AN N-BODY LOW-THRUST TRAJECTORY

PARAMETER OPTIMIZATION PROGRAM

Samuel Pines

Henry Wolf

Jerry Horsewood

Leon Lefton

Conrad Hipkins

Report No. 71 - 11
Final Report, Vol. III
Contract NAS 5-11193

March 1971

ANALYTICAL MECHANICS ASSOCIATES, INC.
9430 LANHAM SEVERN ROAD
SEABROOK, MARYLAND 20801

PRECEDING PAGE BLANK NOT FILMED

FOREWORD

This report describes work performed under Contract NAS 5-11193 for the NASA Goddard Space Flight Center. It consists of two parts. Part I presents the analytical development and Part II describes the ASTOP computer program.

PRECEDING PAGE BLANK NOT FILMED

ABSTRACT

This report describes the modifications to the basic ITEM interplanetary trajectory program which are necessary to provide a capability of generating optimum low-thrust trajectories in an N-body field while satisfying selected hardware and operational constraints. The suggested approach consists of dividing the trajectory into a number of segments or arcs over which the control is held constant. This constant control over each arc is optimized using a parameter optimization scheme based on gradient techniques. As described in this report, the program provides a wide range of constraint, end conditions, and performance index options. An equally important feature is that the basic approach is conducive to future expansion of features such as the incorporation of new constraints and the addition of new end conditions.

PRECEDING PAGE BLANK NOT FILMED

TABLE OF CONTENTS

PART I

ARBITRARY SPACE TRAJECTORY OPTIMIZATION PROGRAM

(ASTOP)

PROBLEM FORMULATION

NOMENCLATURE

Symbol	Definition
a	Area of the solar panels.
A	Transformation matrix.
$\bar{a}, \bar{b}, \bar{c}$	Arbitrary unit vectors.
A_g	Spacecraft acceleration due to gravitational attraction of the planets and sun.
a_i	Coefficients in the polynomial expression for power ratio as a function of solar distance.
A_p	Spacecraft acceleration due to thrust of electric propulsion system.
A_{sr}	Spacecraft acceleration due to solar radiation pressure.
a_T	Semi-major axis of the planetocentric trajectory at the target.
a_1, a_2, a_3	Coefficients in the equation for initial spacecraft mass.
b	Coefficient in the equation for propulsion system efficiency.
B	Matrix representing the outer product of a vector with itself.
b_1, b_2, b_3	Inertial components of a unit vector, $\bar{b}$.
c	Jet exhaust speed of the electric propulsion system.
C	Matrix operator representing the cross product of a unit vector $\bar{b}$ with an arbitrary vector X; i.e., $CX = \bar{b} \times X$.
c_a	Fraction of the total number of photons incident on the spacecraft that are absorbed.
c_r	Jet exhaust speed of the retro stage.
c_{3_b}	Energy of the hyperbolic escape trajectory at launch on the ballistic trajectory used to obtain estimates of the initial position and velocity.

Symbol	Definition
d	Constant appearing in the equation for propulsion system efficiency.
e	Eccentricity of the heliocentric spacecraft trajectory.
e_b	Eccentricity of the ballistic escape trajectory.
e_p	Eccentricity of the reduced energy escape trajectory.
$\bar{e}_r$	Unit vector along the instantaneous heliocentric spacecraft position vector.
e_T	Eccentricity of final orbit about target planet.
$\bar{e}_T$	Unit vector in the direction of thrust.
f	Represents final conditions.
i	Inclination of the launch parking orbit relative to the equatorial plane.
I, J, K	Unit vectors along the axes of a body fixed Cartesian coordinate system. Thrust is assumed to lie along I.
k_c	Ratio of launch energy for low thrust trajectory to that for ballistic first guess trajectory.
k_p	Solar array design factor representing the area of solar panels required to generate a unit of power at unit distance from the sun.
k_r	Retro stage tankage factor.
k_s	Solar constant, equal to the solar radiation pressure on a flat plate normal to the sun line at unit distance from the sun, assuming all photons are absorbed.
k_{st}	Spacecraft structural factor.
k_t	Low thrust propellant tankage factor.
ℓ	Number of thrusters in operation.
m	Instantaneous mass of the spacecraft.

