AN ORBITAL RADAR MAPPER OF VENUS IN THE 1980'S - MISSION DESIGN AND ANALYSIS

By

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September 12, 1973

(NASA-CR-140029) AN ORBITAL RADAR MAPPER OF VENUS IN THE 1980'S: MISSION DESIGN AND ANALYSIS (Martin Marietta Aerospace, Denver, Colo.) 51 p HC $5.75 CSCL 03B

N74-34274

Unclas

G3/30 49031

Backup Document for IAA Synoptic Scheduled for

Publication in the Journal of Spacecraft & Rockets, December 1974

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ABSTRACT

A reasonable approach to the examination of Venus topography, obscured for photographic imaging, is available in the application of airborne radar mapping systems to an orbiter mission about the planet. Extrapolating the improving capabilities of Earth-based radar study of Venus into the 1980's suggests that only a non-uniform, poorly resolved surface profile will be possible relative to the potential for 100% coverage at 100 meter resolution with an orbital radar. The intent of this paper is to define mission opportunities favorable for a Venus orbital mapper during the 1980's, to examine orbit design problems associated with mapping radar systems, to establish what flexibility exists for an adaptive mapping strategy, to contribute to the sizing of particular spacecraft systems, to suggest a reference mission design and demonstrate mission feasibility.

This study has been conducted as part of Contract NAS2-7204 by Martin Marietta Aerospace, for the Ames Research Center, NASA. The author wishes to acknowledge the contributions of J.S. MacKay of the Ames Research Center and D.B. Cross of Martin Marietta in guiding the direction of mission analysis and design, and suggesting areas of productive investigation. Characteristics of the radar system design reported here reflect the creative efforts of F.A. Vandenberg, Martin Marietta. R.J. Boain and D.V. Byrnes of Martin Marietta have contributed to the navigation and orbit determination analyses.

INTRODUCTION

An existing challenge to our exploration of the solar system is the acquisition of a topographical profile of the planet Venus, where surface features are obscured for optical mapping systems by a dense atmosphere. Projecting the improving capabilities of Earth-based radar imaging into the 1980's indicates that the Venus surface might then be resolved in a very non-uniform fashion, with resolutions ranging from 500 meters at the planet equator to 5 km near the polar regions. Only 25% of the total surface area will potentially be mapped at resolutions better than 1 km (Ref 1). In contrast, science requirements generally suggest that a surface resolution nearer 100 meters over at least one entire hemisphere is necessary to provide an adequate map for any comprehensive analysis of surface structure. The application of airborne radar mapping systems, which have proven effective on Earth during the last two decades, to an orbiting spacecraft about Venus offers that degree of mapping potential. In operation within Earth's atmosphere, those systems have successfully resolved terrain features and vegetation characteristics beneath the dense cloud cover over South American jungles (Ref 2).

The structure of the Martin Marietta study (Ref 3) of a Venus orbital radar mission, from which this paper has been derived, was directed toward the feasibility assessment of a broad range of mission opportunities, spacecraft systems configurations, and orbit design philosophies. From that overview have evolved specific conclusions and recommendations relevant to the mission concept. This paper emphasizes those conclusions which pertain primarily to evaluation of mission opportunities, design of the mapping orbit and its interplay with the radar system capability, and the requirements placed upon spacecraft propulsion systems. Key mission/systems trades which form the basis of the presented conclusions and recommendations are detailed. Specifics of the proposed radar system are addressed as they shape mission design, and as they relate to the surface coverage achieved by the suggested reference mission.

The paper has drawn in part from various sources which have recently examined the orbital radar mapping concept. A 1972 report by the Jet Propulsion Lab (Ref 4) has sought to define a radar system suitable for orbital application, and has thus contributed to our general understanding of radar system design. A subsequent JPL study (Ref 1, ibid) expands their early work into many related mission and systems aspects of a Venus orbital radar, and describes important science considerations for the radar exploration of the planet. In that work JPL has emphasized the capabilities of radar mapping from circular to near-circular orbits. Finally, a report by the Ames Research Center (Ref 5) has formed a foundation for much of the mission design and systems evaluations of the Martin study, and has stimulated many productive directions of investigation pursued in this assessment of Venus radar mapping mission feasibility.

MISSION CONCEPT

Mission opportunities for a Venus orbiter have been assessed within the timeframe of the early to late 1980's -- specific launch windows are defined for the years 1983, 1984, 1986, 1988, and 1989. This mission set reflects the philosophy at study inception of the most likely period for planning an orbital mapping of Venus, considering the expected austere climate of space funding. Given that climate, the emphasis here has been toward implementing state of the art technology, with reasonable assumptions of growth into the proposed timeframe, to gain the most science return for the least dependence on new technology development. In keeping with this rationale, much attention has been directed toward the launch capability of Titan IIIE/Centaur (although the possibilities of a Shuttle/Centaur are also addressed), and toward the performance characteristics of Viking class orbit insertion propulsion.

Science objectives for the radar examination of Venus have set the tone for mission design. So too has the desire to provide in this preliminary stage of mission definition a flexible mapping strategy. That strategy should be capable of adaptive response to whatever knowledge of the planet becomes available by way of Pioneer Venus or Russian missions, or through Earth-based radar study, prior to flight. Both considerations support a design which provides access to the entire Venus surface, for either a complete area profile map, or for a mission which concentrates mapping on a variety of widely dispersed points of interest. Primary emphasis here has been toward the objective of contiguous area mapping, at surface resolutions near 100 meters. In either case, the implications for design of the mapping orbit lead to orbit inclinations near polar to gain full latitude coverage, and to a mapping mission duration sufficiently long to access all longitudes. One Venus rotation on axis takes 243 days, and thus an orbiter life of at least 122 days is necessary with a strategy which maps a full 360° about the planet each mapping pass, or of at least 243 days with a strategy that sweeps a 180° pole-to-pole swath each pass. To achieve resolutions approaching 100 meters, mapping altitude should be reasonably low consistent with other elements constraining this parameter. The size of the orbit (eccentricity) has been a principal mission trade in this study, and as such is treated parametrically through much of the analysis. Preference for an orbit design of .5 eccentricity will be developed as various aspects of the mission are examined.

The role played by the radar system in shaping mission design is somewhat analogous to that of an optical imaging system, with a few notable exceptions. The proposed mapping radar is of necessity side-looking; that is, the radar beam vector is displaced out of the plane of motion by a specified angular distance referred to as side-look angle (SLA). This offset provides the shadowing effect necessary for a radar surface relief map, and corresponds to the natural relief provided by sun illumination and shadows in an optical map. For both mapping techniques, limits are imposed on the altitude range from which a desired surface resolution can be achieved. In the case of an optical mapping system these limits are directly tied to the focal length range of the imaging camera, be it photographic or television. For radar imagery these altitude limits derive from radar power requirements and signal interpretation constraints. Radar power requirements increase in proportion to the cube of distance to the target, if resolution is held constant. In addition, for increasingly large radar range, the swept area under a fixed beamwidth grows to a point where the information returned becomes meaningless. The problem is referred to as radar signal ambiguity and represents a situation where the individual radar signal pulses from successive series of bursts along the orbital path cannot be differentiated, and therefore cannot be processed into a cohesive picture (Ref 3).

As is the case for both optical and radar area mapping, design of a circular orbit presents a constant altitude and range profile to the imaging systems and therefore places the least demand on those systems. The circular orbit does, however, exact a significant penalty in large orbit insertion propulsion requirements and reduced inserted payload capability. Design of an eccentric orbit, by contrast, reduces orbit insertion propulsion requirements to much more manageable levels, while forcing the imaging systems to contend with the problems of variable altitude. In the following sections, this latter approach will be developed as the preferred design for the orbital radar mapping of Venus consistent with the guiding objective of establishing a mission of reasonable cost, and a method of dealing with the demands on the radar system will be suggested.

PERFORMANCE CHARACTERISTICS OF VENUS OPPORTUNITIES IN THE 1980'S

Assessment of mission performance was initially directed toward definition of the performance capabilities of the considered Earth-Venus opportunities. The view was one of optimizing inserted weight in orbit for the assumptions of 1) Titan IIIE/Centaur launch, 2) a 20 day launch window, 3) Viking class orbit insertion propulsion, 4) impulsive coplanar insertion into a moderately eccentric orbit ($e = .3$) at 400 km altitude, and 5) a "rubber" propellant load and payload weight. A periapsis altitude of 400 km was selected as a nominal target for insertion, reflecting the expected minimum altitude (300 km) necessary to avoid entry into the sensible Venus atmosphere (at 130 km) over an extended mission, and the expected encounter errors in radius (about 75 km) from navigation uncertainties. For all mission years both Type I and II trajectories (transfer angle less than or greater than 180°) have been examined, but only where performance is comparable (as in 1983 and 1984) have both types been treated to a comprehensive description. To complete the performance assessment, the characteristics of Shuttle/Centaur as a launch alternative are included and discussed.

Isolation of the more promising periods for trajectory transfers to Venus orbit during the decade has drawn in part from the launch and arrival contours presented in Ref 5. Figures 1 and 2 illustrate a typical set of Earth departure conditions (C_3 and launch azimuth declination, DLA) and Venus arrival conditions (Vhp magnitude and declination) for opportunities in 1983. In Table 1 are listed the resulting performance characteristics and trajectory parameters for all of the opportunities, with 20 day launch windows optimized under the assumptions detailed above. The table presents launch and arrival dates, C_3, trip time, transfer angle, Vhp, and a measure of payload weight potential for the start, midpoint, and end of each window. For each mission, a Type I trajectory exists with respectable performance. In 1983 and 1984, Type II trajectories yield comparable performance capability and are included in this list.

Of relevance to the design of the mapping orbit, and as a standard by which the mission years can be assigned more than a qualitative value, the profile of payload weight capability against mapping orbit size (eccentricity) becomes a very important trade. In the curves of Figure 3 this picture is presented for each opportunity, considering the worst day performance over the appropriate 20 day window defined in Table 1. Using as a guide the arbitrary assumption of a 1000 kg desired payload weight, the curves can be related to mission feasibility. For all opportunities to be viable, with sufficient margin to account for normal systems growth and more realistic

