, initial illumination angle, and orbit inclination. The effect of orbital period T on the over-all value function $\bar{V}$, is shown in Fig. 3a. The examples presented are representative of all 12 cases studied. The data indicate that a 12-hr orbit is superior to a 24-hr orbit. This is caused by the greater area coverage afforded by two orbital passes per day. The 24-hr orbit, however, requires a smaller orbit insertion velocity increment than the 12-hr orbit. This is reflected in the higher over-all value for the 24-hr orbit at 0 mission time. Subsequent to orbit insertion, the value function for this consideration is deleted and the over-all value function is renormalized. Hence, after one day, both orbits have essentially the same value. Both orbits show a general increase of over-all value with time. This behavior is primarily

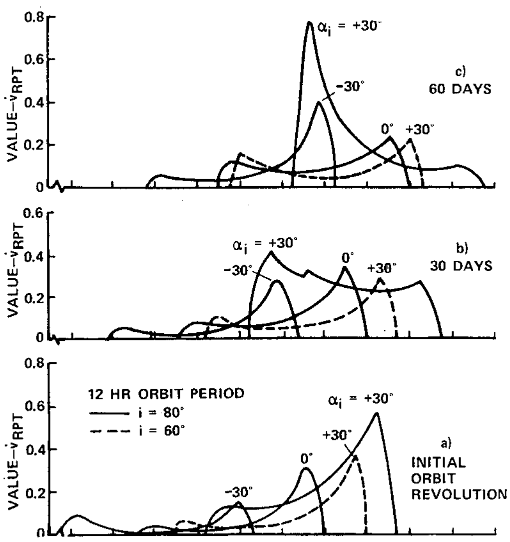

Fig. 6 $\dot{v}_{RPT}$ comparison.

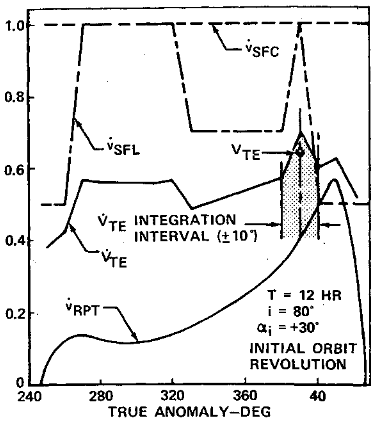

Fig. 7 V_{TE} and associated subvalue functions.

caused by the monotonic increase in the net area being mapped as a function of mission time.

The effects of illumination angle on over-all value are shown in Fig. 3b. The results although given for a 12-hr orbit at an 80° inclination, are representative of those for the other orbit period inclination combinations considered. The data are given for the three initial periapsis illumination angles considered, i.e., +30°, 0° and −30°. The latter two initial illumination angles were included in the analysis because of illumination angle history considerations. For orbit inclinations around 60°, the dominant cause of illumination angle changes is the rotation of the Mars-sun line (≈0.5° per day). Hence, an orbit with an initial periapsis illumination angle of −30° will have a periapsis illumination angle of approximately 15° at the end of a 90-day mission. Such an orbit, therefore, would possess high imaging value late in the mission. Early in the mission, the value of imagery for such an orbit would still be high, since the altitude of the point in the orbit at which the subspacecraft illumination angle is 15° will not be considerably higher than periapsis altitude. The results, however, indicate that an orbit with an initial illumination angle of +30° results in the highest over-all value. This is because wide-angle mapping imagery of acceptable resolution is obtained near the evening terminator early in the mission and near the morning terminator late in the mission. It is therefore, concluded that an initial illumination angle of +30° results in the highest over-all value, provided that mapping near the morning terminator at high southerly latitudes is acceptable, and that the spacecraft science scan platform is compatible with morning terminator imagery.

The data in Fig. 3c shows the effects of orbit inclination on over-all value. The particular case presented is for a 12-hr orbit with an initial periapsis illumination angle of +30°. The data indicate that an orbit inclination of 80° is superior to a 60° inclination. This result is caused by the greater wide-angle mapping coverage of the 80° inclination orbit at acceptable resolution. This is primarily because, for a minimum 15% overlap of adjacent wide-angle frames, the initial periapsis altitude for the 60° inclination is considerably higher than the periapsis altitude for the 80° inclination orbit (see Table 1).