Symbol	Definition
$\bar{m}$	Unit vector normal to the plane defined by $\bar{s}$ and $\bar{x}$ when the IJK and XYZ coordinate systems are aligned.
m_f	Final mass.
m_n	Net spacecraft mass.
m_o	Initial spacecraft mass.
m_p	Low thrust propellant mass.
m_{pp}	Low thrust propulsion system mass.
m_{rp}	Retro stage propellant mass.
m_{rt}	Retro stage tankage mass.
m_{st}	Spacecraft structural mass.
m_t	Low thrust propellant tankage mass.
$\bar{n}$	Unit vector normal to the plane of the solar panels, piercing the panels from front to back.
$\hat{n}$	Unit vector equal to $\bar{n}$ if $\bar{e}_r \cdot \bar{n} > 0$; null vector if $\bar{e}_r \cdot \bar{n} < 0$.
$\bar{p}$	Arbitrary unit vector in the body fixed coordinate system.
p_a	Power available for primary propulsion.
p_d	Power to be dissipated or irradiated into space to maintain constant power into the thrusters that are in operation.
p_i	Heliocentric distance of the $i^{\underline{th}}$ planet.
P_i	Heliocentric position vector of the $i^{\underline{th}}$ planet.

Symbol	Definition
p_o	Reference power, equal to the power generated by the power source at unit distance from the sun with the panels oriented perpendicular to the sun line.
p_r	$= p_o\,(\bar{e}_r \cdot \hat{n})$.
P_T	Heliocentric position vector of the target planet.
p_x	Constant power increment assigned for non-propulsion use.
p_1, p_2, p_3	Components of $\bar{p}$ in the spacecraft coordinate system.
$p_{I_1}, p_{I_2}, p_{I_3}$	Components of $\bar{p}$ in the inertial coordinate system.
$\bar{q}$	Unit vector normal to $\bar{x}$ about which the rotation through the Euler angle ϕ is made.
r	Heliocentric spacecraft distance.
R	Heliocentric spacecraft position vector.
r_a	Aphelion distance of the spacecraft trajectory.
r_p	Perihelion distance of the spacecraft trajectory.
R_{p_b}	Spacecraft position vector at departure of the parking orbit on the ballistic first guess trajectory.
r_T	Planetocentric spacecraft distance relative to the target planet.
R_T	Spacecraft position vector relative to the target planet.
r_{T_a}	Planetocentric apocenter distance at the target planet.
r_{T_p}	Planetocentric pericenter distance at the target planet.

Symbol	Definition
$\bar{s}$	Arbitrary vector, fixed in the body fixed coordinate system. Available for constraining along a prescribed vector.
$\bar{s}_I$	The vector $\bar{s}$ expressed in the inertial coordinate system.
T	Thrust vector.
t_o	Time of departure from the parking orbit.
$t_1, t_2, ---, t_n$	Times at which the n arcs terminate.
U	Identity matrix.
v	Heliocentric spacecraft speed.
V_{p_b}	Velocity at departure of the parking orbit on the ballistic trajectory used to get first guesses.
V_{p_o}	Velocity at departure from the parking orbit.
v_T	Planetocentric speed at the target planet.
X, Y, Z	Inertial Cartesian coordinate system.
$\bar{x}$	Specified unit vector in the inertial coordinate system along which $\bar{s}$ is constrained to lie.
x_{po}, y_{po}, z_{po}	Initial planetocentric position components relative to the launch planet.
y	Represents the state variables.
α	Angle between $\bar{n}$ and its projection in the I-J plane. Also represents the control parameters in the discussion of the transition matrix.
α_{ps}	Specific mass of the propulsion system.
β	Angle between the projection of $\bar{n}$ in the I-J plane and the I-axis. Also denotes spacecraft design and propulsion system parameters in general in the discussion of final conditions.