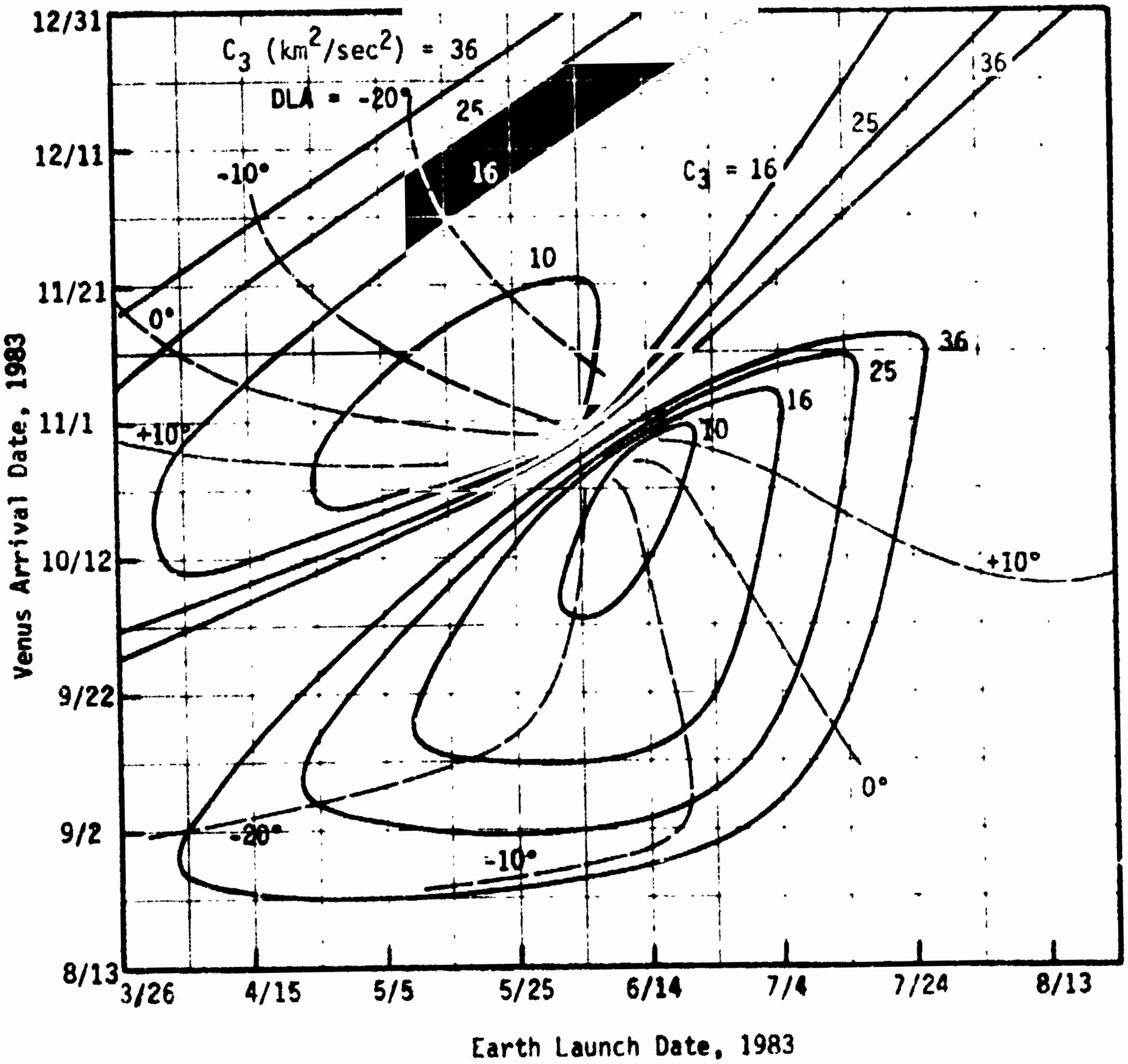

Figure 1 Launch Energy Profile for 1983

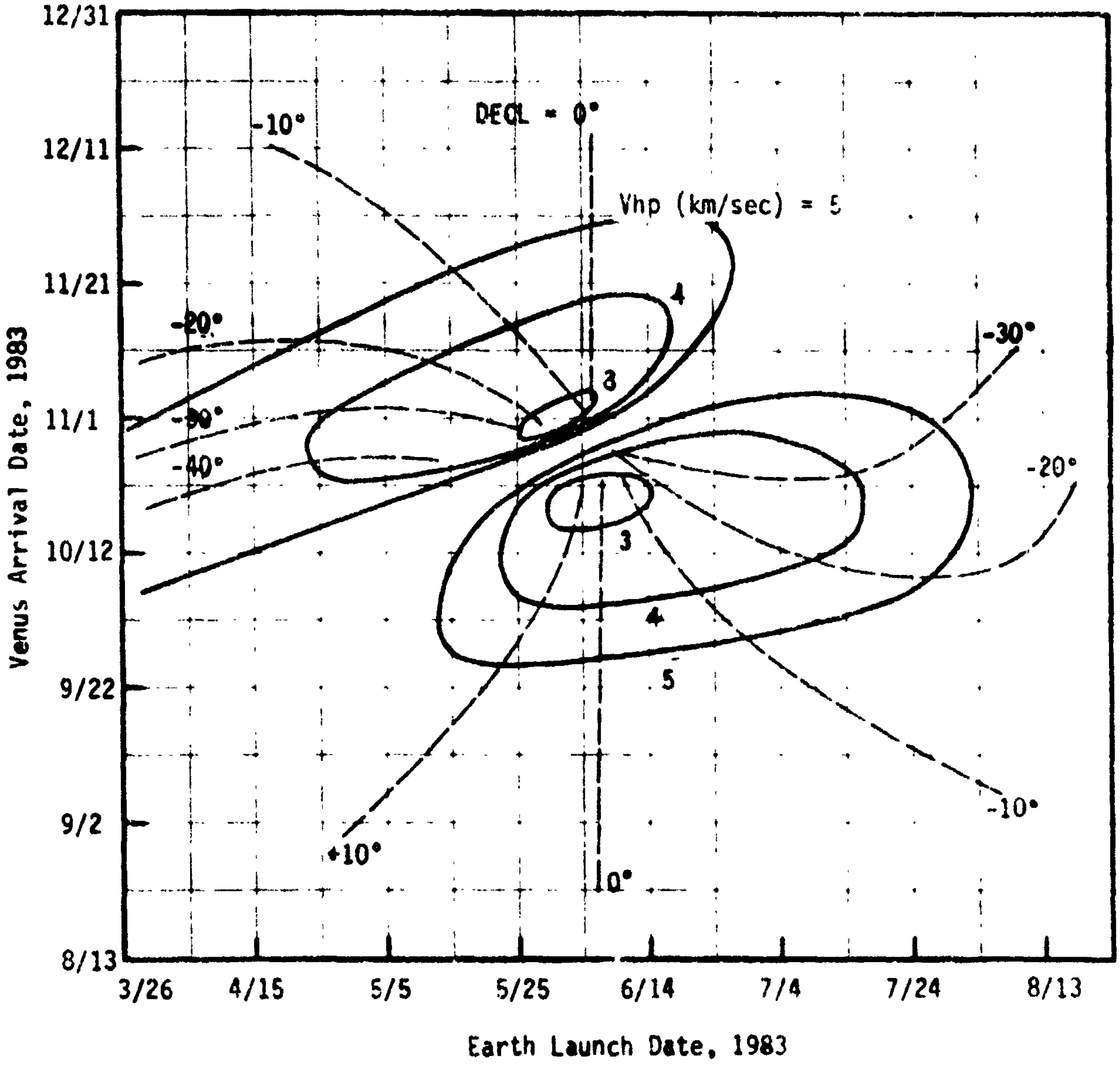

Figure 2 Arrival Vhp Profile for 1983

Table 1 Trajectory Performance Characteristics for All Missions

Mission	Window Day	Launch Date	Arrival Date	Trip Time (days)	Transfer Angle (deg)	C_3 (km^2/sec^2)	Vhp (km/sec)	Payload Weight in Orbit for e = .3 (kg)
1983 I	1	06-02-83	10-15-83	135	162	11.00	3.08	1352
	10	06-11-83	10-26-83	137	171	8.49	2.92	1463
	20	06-21-83	10-23-83	124	157	10.65	3.14	1365
1983 II	1	05-21-83	11-02-83	165	202	6.40	3.55	1380
	10	05-30-83	11-02-83	156	194	6.32	3.36	1427
	20	06-09-83	11-03-83	147	186	10.10	3.09	1376
1984 I	1	12-09-84	05-12-85	154	178	12.96	3.34	1233
	10	12-18-84	05-12-85	145	169	9.25	3.95	1213
	20	12-28-84	05-13-85	136	161	7.98	4.16	1191
1984 II	1	11-21-84	05-21-85	181	211	14.67	3.12	1264
	10	11-30-84	05-22-85	173	203	13.06	2.91	1342
	20	12-10-84	05-27-85	168	201	13.10	2.72	1304
1986 I	1	08-20-86	12-15-86	117	140	8.92	4.91	987
	10	08-29-86	12-18-86	111	136	10.71	4.69	1008
	20	09-08-86	12-21-86	[illegible]	132	14.61	4.46	987
1988 I	1	04-02-88	07-26-88	115	[illegible]	15.10	4.72	914
	10	04-11-88	07-30-88	110	[illegible]	18.14	4.26	959
	20	04-21-88	08-03-88	104	132	24.18	3.89	919
1989 I	1	11-10-89	03-05-90	115	144	15.50	3.65	1140
	10	11-19-89	03-10-90	111	143	16.93	3.20	1197
	20	11-29-89	03-17-90	108	145	20.86	3.55	1140

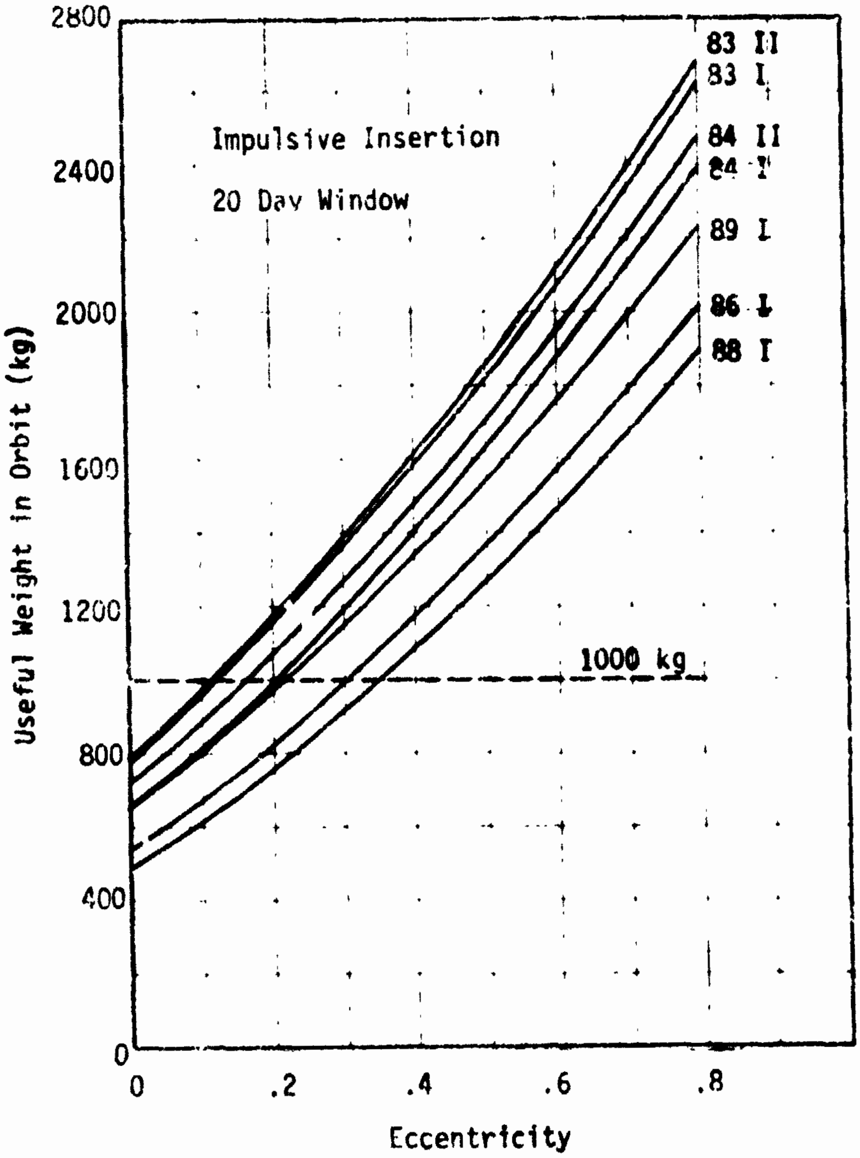

Figure 3 Mission Year Performance Capability (Impulsive)

assessment of orbit insertion ΔV, orbit eccentricity would have to be .5 or greater. As eccentricities lower than .5 are considered by the design, mission years become lost in the order 1988, 1986, 1989 with decreasing eccentricity. An eccentricity of .2 then represents probably the tightest orbit which could be realized for any of the mission years. Again, these preliminary conclusions rest on the set of all assumptions previously discussed, and serve to indicate trends and potential bounds to mission design.

As the mission study progressed into other areas related to performance capability, this payload profile became modified to incorporate certain design refinements. Specifically, the analyses discussed in following sections concerning more sophisticated and complete treatment of orbit insertion requirements indicated that allowance be made for a 20° apsidal shift and the inclusion of significant finite burn losses. These considerations strongly influenced the design of the insertion propulsion system and ultimately led to the treatment of a 3-engine Viking insertion system. That design, and its performance characteristics, were in turn incorporated into the capability profile, and the curves of Figure 4 illustrate the more realistic picture, with inclusion of higher inert weight values, ΔV adjustments and losses. Details of these adjustments are discussed later. It is sufficient to note here that in this more realistic view, the 1986 and 1988 opportunities would be inadequate for orbit eccentricities of .5 or less, given the qualifying assumptions and the 1000 kg payload.

The value of Shuttle/Centaur as a launch alternative can be derived from Figure 5, which reflects the more reasonable insertion assumptions included in Figure 4 for Titan performance. Inserted weights are shown ranging from best to worst mission opportunities of the mission set.

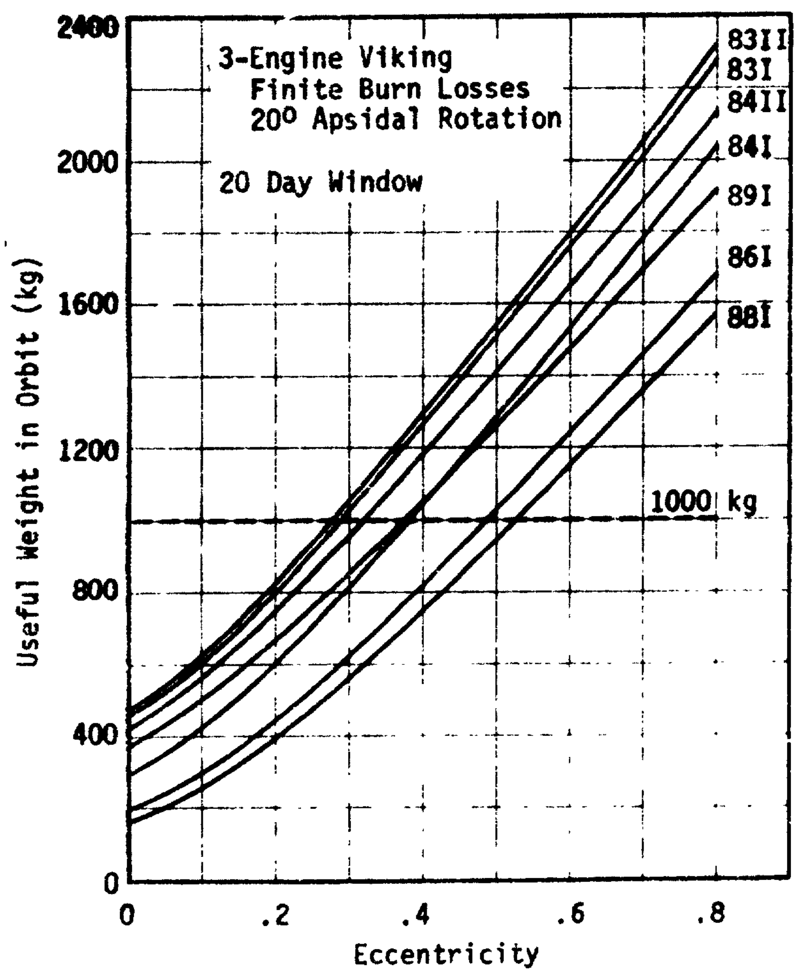

Figure 4 Mission Year Performance Capability
(3-Engine Viking with Losses)