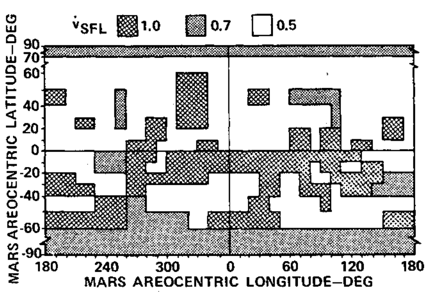

Fig. 8 Value allocations for surface features location $-\dot{v}_{SFL}$.

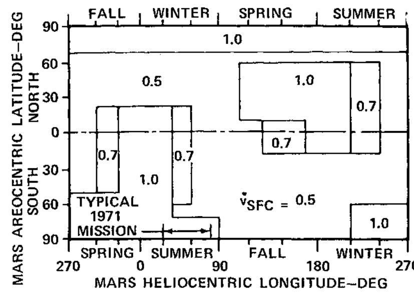

Fig. 9 Value allocations for surface features contrast $-\dot{v}_{SFC}$.

The data shown in Fig. 3c are not representative of all cases considered, as is evident from Fig. 3d, where inclination effects on over-all value are shown for a 12-hr orbit with an initial periapsis illumination angle of −30°. The data show that early in the mission, the 60° inclination orbit is superior to the 80° inclination orbit. This result is primarily caused by attitude control considerations. For the 80° inclination orbit, stray light from the lighted limb of Mars enters the Canopus sensor field of view early in the mission. Since it was a postulated mission requirement to avoid Canopus occultation/interference during the first 30 days of the orbital mission, the 80° inclination orbit was duly penalized for this case.

The foregoing four examples for the over-all value indicated that the imaging value function is, in most instances, the dominant contributor to over-all value. The following paragraphs, therefore, discuss the imaging value function for the candidate orbits. That value function, it will be recalled, combines two value functions; V_{TE} for the telephoto topographic experiment, and V_{ME} for the wide-angle mapping experiment. Figure 4 shows V_{TE} as a function of mission time for several initial periapsis illumination angles. The data are given for a 12-hr orbit period which appears superior to the 24-hr orbit. With the exception of the 80° inclination orbit, with an initial illumination angle of +30°, the general trend of V_{TE} is to decrease with mission time. The behavior of V_{TE} for the 80° inclination/+30° illumination orbit results from high-quality morning terminator photography late in the mission. To understand the general trend of the topographic experiment value function V_{TE} with time, it is necessary to examine the time history of the subvalue functions comprising V_{TE}. Figure 5 shows the subvalue function for the combined effects of the Mars photometric function and the telephoto camera system characteristics $\dot{v}_{RPT}$ as a function of true anomaly for the initial orbital pass of a candidate orbit. The time history of the illumination angle and spacecraft altitude are also shown. The effects of several initial illumination and inclination angles on the variation of $\dot{v}_{RPT}$ with true anomaly, for a 12-hr orbit, are shown in Fig. 6 for the initial orbital pass and for the orbital passes occurring on the 30th and 60th mission days. The increase in $\dot{v}_{RPT}$ for the +30° initial illumination angle orbit near the morning terminator as the mission progresses is evident.

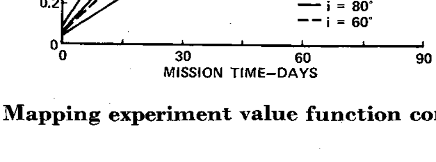

Fig. 10 Mapping experiment value function comparison.

Table 3 Candidate orbit value comparison

T, hr.	i, deg.	a_i, deg.	IMAGING	IR SPECTROMETER	IR RADIOMETER	EARTH OCCULT	CELESTIAL MECH	INSERTION ΔV	DATA RETURN	ATTITUDE CONTROL	BATTERY SIZING	SOLAR ARRAY SIZING	OVERALL VALUE
12	80	+30	H	M	H	None	H	H	H	H	H	H	H
		0	H	L	L	L	M	L	H	H	H	H	M
		-30	M	L	M	M	L	L	L	L	H	H	L
12	60	+30	M	H	H	L	M	H	H	H	H	H	M
		0	L	H	L	M	L	L	L	H	H	H	L
		-30	L	H	M	H	L	L	L	H	H	H	L
24	80	+30	H	M	M	None	H	H	H	H	H	H	H
		0	M	L	L	L	M	L	H	H	H	H	M
		-30	L	L	M	M	M	L	L	H	H	H	L
24	60	+30	L	H	H	L	M	H	H	H	H	H	M
		0	L	H	M	M	M	L	L	H	H	H	L
		-30	L	M	L	H	L	L	L	H	H	H	L