Symbol	Definition
γ	Power ratio. Also denotes heliocentric flight path angle in discussion of end conditions. Also represents general independent parameters in discussion of the partial derivative matrix.
γ_T	Planetocentric flight path angle at the target planet.
δ	Angle between $\bar{s}$ and its projection in the I-J plane.
Δa	Specified increment of spacecraft area.
Δp	Increment of power for each individual thruster.
Δv	Incremental velocity required of the retro stage.
$\Delta \phi$	Angle through which the launch point from the ballistic first guess trajectory is rotated to commence the low thrust trajectory.
ϵ	Angle between the projection of $\bar{s}$ in the I-J plane and the I-axis.
η	Propulsion system efficiency.
ξ, ν, ζ	Set of Euler angles defining the spacecraft orientation, relative to the inertial coordinate system in the unconstrained mode.
μ	Gravitational constant of the sun.
μ_i	Gravitational constant of the $i^{\underline{th}}$ planet.
μ_p	Gravitational constant of the launch planet.
μ_t	Gravitational constant of the target planet.
σ	Angle between $\bar{s}$ and $\bar{x}$ assuming the IJK coordinate system is aligned with the XYZ coordinate system.
ϕ, θ, ψ	Set of Euler angles defining the spacecraft orientation relative to the inertial coordinate system in the constrained mode.

Symbol	Definition
Φ	State transition matrix.
ψ_{max}	Maximum permissible value of the orientation angle ψ.
ω	Argument of perigee of the launch parking orbit.
Ω	Longitude of ascending node of the launch parking orbit on the equator.

INTRODUCTION

Over the past several years there have been a limited number of attempts to develop a low-thrust, N-body trajectory optimization program. Most, if not all, of these have been calculus of variations programs which basically differ from the widely used two-body program formulations only in the number of attracting bodies included in the gravitational field simulation. That is to say, there has been little or no concerted attempt to incorporate more realistic system and subsystem simulations and/or hardware and operational constraints in the N-body formulations. In fact, the step from two-body to N-body has been a formidable one and has not as yet been successfully completed. The problem appears to be due to the high non-linearities existing in the proximity of a planet which cause extreme convergence difficulties in the solution of the boundary value problem. And, of course, introducing constraints and increasing the complexity of the models will only serve to magnify this problem.

In this report a different approach to the development of a low-thrust, N-body trajectory optimization program is proposed. The approach is not new in that it has been successfully employed in the optimization of launch vehicle trajectories for a few years. However, it is believed that this is the first application to electric propulsion missions. The underlying principle is that the variational problem is replaced by a series of trajectory segments over which the control is held constant. Thus, over each segment (or arc), the control is represented as a vector of parameters rather than a set of time-varying functions. Consequently, the optimization problem is no longer one of the calculus of variations, but instead is a parameter optimization problem for which a variety of gradient techniques have been developed and successfully employed. An extremely important side benefit of this approach is that constraints may be added and models made more complex without a great amount of additional effort.

Specifically, the proposed approach is as follows. A trajectory is divided into a selected number of segments ending at specified times. Over each segment the thrust parameters are held constant. Using specified values of initial conditions and estimates of all independent parameters to be optimized, a trajectory is numerically integrated. A differential corrections scheme is employed to drive a selected set of end conditions to their desired values while simultaneously attempting to improve the selected performance index on successive trajectories. Typical parameters available for optimization include the thrust direction angles of each segment, segment end times, certain initial conditions, and selected propulsion system and spacecraft design parameters. Hardware constraints include the provision for making discrete steps in power as a function of distance and fixed solar array orientation relative to the thrust vector. Operational constraints include provisions for limiting the spacecraft attitude (and thereby the thrust vector) relative to a number of optional reference directions.

The proposed approach is to be built around the existing ITEM program. A brief description of the features and capabilities of ITEM is given in the following section. Subsequent sections of this report are devoted to the development of the equations required to provide the desired additional capability.

BASIC ITEM PROGRAM

The ITEM program is a general purpose Interplanetary Trajectory program using a modified Encke method. The program is designed to give maximum available accuracy without incurring prohibitive penalties in machine time.