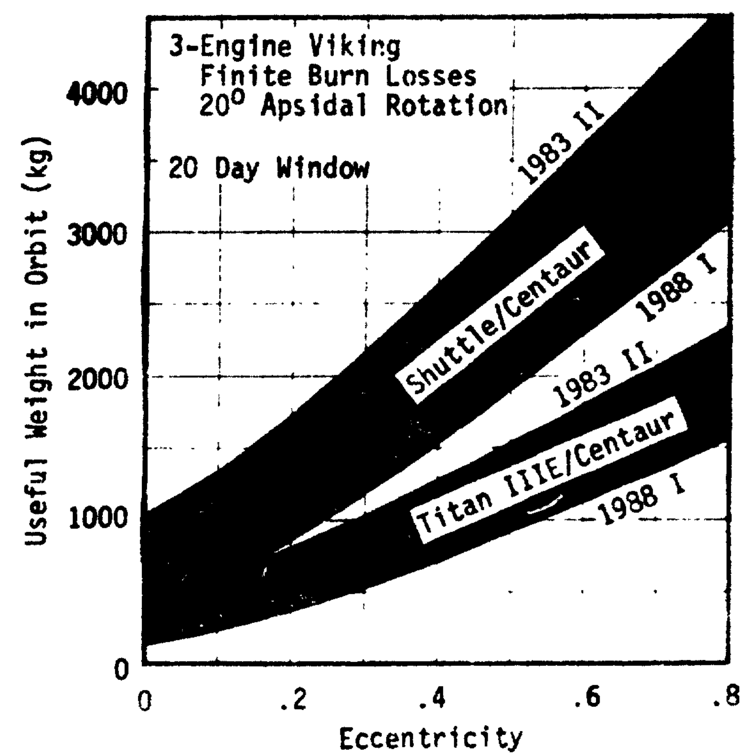

Figure 5 Performance Comparison, Titan and Shuttle
(3-Engine Viking with Losses)

The potential capability of Shuttle/Centaur for delivering a payload to Venus is about twice that of Titan IIIE/Centaur, and this opens the possibility of a dual spacecraft mission. With two spacecraft in orbit a variety of options becomes available. Probably the most attractive mission alternative would involve a time-staggered insertion for the two vehicles which would place them into identical but lagging orbits. In this concept, the leading orbiter could map in a wide area resolution mode, gaining the desired complete, overlapping surface profile. The second spacecraft in its trailing orbit, phased to allow digestion of the area map, could then be a high resolution mapper concentrating on near real-time selected features of interest. The important conclusion to be drawn is that consideration of Shuttle/Centaur provides a flexibility to mission design beyond the simple performance enhancement of existing launch opportunities or the opening of marginal opportunities.

COMPARATIVE ENCOUNTER GEOMETRY

An examination of planet/trajectory geometry at Venus encounter was necessary to determine the degree to which orbit design could be divorced from mission year dependency. Determination of those parameters which were significantly variable with mission opportunity would serve to indicate the range of situations which the mission analysis would have to treat.

Table 2 lists characteristic orientations of the spacecraft, Earth, sun, and Vhp vector at Venus arrival for all years, corresponding to the same launch windows specified previously. It should be noted at this point that the convention used throughout this study for directions with respect to

Table 2 Trajectory and Vhp Arrival Characteristics For All Mission Years

Mission	Window Day	Arrival Date	Sun-S/C-Earth Angle (deg)	Vhp (km/sec)	Vhp Decl. (deg)	Vhp Right Asc. wrt Sun (deg)	Vhp Right Asc. wrt Earth (deg)
1983 I	1	10-15-83	104	3.08	14	293	189
	10	10-26-83	95	2.92	-13	263	168
	20	10-23-83	97	3.14	-22	271	174
1983 II	1	11-02-83	91	3.55	-30	241	150
	10	11-02-83	91	3.36	-26	244	153
	20	11-03-83	90	3.09	5	244	154
1984 I	1	05-12-85	115	3.34	4	309	194
	10	05-12-85	115	3.95	29	315	200
	20	05-13-85	114	4.16	32	317	203
1984 II	1	05-21-85	107	3.12	26	269	162
	10	05-22-85	106	2.91	18	275	169
	20	05-27-85	102	2.72	-3	269	167
1986 I	1	12-15-86	114	4.91	-42	318	204
	10	12-18-86	111	4.69	-43	312	201
	20	12-21-86	108	4.46	-43	303	195
1988 I	1	07-26-88	110	4.72	39	318	208
	10	07-30-88	106	4.26	38	307	201
	20	08-03-88	103	3.89	34	293	190
1989 I	1	03-05-90	108	3.65	-31	305	197
	10	03-10-90	104	3.20	-24	290	186
	20	03-17-90	99	3.55	-13	268	169

the Venus surface is referenced to a system defined in the Venus orbit plane, with the positive x-axis directed toward the Venus vernal equinox of date. This convention preserves a common sense of direction among all the bodies considered in the mission, and avoids the reversed equatorial system of Venus arising from the planet's reverse spin.

From Table 2 can be observed a general similarity in arrival geometry for all cases in the Venus orbit plane. Noting the right ascension of Vhp with respect to Earth, the Vhp vector is directed opposite the Earth vector, within $\pm 30^\circ$ of an exact 180° displacement, for all mission opportunities. The implication for mission design is that insertion out of Earth view can probably be expected for most considered Venus arrivals. In addition to this, the arrival angle between sun and Earth vectors is restricted to between 90° and 115° in all cases. This relatively invariant orientation of sun, Earth, and Vhp vector in the orbit plane implies that for any specific orbit design, characteristics of occultation patterns, orbit stability, power acquisition, thermal load, and communications scheduling should be generally independent of mission opportunity.

The significant variations in the arrival picture involve characteristics of the Vhp vector other than its in-plane location. Both Vhp magnitude and declination exhibit substantial variation and mission year dependency, even within the 20 day launch window defined for each opportunity. Vhp magnitude varies from just under 3 km/sec for missions in 1983 and 1984 to a maximum near 5 km/sec in 1986. This condition has a direct influence on the orbit insertion ΔV requirements for transfer to an orbit of any eccentricity, since this parameter measures the amount of energy which must be removed from the heliocentric trajectory to effect the orbit transfer. Vhp also varies considerably in declination out of the orbit plane, ranging from about 40° south in 1986 to 40° north in 1988. This declination variation (along with Vhp magnitude) is an important determinant of the locus of orbit orientations and periapsis latitude solutions achieved by coplanar transfer, and is addressed in detail in the next section.

ARRIVAL ORBIT ORIENTATION POSSIBILITIES

For each arrival date and Vhp vector, of each launch/arrival date set for the considered missions, there exists a set of orbit inclination/periapsis latitude combinations which can be achieved under the assumption of nominal coplanar transfer into orbit. This set is derived from rotation of the Vhp vector 360° about the target planet in the aimplane, when the trajectory is targeted to specific arrival conditions. The resulting curve of periapsis location is referred to as the subperiapsis locus. When this locus is related to the corresponding orbit inclination, orbit orientation possibilities can be illustrated by football shaped curves. Figure 6 shows these conditions for a 1983 Type I mission where each "football" represents orbit orientation loci for the appropriate launch/arrival date set at the beginning, midpoint, and end of the specified 20 day window.

If only circular orbits were to be considered, periapsis location in latitude would have significance only in positioning the insertion burn, and the primary concern would be that of achieving a polar orbit inclination. But given the attractive orbit insertion propulsive requirements of eccentric orbits, the nominal periapsis latitude range for each mission year assumes a particular importance. Latitude of periapsis has a direct bearing on where,

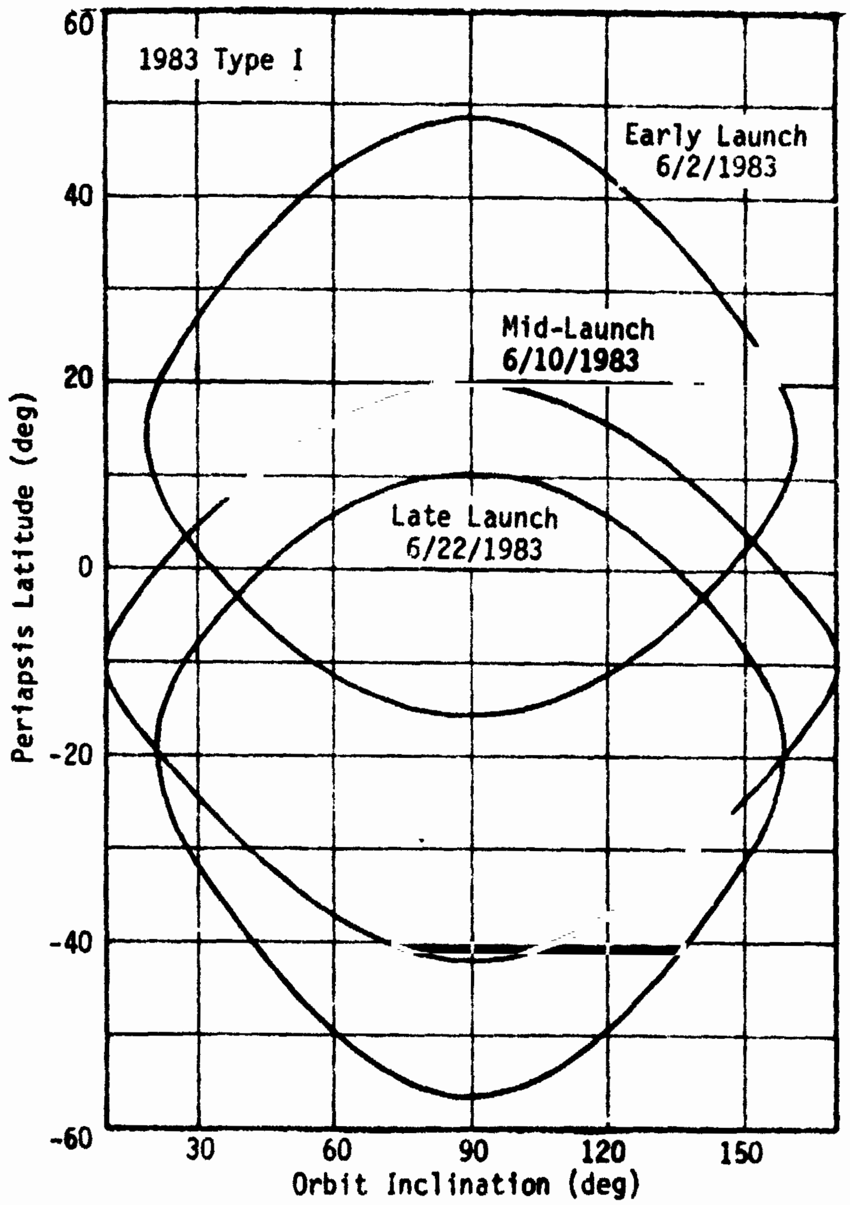

Figure 6 Orbit Orientation Possibilities For 1983 Type I

with respect to the surface, mapping is centered if that mapping is restricted to a range of altitudes about periapsis. The presumption is that for an orbit of medium to large eccentricity, radar mapping cannot be achieved over a complete 360° orbit transit, given the radar constraints discussed earlier.

From Figure 6 it can be observed that if only polar orbits ($i = 90^{\circ}$) are considered for the mapping mission, there exists a double-valued periapsis latitude solution for nominal insertion, corresponding to northern and southern locations, and these solutions can shift considerably over the launch window. This shift is directly the result of variations in Vhp declination already discussed. Northern periapsis locations correspond in all considered cases to periapsis near the orbit descending node, with southern solutions corresponding to an ascending node periapsis. This situation has important implications for orbit stability, and will be expanded upon in a subsequent section.

In Figure 7 are illustrated the periapsis latitude solutions for just the polar orbit inclination for each opportunity, as these solutions vary over the launch window appropriate for each mission. The curves indicate first of all that in considering the set of all Venus opportunities during the 1980's, periapsis for an eccentric orbit could be placed almost anywhere over the Venusian surface, from near the south pole in 1986 to the north pole in 1988, with points in between in other years. The second type of information which can be derived from the figure is the amount of apsidal rotation which would be required to place periapsis at some specified latitude for each opportunity. If a balanced treatment were to be given each hemishere, for example, an equatorial periapsis location would be preferred, and Figure 7 would indicate the potential cost of gaining that location.