LEGEND: H—HIGH, M—MEDIUM, L—LOW

The topographic experiment value function V_{TE} also includes the contributions of surface features location ($\dot{v}_{SFL}$) and surface wave-of-darkening contrast ($\dot{v}_{SFL}$). The time history of the three subvalue functions comprising V_{TE} are shown in Fig. 7 for the initial pass of a candidate orbit. The area weighting factors used in this example for surface location and contrast coverage are shown in Figs. 8 and 9, respectively. The time history of V_{TE} (i.e., $\dot{V}_{TE}$), is also indicated in Fig. 7. To obtain a representative V_{TE} for each orbital pass, $\dot{V}_{TE}$ is averaged over a true anomaly range of 20° centered about the true anomaly at which $\dot{V}_{TE}$ is a maximum.

The history of the value of the wide-angle mapping element of the imaging experiment V_{ME} is shown in Fig. 10. For all the illumination and inclination angles considered, V_{ME} increases monotonically with mission time. This is primarily caused by v_{MAC}, the subvalue function reflecting the total area mapped to acceptable resolution within the required latitude band. As previously indicated, acceptable resolution for the mapping experiment occurs when $\dot{v}_{RPM} \geqslant 0.1$, where $\dot{v}_{RPM}$ is a subvalue function reflecting photometric and wide-angle camera system characteristics. A time history of $\dot{v}_{RPM}$ for an orbital pass of a candidate orbit is shown in Fig. 11. The true anomaly range over which v_{MAC} contributes to the mapping experiment function is indicated.

The results obtained from evaluating the 12 candidate orbits for the hypothetical 1971 Mars orbiting mission indicate that the 12-hr, 80° inclination orbit with an initial periapsis illumination angle of +30° has the highest over-all value. The results further indicate that 1) the largest contribution to this high over-all value comes from the imaging experiment

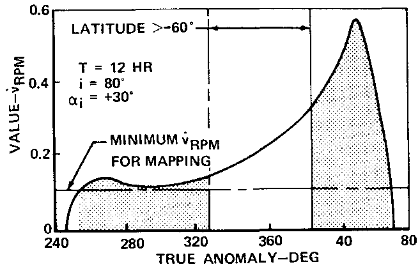

Fig. 11 History of $\dot{v}_{RPM}$ with coverage constraints.

value function, and 2) the imaging experiment value function is sensitive to the inclination and initial periapsis illumination angles of the orbit. Despite its highest over-all value, this orbit has shown shortcomings that may render it unattractive. This is indicated in Table 3, where the contribution of each of the 12 candidate orbits to each of the 10 scientific and engineering value functions is indicated qualitatively. Note that the aforementioned orbit does not satisfy the objectives of the Earth occultation experiment. That orbit is free of Earth occultation for the entire 90-day orbital mission. This situation can be remedied by either decreasing the initial illumination angle of that orbit from +30° to a lower value, e.g., +10°, or by changing the inclination from 80° to 60°. This will result in Earth occultations for the first few orbital passes early in the mission. Also, the orbit insertion ΔV requirements will be reduced. However, in either case, the value of the imaging experiment (as represented here) will be degraded. An alternate solution would be to place two spacecraft in two different orbits about Mars. The first orbit could be the 80° inclination, +30° illumination, 12-hr orbit, which is preferred for all experiments except Earth occultation. The second orbit could be the 60° inclination/0° illumination orbit. Such an orbit provides Earth occultation for the first few weeks of the orbital mission, and satisfies the ir spectrometry experiment better than the first orbit. A 24-hr period can be selected for the second orbit in the event that orbit insertion ΔV requirements for a 12-hr period are deemed excessive.

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