The Encke method consists of a numerical integration of the deviation of the actual orbit from a Kepler orbit. The Kepler orbit is formulated in terms of a universal variable providing an accurate formulation without singularities. The integration proceeds in equatorial coordinates. The perturbations are integrated by a backward difference method of the sixth-order with a Runge-Kutta starter. As a recent option, a twelfth-order backward difference method with iterative self-starter is included. The integration uses the incremental eccentric anomaly as the independent variable, thus providing an expanding time step as distance from the singularity increases and avoiding the necessity of iterative solution of the modified Kepler equation. The capability for taking a fixed time step, for output purposes as well as for precisely determining the advent of events triggered on time, is also available.

The perturbations which are considered are: the planetary perturbations (planetary positions are obtained by interpolating JPL ephemeris tapes); the spherical harmonics of earth and moon to arbitrary order; drag; radiation pressure; and small thrust produced by various thrusting schemes.

The output capabilities are varied and may be called at regular intervals as well as at nodal crossings, closest approaches, and arbitary input times.

An iteration scheme is included in the program which permits the determination of initial conditions which satisfy a number of final conditions expressed in terms of osculating elements and/or impact parameters. The iteration logic is essentially similar to the MINMAX iterator. An initial guess generator based on a Lambert problem for planetary missions is also provided.

Numerous special features are provided for ease of use and flexibility of operation. The accuracy of the program has been thoroughly tested by comparison with real observations and well established analytic theories.

PROPOSED PROGRAM EXTENSIONS AND MODIFICATIONS

Equations of Motion

The equations of motion of a spacecraft in the combined gravitational field of n attracting bodies, while being acted upon by the perturbing forces of an electric propulsion system and solar radiation pressure, may be written as the sum of the accelerations due to each perturbation. That is,

$$\ddot{R} = A_g + A_p + A_{sr}$$

where the subscripts g, p, and sr refer to gravitational, propulsion and solar radiation pressure perturbations, respectively, A denotes acceleration, and R shall be taken as the heliocentric position vector of the spacecraft.*

Let P_i denote the position of the i^{th} attracting body relative to the sun, and let μ_i be that body's gravitational constant. Then we may write

$$A_g = -\mu \frac{R}{r^3} - \sum_{i=1}^{n-1} \mu_i \left[\frac{(R-P_i)}{|R-P_i|^3} + \frac{P_i}{p_i^3} \right]$$

where μ is the sun's gravitational constant, $r = |R|$, and $p_i = |P_i|$. Of course, the $(n-1)$ bodies included in the summation pertain to all attracting bodies exclusive of the sun.

---

* Unless otherwise specified, the notation in this report will conform to the following convention: upper case symbols denote vectors; lower case symbols with bars denote unit vectors; and lower case symbols without bars denote scalars.

The acceleration due to the thrust of the propulsion system may be written

$$A_p = \frac{T}{m}$$

for any propulsion system. Here T represents the thrust vector and m is the mass of the spacecraft. For an electric propulsion system, the power output is limited and it is convenient to write the thrust acceleration in terms of parameters more directly related to the propulsion system design. One such parameter is the reference power, p_r, that is generated by the power source. This is related to the thrust magnitude through the equation

$$|T| = \frac{2p_r\eta}{c}$$

where η is the propulsion system efficiency factor and c is the jet exhaust speed. The characteristics of the propulsion system parameters will be discussed subsequently.

Considering the total number of photons impinging on the spacecraft in unit time, let us assume that a ratio c_a $(0 \le c_a \le 1)$ is absorbed by the spacecraft, and that the remainder is reflected with a reflection angle equal to the angle of incidence. Then, we may write the resultant spacecraft acceleration due to solar radiation pressure, assuming that the spacecraft may be approximated as a flat plate of area a;

$$A_{sr} = \frac{k_s a}{mr^2} |\bar{e}_r \cdot \bar{n}| \left[2(1-c_a)(\bar{e}_r \cdot \bar{n})\bar{n} + c_a \bar{e}_r \right]$$

where $\bar{e}_r$ is a unit vector along the heliocentric position vector (i.e., $R = r\bar{e}_r$), $\bar{n}$ is a unit vector normal to the assumed flat plate representing the spacecraft, and k_s is a solar constant representing the solar radiation pressure on a flat plate normal to the sun line at unit distance from the sun, assuming all photons are absorbed.