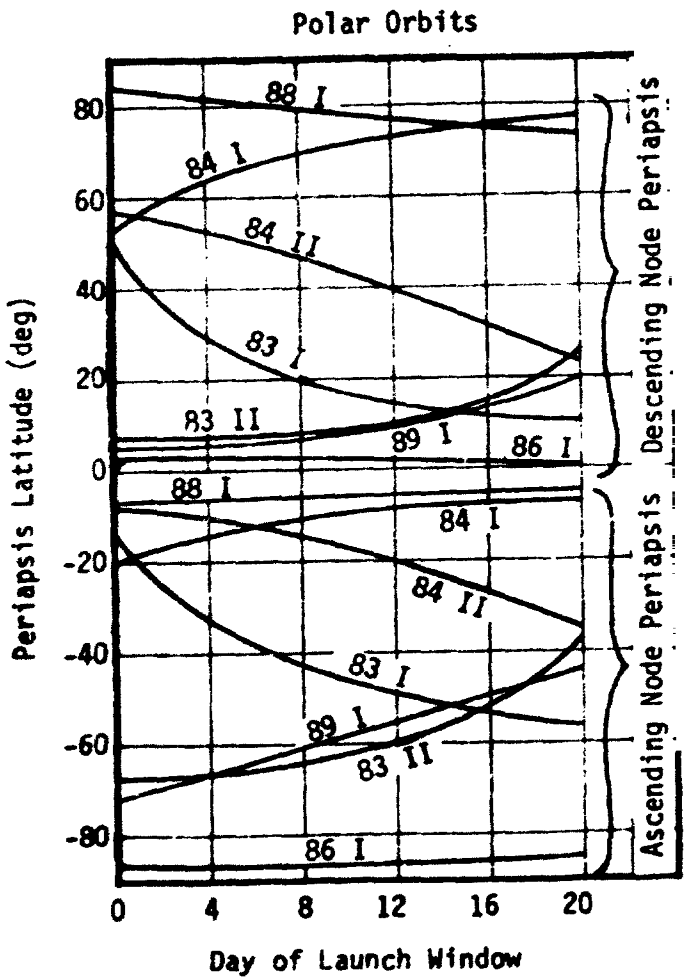

Figure 7 Nominal Periapsis Latitude Locations in Polar Orbit

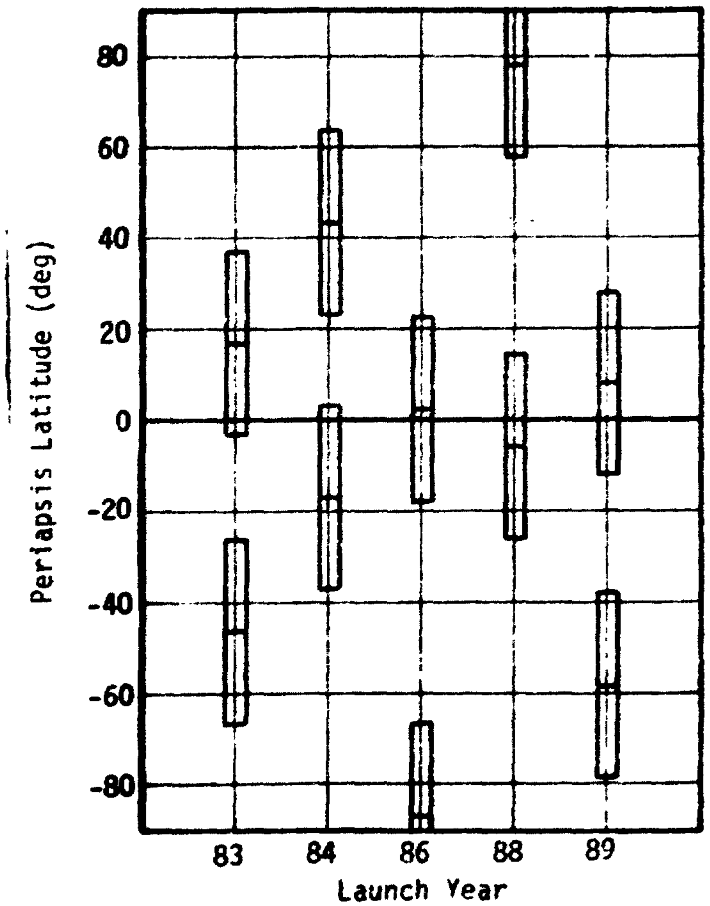

Figure 8 Periapsis Latitude Range in Polar Orbit With $\pm$ 20° Apsidal Shift

Another product of this construct is establishment of a criterion based on insertion propulsion limits to apsidal shift, for the selection of the proper orbital motion at periapsis, either ascending or descending. To illustrate, if periapsis were designed for an equatorial location in 1984 (Type I), the preferred motion at periapsis would be ascending, since this would nominally bring periapsis within 8° to 20° of the equator. Descending orbit motion would place periapsis at latitudes ranging from 50° to 75° north, requiring an unacceptably large apsidal shift to relocate periapsis. If the nominal design were indeed for an equatorial periapsis, missions in 1984 and 1988 would prefer ascending motion at periapsis, while descending motion would be appropriate in 1983, 1986, and 1989. Of course for periapsis locations at other specific latitudes, each mission year would have to be assessed individually with the aid of Figure 7 to determine the most efficient mode to minimize apsidal shift. It is assumed that radar mapping can be gained equally well from either orbital motion.

For much of the mission analysis, specifically for the assessment of performance characteristics, a nominal shift capability of 20° has been provided. This has been chosen as a mean value which allows access to most of the desirable latitude regions for most of the mission opportunities, provides the possibility of an equatorial periapsis for all years, yet is not an unmanageable shift from an orbit insertion view. The amount of this shift actually yields a $\pm$ 20°, or 40° spread in latitude around the nominal periapsis location for polar orbits, so where periapsis is not constrained to a particular hemisphere, a high degree of flexibility usually exists, as best illustrated by Figure 8. Exceptions do occur for the 1986 and 1988 opportunities, where the double-valued periapsis solution falls near a pole.

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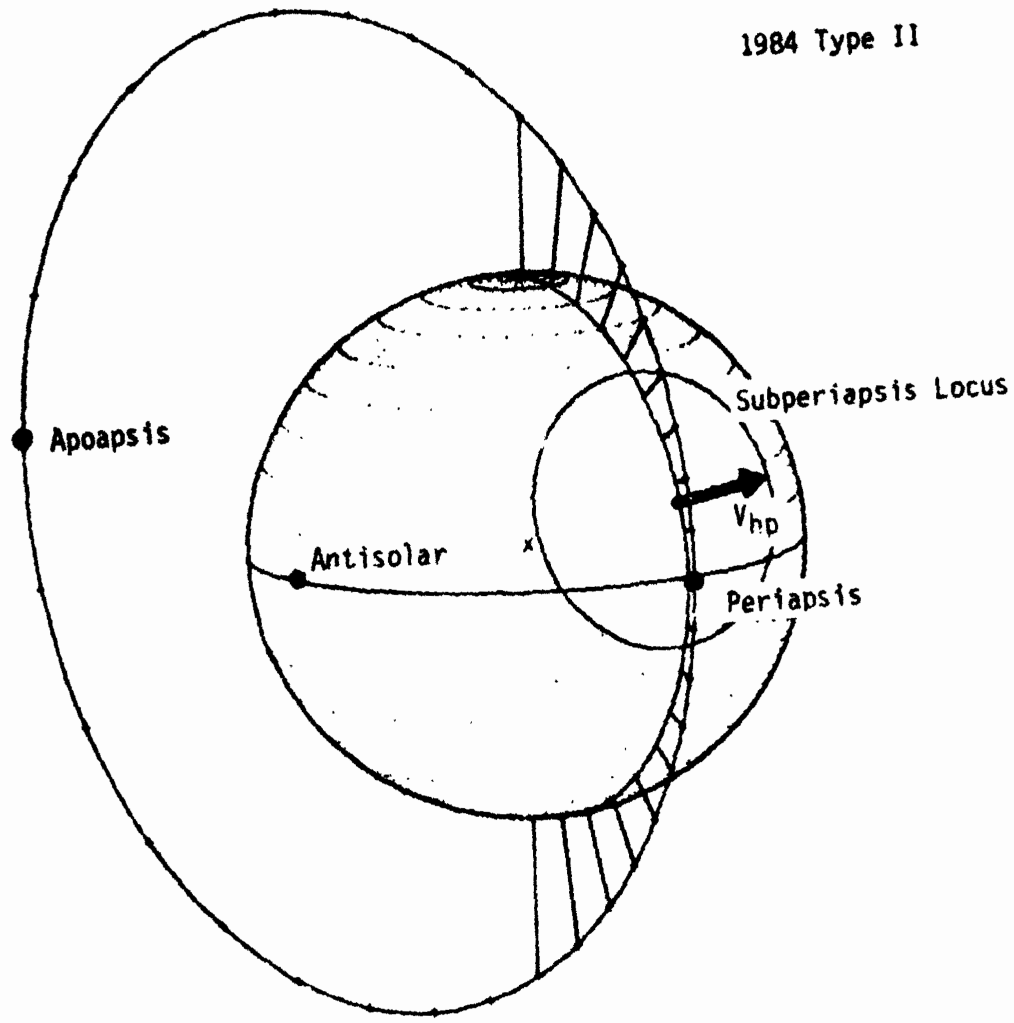

Figure 9 Typical Orbit Geometry at Arrival

and near the equator. For these cases an apsidal shift of $\pm 20^{\circ}$ leaves inaccessible the regions at mid-latitudes.

In summary, this analysis of arrival characteristics as illustrated by Figure 7 for polar orbits, serves to indicate those areas nominally accessible for each mission year, and the amount of apsidal shift needed to locate periapsis somewhere other than its nominal latitude. A typical orientation for a polar mapping orbit with .5 eccentricity is shown in Figure 9 on the preceding page. In this case periapsis has been shifted off the subperiapsis locus for placement over the equator. Mapping is depicted from pole-to-pole, or $\pm 90^{\circ}$ in true anomaly.

ORBIT INSERTION ANALYSIS

As indicated in the study of performance characteristics for the various mission opportunities, the initial evaluation of performance has treated the capabilities for inserting a "rubber" payload into orbit. The analysis also considered a simplified insertion picture to set a preliminary foundation for a more complete assessment of each opportunity in terms of the specific requirements on orbit insertion propulsion. Study of orbit orientation possibilities for these missions has indicated that provision be made for ● 20° apsidal shift during the insertion maneuver, arising from the desirability of relocating periapsis for eccentric orbits to some specified latitude. In addition, the Vhp range associated with the mission set yields rather long burn times and arcs during insertion for the considered propulsion system, and suggests that finite burn loss effects will not be insignificant.

The early analysis of orbit insertion considered the effects of transfers to orbits of varying eccentricity, where propellant load, delivered payload, and insertion ΔV were the principal dependent parameters. A pre-insertion weight of 4000 kg was assumed as nominal, based on the Titan IIIE/Centaur capability to deliver that weight to Venus for most mission years. Burns were considered to be at fixed attitude. Orbit eccentricity and Vhp magnitude were the independent variables. These assumptions led to a somewhat pessimistic evaluation of finite burn loss; treating a fixed attitude insertion is conservative, and the possibility of off-loading the spacecraft weight to gain a fixed payload in orbit is not considered. Given the thrust level typical of the 2-engine Viking insertion propulsion initially examined (600 lbf, or 2669 N), the burn times and arcs for insertion into even a relatively low energy orbit of .5 eccentricity were found to result in unacceptable losses. The ΔV penalties assessed orbit insertion ranged between 400 and 700 m/sec for the considered Vhp range, with losses of far greater magnitude for the less eccentric orbits. The dashed line of Figure 10 presents the losses, including apsidal shift, with two engines for .5 eccentricity.

To counter this high level of finite burn loss a 4-engine system was evaluated, with net thrust level doubled, and predictably, losses were reduced to more reasonable levels. In Figure 10 the family of solid curves presents the burn loss/apsidal shift requirements for the 4-engine configuration, for varying eccentricity and Vhp. Systems studies at this point indicated that from an operational view growth to a 3-engine Viking system would be far superior to anything larger. Losses associated with that system were then examined and found to be nearer the 4-engine loss figures

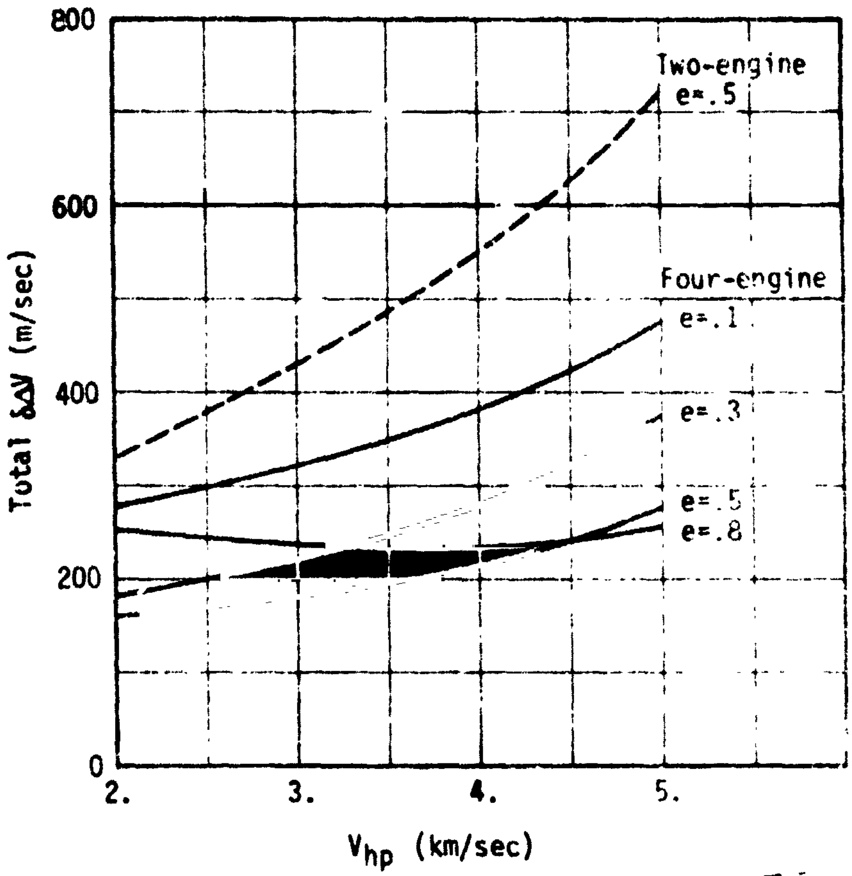

Figure 10 Combined Finite Burn Loss and Apsidal Shift Penalty For 2 and 4 Engine Viking

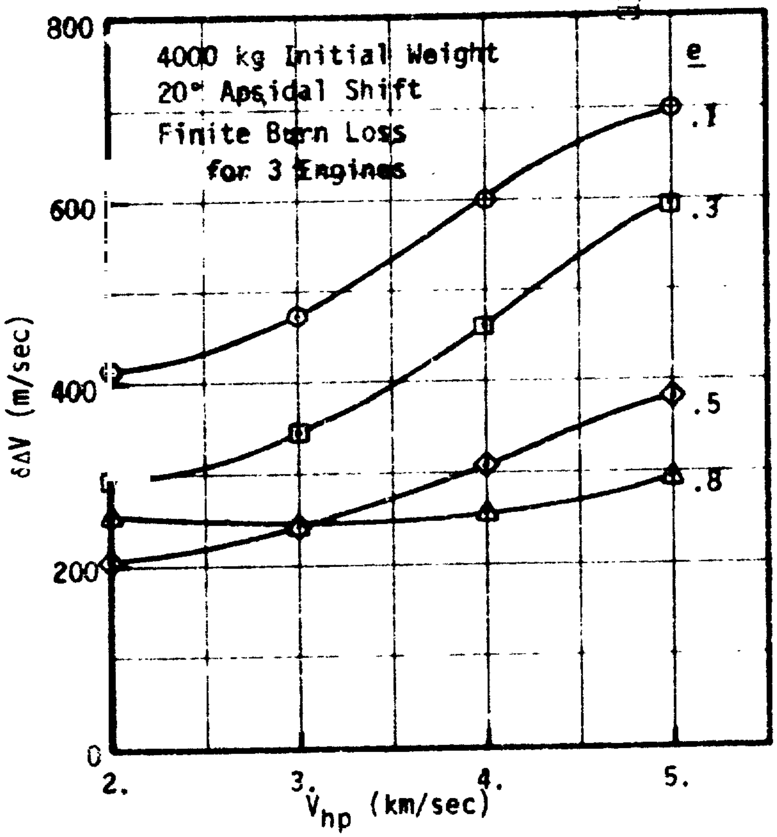

Figure 11 3-Engine Viking Losses Against Vhp (Includes 20° Apsidal Shift)

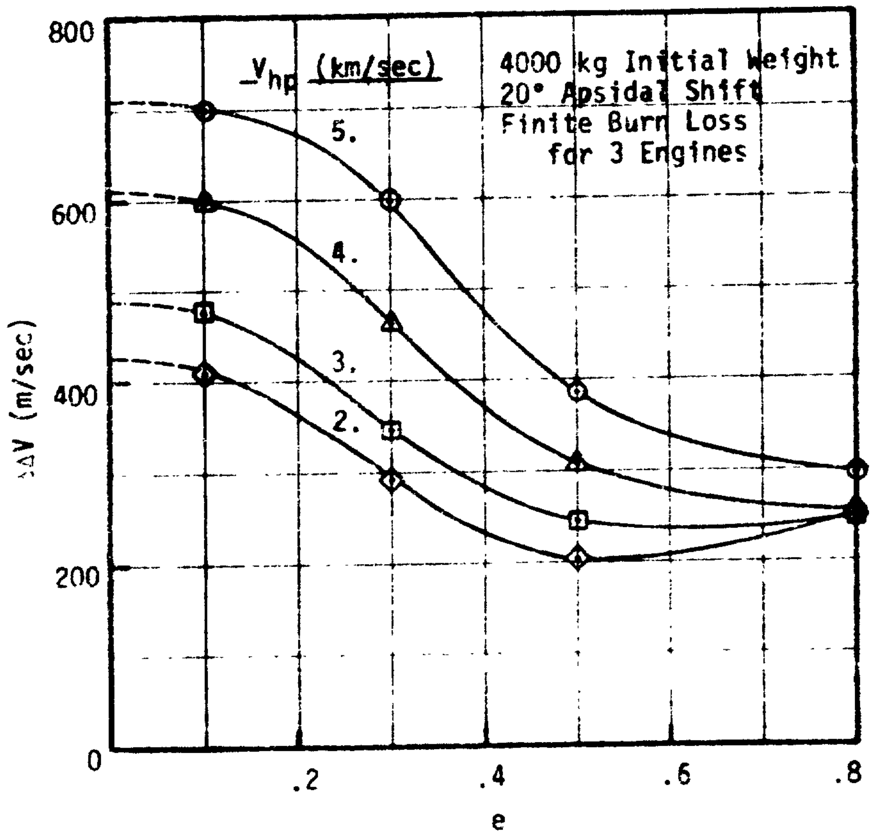

Figure 12 3-Engine Viking Losses Against Eccentricity (Includes 20° Apsidal Shift) than they were to the 2-engine losses, and still of a manageable nature. Loss levels for the 3-engine system are illustrated in Figure 11 against Vhp magnitude, and in Figure 12 against eccentricity. All curves include allowance for the 20° apsidal shift. These ΔV adjustments underlie the modification to mission performance capability presented earlier in Figures 4 and 5.

Where the intent is instead that of assessing the quantitative nature of finite burn loss for some fixed design payload weight, for various propulsion thrust levels considering different orbit sizes and mission year performance characteristics, the analysis approach must be modified from the treatment described above. A convenient parameter for the generalized measurement of finite burn loss is the pre-insertion thrust to weight ratio (T/W). With T/W as the principal independent parameter, a family of curves has been developed, Figure 13, which relates the amount of burn loss to the impulsive ΔV requirement. This construct then allows each mission opportunity to be evaluated in terms of defining adequate thrust levels, given variations in Vhp and orbit eccentricity which can be directly translated to impulsive ΔV. In Figure 14 impulsive ΔV requirements are shown for each opportunity, adjusted for a 20° apsidal shift, and varying with orbit eccentricity. Here the highest Vhp magnitude of each launch window has been selected for worst case analysis.

In Figure 13 curves are included for impulsive ΔV levels of 1800, 2400, 3000, and 3800 m/sec, which roughly correspond to the range expected for the considered Vhp magnitudes and orbit eccentricities. The dashed lines portray burn loss as a constant percentage of impulsive ΔV. To establish a criterion for determining the thrust level (number of Viking engines) adequate for