Propulsion System Characteristics

It is essential that the program be capable of simulating the characteristics of either nuclear or solar electric propulsion systems. The mathematical descriptions of the two types of propulsion systems differ primarily in the representation of p_r, the power generated. For a nuclear electric system, p_r is generally assumed to be constant over a trajectory, while for a solar electric system, p_r will vary with the solar distance and with the angle of incidence of the sun's rays to the solar panels. Let p_o represent the power that can be generated by the solar panels at unit distance from the sun if the panels are normal to the sun line, and let γ denote the ratio of power available at any specific solar distance to that available at unit distance. Furthermore, let $\bar{n}$ be the unit vector normal to the panels with a direction such that the solar cells are facing the sun providing $\bar{e}_r \cdot \bar{n} > 0$. Then we may write, for the power available to the propulsion system, p_a

$$p_a = p_o \gamma (\bar{e}_r \cdot \hat{n}) - p_x$$

where p_x represents a constant increment of power that may be assigned for purposes other than primary propulsion. Unless otherwise specified, it will be assumed that $p_r = p_a$. $\hat{n}$ is related to $\bar{n}$ as follows:

$$\hat{n} = \begin{cases} \bar{n} \text{ if } \bar{e}_r \cdot \bar{n} > 0 \\ \bar{0} \text{ if } \bar{e}_r \cdot \bar{n} < 0 \end{cases}$$

The assumed form of γ is

$$\gamma = \frac{1}{r^2} \sum_{i=0}^{4} a_i r^{-i/2}$$

where a_i, $i = 0, 1, ---, 4$, are input constants. Nuclear electric propulsion is simulated by setting

$$\gamma (\bar{e}_r \cdot \hat{n}) = 1 .$$

For either type of propulsion system, the jet exhaust speed is assumed constant over the trajectory. The efficiency factor is taken to be a function of the jet exhaust speed of the form

$$\eta = \frac{bc^2}{c^2 + d^2}$$

where b and d are specified constants. The mass flow of the propulsion system is

$$\dot{m} = - \frac{|T|}{c} .$$

It should be noted that the acceleration due to radiation pressure will generally be a factor of interest only for solar-electric systems. For such systems, the assumption that the spacecraft is a flat plate in the computation of the solar pressure is appropriate since the solar panels comprise the major portion of the spacecraft area exposed to the sun. The area a of the solar panels, which appears in the equation for solar pressure, is related to the power capacity of the panels through the equation

$$a = k_p p_o + \Delta a$$

where k_p is a specified quantity representing the area of solar cells required to generate a unit of power at unit distance from the sun, and Δa is a specified constant.

Depending upon the particular propulsion system hardware involved, it may be desirable to force the power input to a thruster to be constant. Since, for a solar electric system, the power generated will not be constant, it will be necessary to modulate the total power fed to the thrusters and to control the number of thrusters in use at each instant in time. To this end, define a power increment Δp which represents the desired power level of each individual thruster. Then for a given power available to the propulsion system, p_a, the number of thrusters that may be in operation is

$$\ell = \mathrm{mod}\,(p_a,\ \Delta p)$$

where mod (a, b) is the largest integer that is less than or equal to the quotient a/b. This means that a portion of the power available, p_d, given by

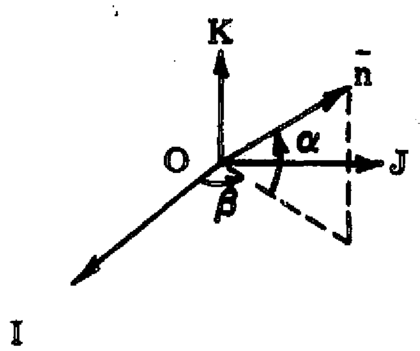
$$p_d = p_a - \ell\,\Delta p$$

must be discarded. The actual power used by the propulsion system is

$$p_r = \ell \, \Delta p \quad .$$

<u>Spacecraft Design Considerations</u>

For reliability reasons, it is desirable to fix the orientation of the solar panels of an electrically propelled spacecraft relative to the central portion of the spacecraft. Consequently, the normal vector, $\bar{n}$, that appears in both the propulsion and the solar pressure terms of the equations of motion must be expressed in a body-fixed coordinate system. For this purpose, introduce the Cartesian frame OIJK which is fixed in the spacecraft. Relative to this system