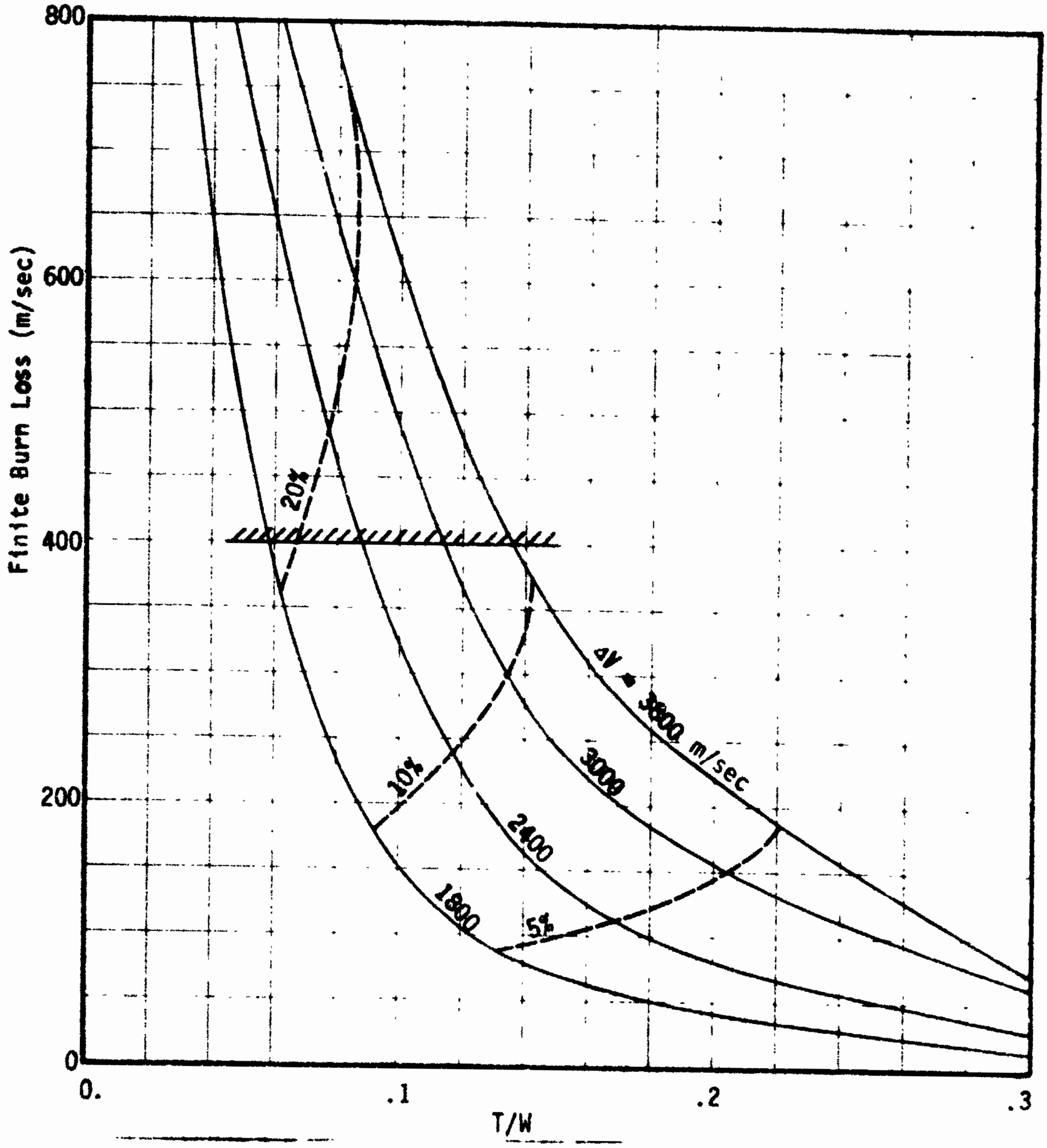

Figure 13 Finite Burn Loss for Varying Initial Thrust to Weight

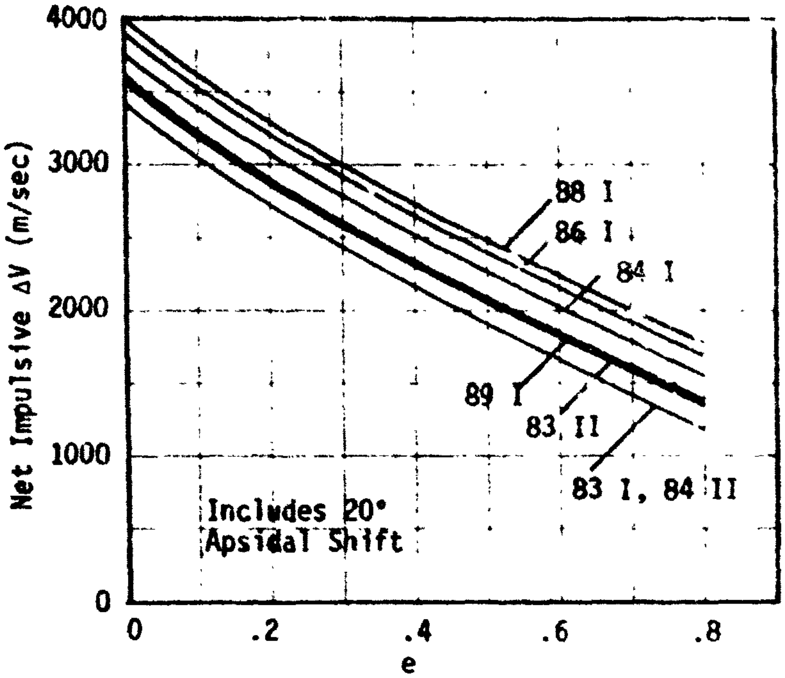

Figure 14 Impulsive Delta-V Requirement for All Missions

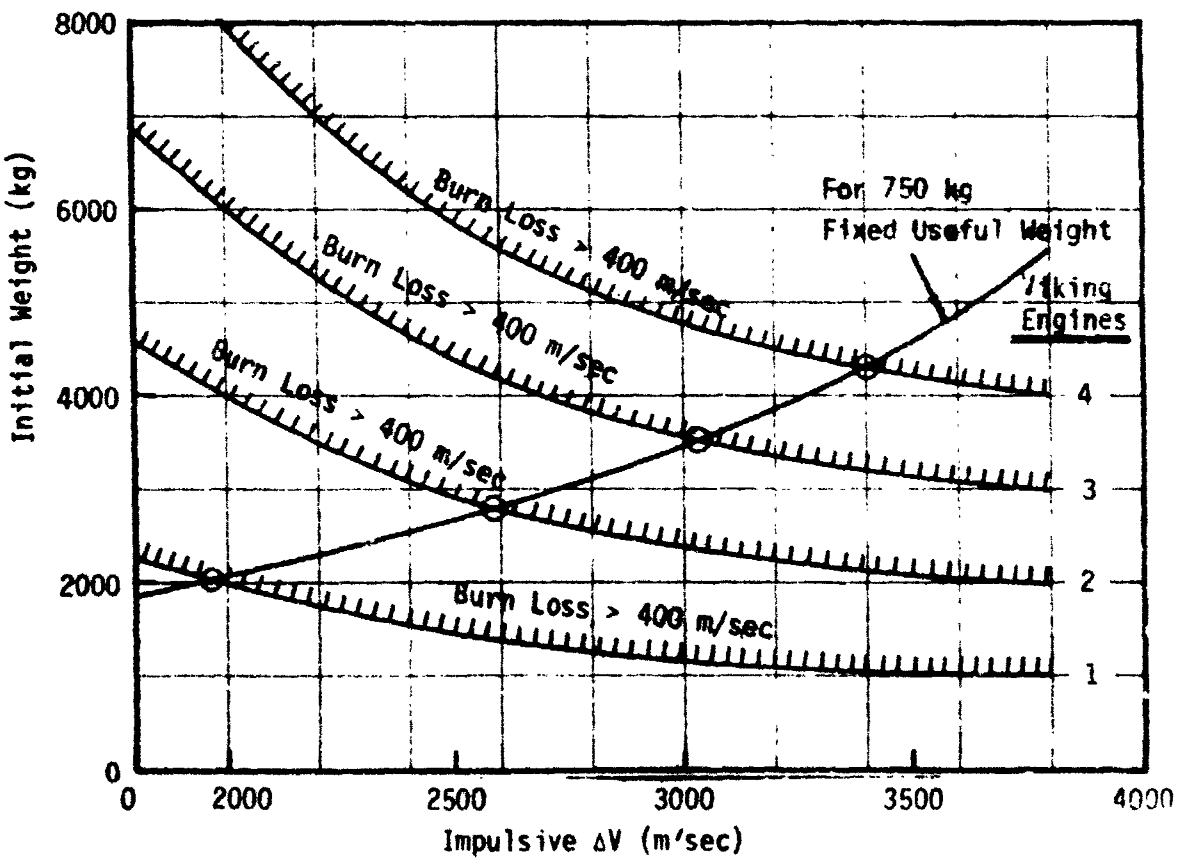

Figure 15 Pre-insertion Weight Limits for Varying Thrust Levels With Finite Burn Loss Less Than 400 m/sec

a given ΔV requirement and initial (pre-insertion) weight, an arbitrary limit on allowable burn loss magnitude has been assumed. A fixed magnitude limit in this situation is probably a more reasonable constraint than a fixed percentage of impulsive ΔV, since for a fixed percentage limit higher losses would be allowed where higher basic ΔV levels already exist. On this premise, a burn loss limit of 400 m/sec has been considered, as shown by the shaded boundary of Figure 13.

With this limit a minimum T/W can be defined for the various impulsive ΔV levels, and as thrust is equated to the 1, 2, 3, and 4-engine combinations, maximum initial weight limits are established for each system. This picture is then illustrated by Figure 15, where the propulsion system design "switchover" points are related to ΔV and initial weight. On the same plot can be superimposed a curve which represents the initial weight required to achieve a fixed, desired weight in orbit, also varying with ΔV. Based on detailed systems' assessments of weight in orbit requirements, a nominal payload of 750 kg has been considered adequate for these missions. This analysis leads to the definition of the maximum initial weight which will achieve the desired payload in orbit yet hold finite burn loss under 400 m/sec, appropriate to each engine combination and various ΔV requirements. The critical points are circled in Figure 15.

From the philosophy outlined above, all mission years were then re-examined, not on the basis of defining payload capability, but instead with the view of defining required insertion propulsion characteristics for gaining the 750 kg desired payload. Figure 16 therefore represents an alternative picture of performance for the Venus orbital radar missions. Naturally, in this treatment which emphasizes orbit insertion, Vhp magnitude

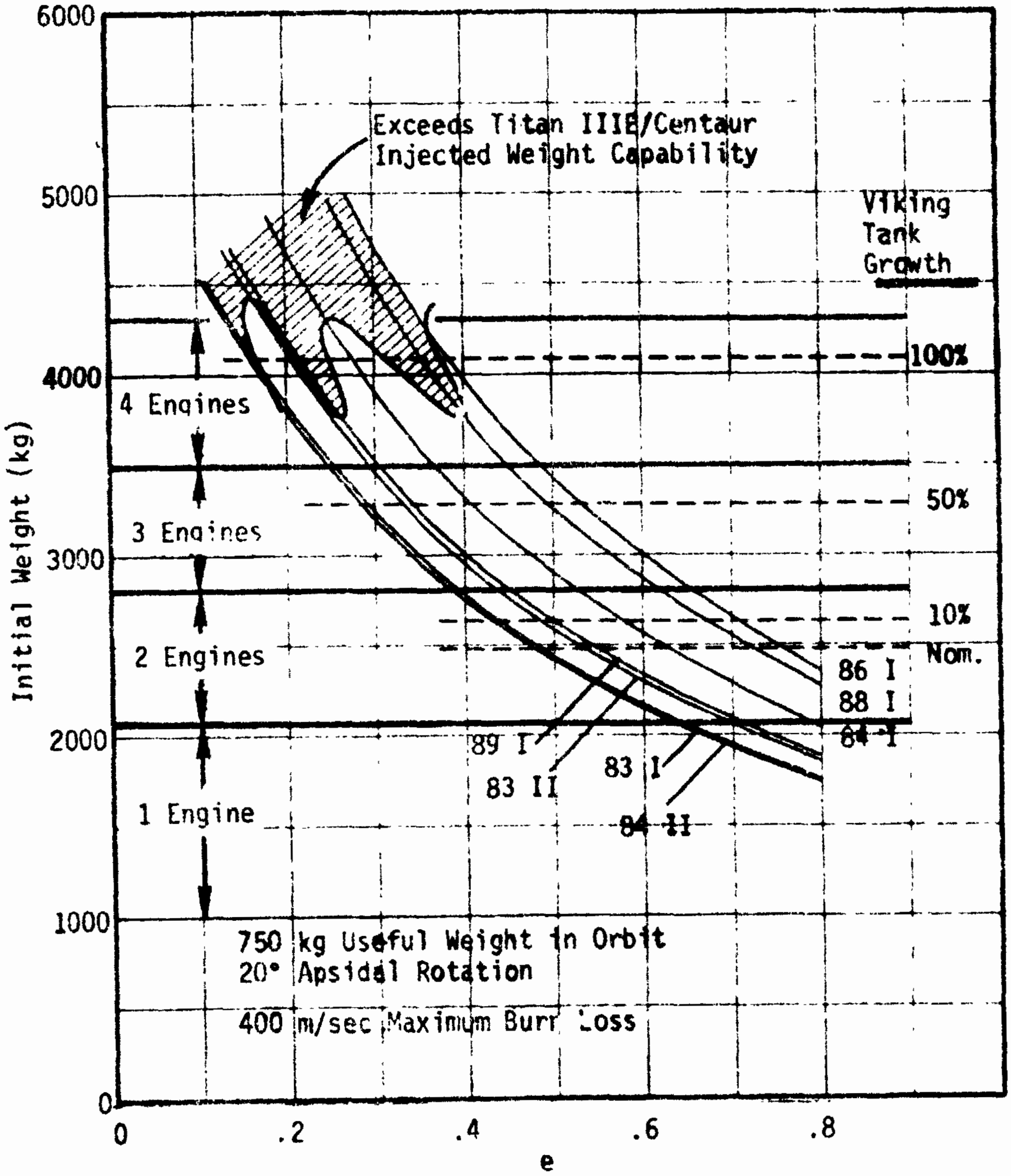

Figure 16 Pre-insertion Weight Requirements for All Missions With Associated Propulsion System Characteristics

assumes a greater importance in the evaluation of mission years. The curves of Figure 16 illustrate pre-insertion weight requirements for each opportunity and are functions of arrival Vhp and the desired orbit eccentricity, for the assumed 750 kg payload in orbit. These initial weight requirements in turn define the required propulsion system. In this profile the engine combination "switchover" points are indicated, along with propellant load expressed as a percentage of nominal Viking propellant tank capacity. Propellant load is directly related to the required initial weight when payload weight is fixed, and is also a function of the propellant inerts assumed for the propulsion system. The cross-hatched region of the plot indicates limits to pre-insertion weight which derive from the capability of Titan IIIE/Centaur, based on the launch energy (C_3) associated with each mission.

Inspection of Figure 16 shows that for an orbit of .5 eccentricity, all mission years can achieve the desired inserted payload with an insertion propulsion system of no more than 3 engines, and with propellant requirements ranging from a nominal Viking tankage situation in 1983 and 1984 to a 60% growth in 1986. With an eccentricity of .3, missions in 1986 and 1988 would be unattainable from launch energy limits, while the 1984 Type I mission would need 4 engines. For orbit eccentricities below .2, nearly all years would require a higher performance insertion system and much larger propellant allocations with Titan IIIE/Centaur as the launch vehicle.

Current indications point to the 3-engine configuration as an operationally workable Viking growth alternative, and although a 60% growth in the nominal Viking tankage is not insignificant, the primary conclusion to be drawn from this analysis is the attractiveness of the .5 orbit eccentricity for the reference mapping orbit design.

ORBIT STABILITY

Concurrently with the analysis of orbit insertion requirements, which suggested that a mapping orbit size of .5 eccentricity would provide the flexibility of a viable mission for each opportunity in the 1980's, the stability characteristics of eccentric, polar orbits were examined. Two preliminary hypotheses involving these characteristics were tested prior to a general parametric evaluation of stability against orbit eccentricity and periapsis latitude.

First, from the similarities in celestial geometry at encounter for all missions observed earlier, the sensitivity of the stability picture to mission opportunity was assessed and found to be minimal. Secondly, the effect of considering orbit inclinations near, but not exactly, polar (90°), was addressed. Since a side-looking radar would require some off-polar inclination bias to view the true pole of the planet, inclinations of up to $\pm 15^\circ$ off true polar were examined to determine whether their stability characteristics were sufficiently close to those of 90° inclination orbits to allow the simplifying polar orbit assumption to be made for the parametrics. Differences in periapsis altitude variation over a 243 day mission lifetime were found to be on the order of only 5 km for orbits of .5 eccentricity, considering inclinations of $90^\circ \pm 15^\circ$, and the polar orbit assumption has therefore been assumed valid for the following analysis.

Other assumptions relevant to this study involve the influence of various perturbative terms on the life of a Venus orbiter. Gravity harmonics which can be estimated for Venus based on the little knowledge available, such as the J2 term from oblateness, appear to be insignificant. Higher order harmonics remain unknown. The existence of mascons would only be speculative, certainly unmodelable. Given this situation and the proximity of Venus to the sun, the dominant influence on orbit stability has been assumed to be from solar perturbations on the orbit.

The type of instability that can exist for a polar orbit is illustrated most clearly by Figure 17. This represents a rather extreme case, for a high eccentricity of .8, but shows the crucial influence of periapsis location in latitude. This case corresponds to periapsis near the orbit ascending node. If periapsis is located on or very near the planet equator, variation in periapsis altitude is minimal over the life of the mission. As periapsis is located much off the equator, however, the altitude variation becomes significant. A location at only 20° S causes the initial periapsis altitude of 400 km to grow to 800 km in 250 days, while a location at 20° N results in entry into the atmosphere only 160 days after insertion. Locations further removed from the equator tend to aggravate the variation, although the amount of the increased variation is not as large at the higher latitudes, as seen by the differences between the 40° and 20° curves of Figure 17. If periapsis were near the descending node of the orbit, the curves would reverse for positive and negative latitudes. That is, with ascending node motion about periapsis, a northern periapsis causes altitude decay. With descending motion about periapsis, the same northern location would cause periapsis growth at about the same rate. To summarize this effect, periapsis altitude in polar orbit decays for a northern periapsis when nearest the ascending node, and for a southern periapsis when nearest the descending node. To avoid altitude decay then, ascending node missions should seek a southern periapsis, and descending node missions a northern periapsis. Referring to

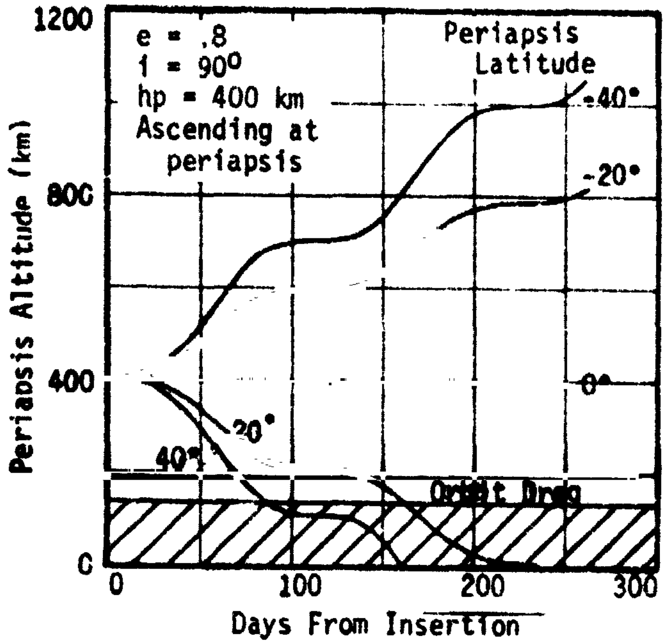

Figure 17 Periapsis Altitude History in Polar Orbit for .8 Eccentricity and Varying Periapsis Latitude

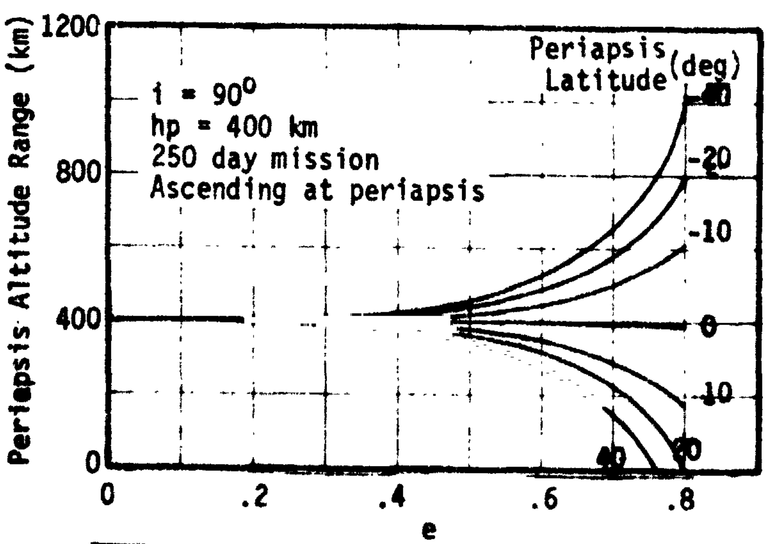

Figure 18 Periapsis Altitude History in Polar Orbit for Varying Eccentricity and Periapsis Latitude

the earlier discussion of arrival orbit orientations, this desired situation arises naturally from the characteristics of nominal transfer for all the considered missions.

Of primary interest to the mission parametric trades is the effect of eccentricity on this picture of polar orbit stability. Figure 17 illustrated an extreme situation for high eccentricity. In Figure 18 the maximum variation in periapsis altitude is shown varying with eccentricity, with periapsis latitude a field parameter, again considering ascending motion about periapsis. From this view, orbit stability becomes much less of a problem where eccentricity is kept under about .6. At an eccentricity of .5 the variation in altitude is no more than 50 to 60 km even with periapsis at $\pm 40^{\circ}$, and this design would require two orbit trims of 5 m/sec each to hold within a $\pm$ 10 km deadband around the nominal 400 km altitude. For eccentricities of .4 or less the variation becomes negligible.

To summarize, an eccentricity of .6 for a polar orbit mission represents a probable upper limit for manageable orbit stability characteristics, if off-equatorial periapsis locations are to be considered. Lesser eccentricities yield increasingly more stable orbits, while eccentricities above .6 tend to become unstable exponentially. At these higher eccentricities the location of periapsis off the equator determines the degree of instability.

OCCULTATIONS

Occultation histories in polar orbit have been generated to facilitate system design trades which are closely related to sun and Earth shadow times. Figure 19 illustrates characteristic occultations for an orbit of .5 eccen-

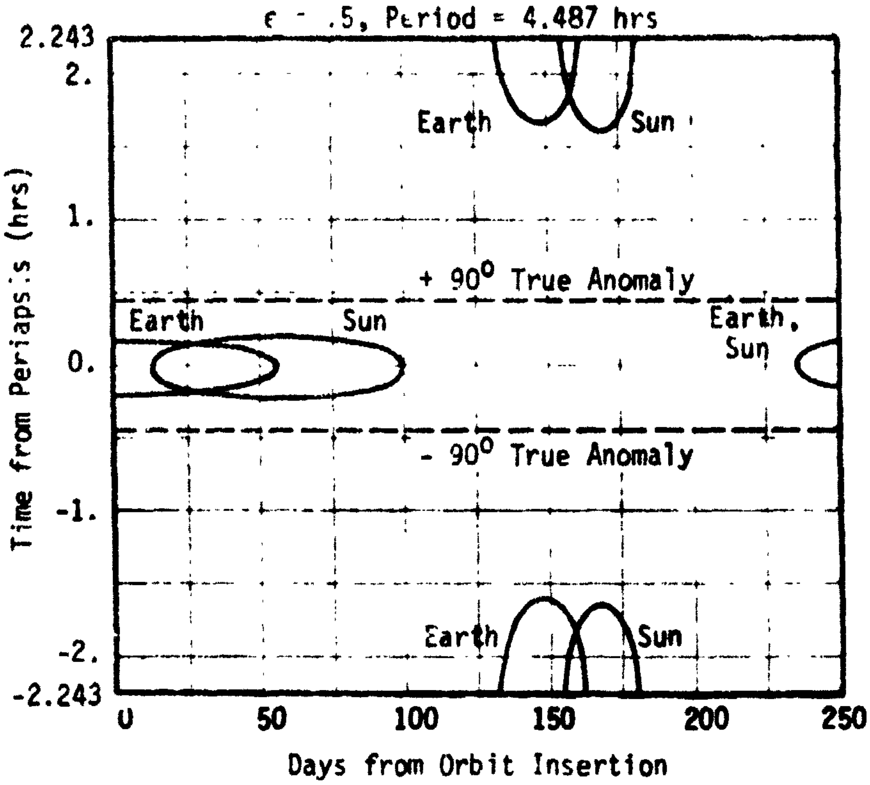

Figure 19 Typical Occultation History in Polar Orbit (1984 I1)

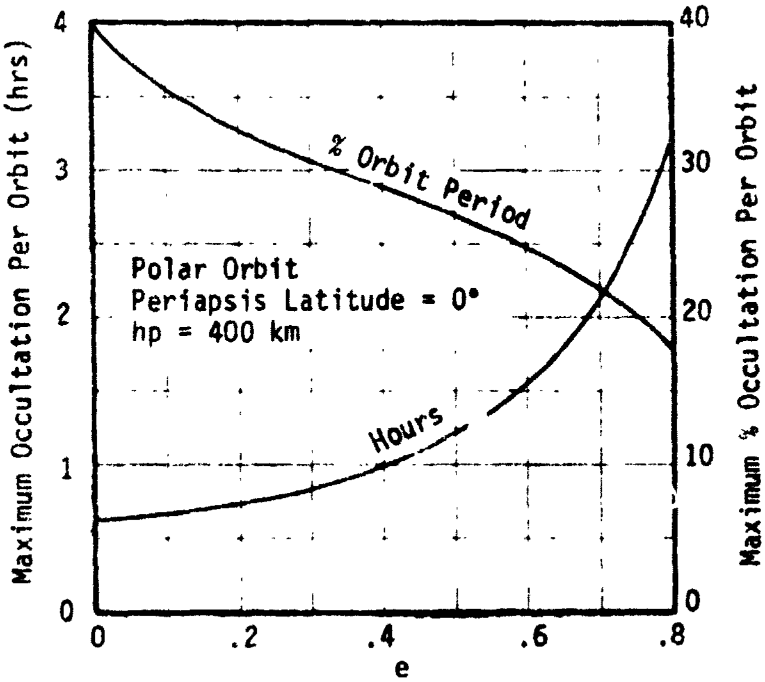

Figure 20 Maximum Occultation Time Versus Eccentricity

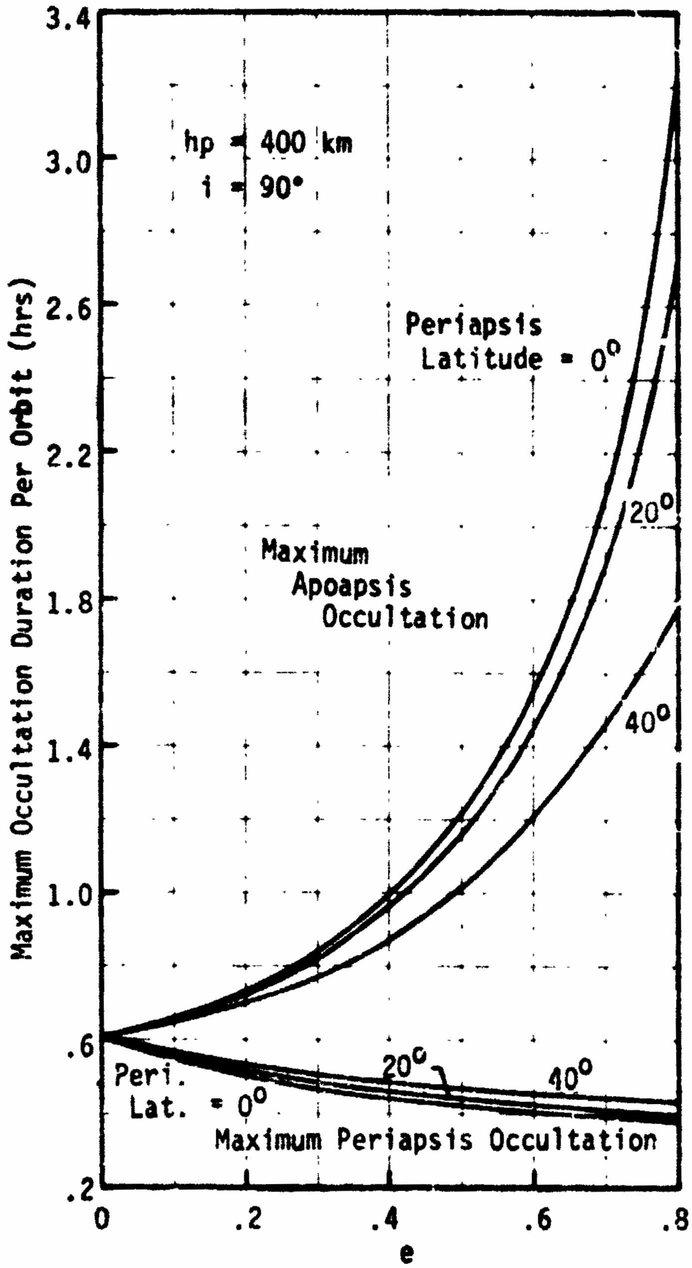

Figure 21 Maximum Occultation Times for Varying Periapsis Latitude and Eccentricity

tricity, 90° inclination, with periapsis located at the equator and at an altitude of 400 km. The particular mission opportunity for this profile is a 1984 Type II. As indicated by the examination of encounter geometry, orbit insertion occurs out of Earth view. Soon after insertion periapsis begins a period of occultation from the sun. Midway into the orbiter mission apoapsis passes through Earth shadow, and 30 days later, through the sun shadow. This period represents maximum per orbit occultations with which the spacecraft must contend. The duration of these maximum occultations is presented in Figure 20 for varying eccentricity, shown as total hours occulted per orbit and percent of orbit period. As indicated by the two curves, although the occultation time per orbit increases with increased eccentricity, the percent of the orbit period spent in occultation actually decreases, representing a net decrease in total occultation time over the mission duration.

Figure 21 then illustrates how the occultation picture in polar orbit varies when periapsis is located off the equator. The effect of off-equatorial periapsis locations is that of increasing the duration of near-periapsis occultation while decreasing the duration of near-apoapsis occultation. In the figure maximum periapsis and apoapsis occultation times are shown varying with both eccentricity and periapsis latitude. The more significant effect is from eccentricity, but periapsis latitude becomes significant at the higher eccentricities.

As suggested by encounter geometry similarity, the occultations were found to be relatively independent of mission opportunity. Over the set of all opportunities in the 1980's, the timing of the overall occultation pattern might shift by $\pm$ 30 days with respect to orbit insertion, but the relationship of particular occultations with one another remains unchanged. In all cases, the 243 day mission duration necessitates passage through both periods of maximum periapsis and apoapsis occultation.

CONCLUSIONS FOR ORBIT DESIGN

From the various elements shaping orbit design for a Venus orbital radar mapper which have so far been treated in this study, conclusions can be drawn at this point for the reference design. Orbit inclinations near polar are necessary for area mapping of all latitudes, and nothing prohibitive of that design has been disclosed. For eccentric mapping orbits, nominal periapsis location in latitude has much flexibility when considering the set of all mission opportunities for the 1980's. The provision of a $\pm 20^\circ$ apsidal shift during orbit insertion further enhances periapsis latitude possibilities, and allows positioning periapsis over the equator in all mission years. When mapping from an eccentric orbit, an equatorial periapsis is desirable to achieve a complete surface map balancing coverage over both hemispheres. Orbital motion at periapsis, ascending or descending, is determined principally by the specific latitude, or hemisphere, desired for periapsis, so becomes a function of radar mapping strategy, required apsidal shift, and is mission year dependent. To gain access to all longitudes, the duration of the orbiter mission must be 243 days mapping pole-to-pole in an eccentric orbit, or 122 days mapping a full 360° from a circular orbit.

The size of the orbit, defined by eccentricity, has become the most crucial design trade of the mission study. Circular orbits would afford the simplest mission for the radar systems, while placing very heavy demands on the launch and orbit insertion propulsion, and hence imply considerable dependence on new technology development to achieve adequate performance for all considered years. Design of an eccentric orbit conforms more to the philosophy of establishing a mission of attractive cost, mission feasibility for all available launch opportunities, and full implementation of state of the art technology. This would however require for the radar system the development of some technique which allows adequate mapping from a varying altitude and range history, so that the potential for a complete surface map from $\pm 90^{\circ}$ in latitude could be preserved. A radar mapping technique which does resolve the problems of variable range, increased power, and signal ambiguity has been defined by the systems' analyses of the Martin study (Ref 3), and its characteristics will be discussed in the following section.

Given a resolution of the eccentric orbit mapping problems associated with an orbital radar, the mission analysis has converged on a preferred orbit size. As addressed in the evaluation of mission performance and orbit insertion requirements, an eccentricity of .5 with a 400 km periapsis altitude provides mission feasibility for all launch years of the 1980's. Launch can be achieved with Titan IIIE/Centaur, and orbit insertion with a slightly modified Viking propulsion system of 3 engines and somewhat enlarged propellant tankage. Eccentricities below .5 would cause the loss of some mission years, given the proposed launch/insertion systems. An upper bound to orbit size derives from the assessment of orbit stability. For eccentricities of .6 or greater, periapsis altitude variation in polar orbit can become unmanageable when coupled with periapsis locations at latitudes much removed from the equator. Although an equatorial periapsis would nominally be prefe. ~i for the surface mapping, the possibility of an off-equatorial periapsis should be considered for mission evolution flexibility. For the orbit eccentricity of .5, with periapsis at $\pm 40^{\circ}$ latitude, variation in periapsis altitude over a 243 day mission life is about 50 km, and could be held to within $\pm$ 10 km of the desired 400 km with two orbit trims of 5 m/sec each.

In summary, the orbit design suggested as the best approach to gain area topographical mapping of Venus with radar imaging would be of polar to near-polar inclination, of .5 eccentricity with a 400 km periapsis ideally located near the planet equator, and of either ascending or descending motion about periapsis. This implies the existence of a mapping strategy which gains coverage over $\pm 90^{\circ}$ in latitude during each mapping pass, and thus requires an orbiter life of at least 243 days to achieve full coverage in longitude. This design makes realizable mission opportunities for all considered years.

RADAR DESIGN AND SURFACE COVERAGE CHARACTERISTICS

The problems of radar imaging from a varying range to the surface have been brought under control by use of a variable side-look program for the radar beam during each mapping pass. This technique involves allowing side-look angle, measured out of the orbit plane from the radius vector to the radar beam, to vary between 50° at periapsis and 12° at $\pm 90^{\circ}$ true anomaly, for an orbit of .5 eccentricity. By using a variable beam offset from radial, rather than a fixed offset, the range increase with true anomaly and spacecraft altitude is reduced, and the width of the mapping swath is held nearly constant. The direct result of this strategy translates into reduced radar power requirements, reduced signal ambiguity problems, and a wider altitude range over which radar mapping can be effected. In fact, this particular SLA profile defined for the orbit of .5 eccentricity does allow mapping $\pm$ 90° true anomaly, and thus achieves the desired pole-to-pole mapping potential.

With pole-to-pole mapping, the preferred periapsis location in latitude is near the equator, with coverage balanced over both hemispheres. For the suggested design of .5 eccentricity and 90° inclination, surface resolution gained by varying SLA between 50° and 12° becomes variable with true anomaly, such that 50 meter resolution is achieved over the equator, 100 meters at 60° latitude, and 200 meters over the polar regions. Typical radar power requirements are on the order of 333 watts for each mapping pass. The width of the radar beam has been sized at 2.13° to provide ample swath overlap from successive mapping passes. For the proposed orbit design and mapping strategy of imaging every orbit, the amount of that overlap ranges from a minimum of 30% at the equator to near 100% at the polar regions.

The ultimate result of both radar and mission design is the surface coverage actually realized by the proposed mapping strategy. Typical mapping swaths are illustrated in Figure 22 drawn on a global representation of the Venus surface. For clarity, individual swaths have been shown spaced at intervals corresponding to every 10th mapping pass. Mapping actually occurs during the $\pm$ 90° true anomaly range of each orbit. The orbit trace associated with swath "0" is indicated, with periapsis location denoted at the planet equator. North, in the Venus orbit plane system, is toward the top of the figure. As the planet's reverse spin progresses the inertial orbit from

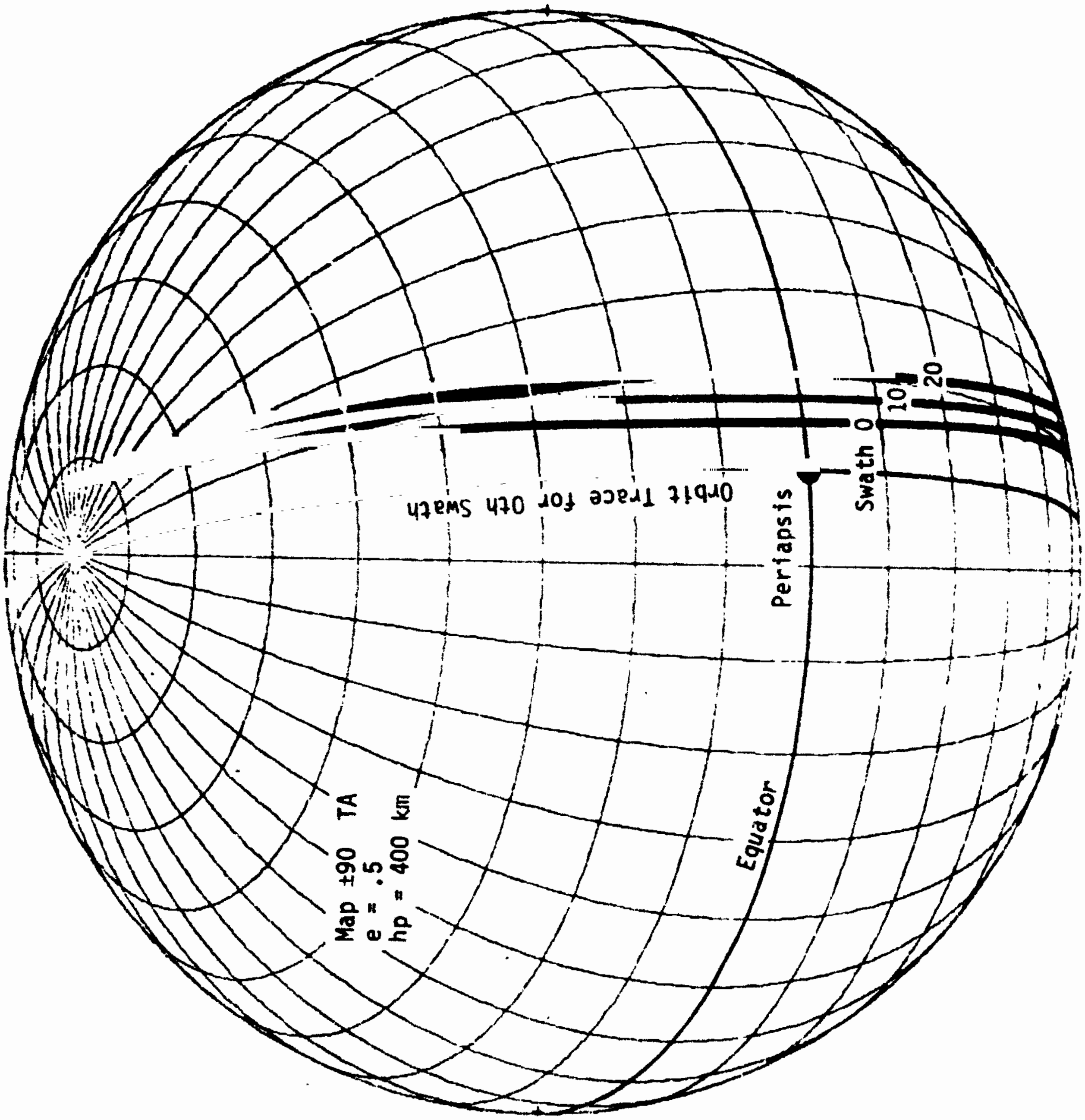

Figure 22 Typical Radar Mapping Swaths for the Reference Mission Design

west to east with respect to surface points, the swath advances in that direction with time. Here the orbit inclination is shown as true polar, so that coverage of the exact poles is not achieved. The amount of inclination offset to gain coverage of one planet pole would be $\pm$ 7.4° from 90°, given the programmed SLA history appropriate to this reference design.

NAVIGATION AND ORBIT DETERMINATION

The missions to Venus in the 1980's have been examined to determine navigation characteristics associated with their interplanetary cruise phases and to assess the accuracy with which orbit determination can define the resultant orbit about the planet. Midcourse ΔV requirements have been evaluated for two of the missions considered representative of the entire set -- a Type II trajectory in 1984 and a Type I trajectory in 1988. For each mission an Encke integrated trajectory, targeted to the desired arrival conditions, has been generated to allow the statistical definition of mid-course ΔV and trajectory dispersions. The analysis has considered a somewhat pessimistic assumption of navigation error levels existent in the 1980's. Neither Quasi-Very-Long-Baseline Interferometry tracking/processing, nor charged particle calibration capability, have been assumed. Hence, the equivalent station location errors are sized at 4.5 m for spin radius, 4.25m for z-height, and 5.0 m for longitude, all one-sigma values. Execution errors for the midcourse corrections correspond to .25% proportionality, 25 mm/sec resolution, and .5° pointing. Errors in the Venus ephemeris have also been considered, and assumed to be 20 km spherical in position.

For each mission two midcourse corrections are considered adequate.

The first is executed 5 days after launch to remove injection errors. A second is executed 10 days prior to Venus arrival to remove those trajectory errors which grow from the execution of the initial midcourse, and its effect is that of fine-tuning the trajectory to achieve the desired encounter conditions. Table 3 presents the resulting statistical ΔV magnitudes (mean + three-sigma) for both missions, along with resulting aimplane dispersions mapped to encounter.

Table 3 Navigation Analysis

Mission	ΔV (m/sec) M/C 1	M/C 2	Total	One-sigma Encounter Errors σRp(km)	$\sigma\omega$(deg)	σi(deg)
1984 II	**28.54**	**8.00**	**36.54**	**38.**	**.51**	**.18**
1988 I	26.53	3.47	30.00	112.	.42	.64

The analysis indicates that 1) midcourse ΔV requirements are about the size anticipated and of a manageable nature, 2) neither errors in periapsis location ($\sigma\omega$) or inclination (σi) are significant, and 3) dispersions in the radius of closest approach (σRp) may be larger than expected. Considering the 3-sigma radius error for 1988 yields a potential dispersion of 336 km, which would not be acceptable when targeting to a nominal altitude of 400 km.

Two techniques exist which can alleviate the problem of large radius error. The first involves incorporation of charged particle calibration into the navigation process. This would necessitate inclusion of both X-band and S-band capability onboard the spacecraft, but would reduce encounter errors by about 50%. A second alternative involves biasing the trajectory aimpoint at Venus sufficiently to ensure encounter radius falls beyond any point which might lead to rapid altitude decay. This technique implies insertion at an altitude greater than the desired 400 km for nominal orbit periapsis, and so requires some provision for a small orbit trim maneuver to be executed soon after insertion to bring periapsis back to the desired altitude. This technique has been successfully used for most of the Lunar Orbiter missions of the late 1960's and has proven effective.

The considered mapping orbits have been treated to a cursory evaluation to assess the degree to which nominal orbit determination can define position and velocity of the spacecraft in its orbit. Orbit sizes corresponding to eccentricities of 0., .3, and .5 have been considered, with periapsis altitude at 400 km and 90° inclination, and the in-orbit uncertainties associated with each after 24 hours of tracking are listed in Table 4. The results indicate that although the circular orbit does exhibit a somewhat higher level of uncertainty than the eccentric orbits, in σRAD and σVEL, the difference is minimal. One-sigma errors range from 17 to 25 m in radius for all orbits, with velocity uncertainty between 3 and 8 m/sec, as eccentricity varies from .5 to circular. None of the observed error magnitudes suggests any impairment of the radar mapping capability.

Table 4 Orbit Determination Uncertainty

e	σ RAD	σ POS(rms)	σ VEL(rms)
0.	25m	2.14km	8.18m/sec
.3	20	2.54	5.63
.5	17	1.60	2.81

CONCLUSION

The approach suggested by this study for area topographical mapping of Venus with an orbital radar involves gaining that map from a polar orbit of

.5 eccentricity. Periapsis is designed for an altitude of 400 km, located near the planet equator. This recommendation for eccentric orbit mapping rests on the desire to provide mission viability for all considered opportunities of the 1980's, holding to a reasonable program cost. For this design, launch can be achieved with Titan IIIE/Centaur, and orbit insertion with a modified Viking insertion propulsion system, in all mission years.

The power and signal ambiguity problems of radar imaging from a varying altitude and range are resolved by programming a variable side-look angle for the radar beam over each mapping pass. This technique reduces the range increase with true anomaly and achieves a nearly constant swept surface area Mapping occurs every orbit over a $\pm 90^{o}$ spread in true anomaly about periapsis, yielding a contiguous pattern of approximate pole-to-pole swaths. Provision for effecting a $\pm 20^{o}$ apsidal shift during orbit insertion is recommended to allow locating periapsis at the equator for all years. Duration of the orbiter mission must be at least 243 days. The mapping strategy so defined gains surface coverage approaching 100% of the planet area, with overlap between 30% and 100%, at resolutions ranging from 50 meters at the equator to 200 meters near the poles.

REFERENCES

1) Friedman, Louis, et al, "Venus Orbiting Imaging Radar," Final Report to NASA Hq., 27 April 1973, Jet Propulsion Lab.

2) Miller, Barry, "Side Looking Radar Plays a Key Role," Aviation Week and Space Technology, pp 44-46, 17 July 1972.

3) Scofield, W.T., Cross, D.B., et al, "A Study of an Orbital Radar Mapping Mission to Venus," Final Report Contract NAS2-7204 for the Ames Research Center, NASA, September 1973, Martin Marietta Corp.

4) Brown, W.E., et al, "Planetary Imaging Radar Study," JPL No. 701-145, 1 June 1972, Jet propulsion Lab.

5) MacKay, J.S., et al, "A Preliminary Analysis of a Radar-Mapping Mission to Venus," 11 October 1972, Ames Research Center.