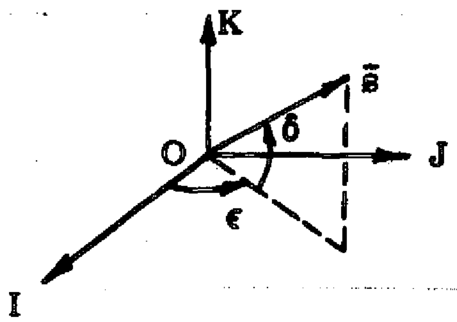


$\bar{n}$ would be written

$$\bar{n} = \cos\alpha\cos\beta\, I + \cos\alpha\sin\beta\, J + \sin\alpha\, K$$

where the angles α and β would be design parameters that are specified and held constant throughout the trajectory.

It is also useful to define an additional arbitrary unit vector, say $\bar{s}$, that is fixed in the OIJK system, i.e.,



$$\bar{s} = \cos\delta \cos\epsilon\, I + \cos\delta \sin\epsilon\, J + \sin\delta\, K$$

The angles δ and ϵ are also specified design parameters that are held constant throughout the trajectory. The purpose of this vector will be discussed subsequently in the section pertaining to flight mode options.

The thrust direction, defined by the unit vector

$$\bar{e}_T = T / |T|$$

is assumed fixed in the OIJK coordinate system and directed through the center of gravity. We will, without loss of generality, assume that

$$\bar{e}_T = I$$

Flight Mode Options

It is desired to provide the capability of generating approximate optimum trajectories for a given mission for a variety of flight modes. For all flight modes, the trajectory is divided into a series of arcs. Over each arc the control parameters are held constant, and we seek the optimum control parameters of each arc. In general, the available control parameters will include certain initial conditions, propulsion system parameters, engine on and off times, and the thrust angles. Two basic flight modes will be considered. The first is termed the unconstrained mode, for which complete freedom is permitted in selecting the optimum spacecraft orientation for a given arc. The second mode is termed the constrained mode because the body-fixed unit vector $\bar{s}$, discussed previously, is required to continuously lie within a cone of specified half-angle about a known vector. The orientation of $\bar{s}$ within the cone is arbitrary as is the roll orientation of the spacecraft about $\bar{s}$.

A. Unconstrained Mode

Let XYZ represent the inertial frame of reference in which the trajectory computations (heliocentric) are made. We define the orientation of the spacecraft in terms of the IJK axes, relative to the inertial frame through the angles ξ, ν, ζ. Let ξ denote the nodal angle which is a rotation about the Z axis to the node of the I - J and X - Y planes; ν denotes a rotation about the nodal line and is the inclination of the I - J plane relative to the X - Y plane; ζ denotes a rotation in the I - J plane from the node to the I - axis. The transformation from the XYZ to the IJK system is

$$A(\xi, \nu, \zeta) = \begin{bmatrix} (c\zeta c\xi - s\zeta s\xi c\nu) & (c\zeta s\xi + s\zeta c\xi c\nu) & (s\zeta s\nu) \\ (-s\zeta c\xi - c\zeta s\xi c\nu) & (-s\zeta s\xi + c\zeta c\xi c\nu) & (c\zeta s\nu) \\ (s\xi s\nu) & (-c\xi s\nu) & (c\nu) \end{bmatrix}$$

where s and c denote sine and cosine, respectively. Since the transformation is orthogonal, the inverse of $A(\xi, \nu, \zeta)$ is equal to its transpose, facilitating the transformation from the IJK to the XYZ system. Thus, the vectors $\bar{n}$ and $\bar{e}_T$ in the inertial coordinate system, which are required for the equations of motion, are

$$\bar{n} = A(\xi, \nu, \zeta)^T \begin{bmatrix} \cos\alpha\cos\beta \\ \cos\alpha\sin\beta \\ \sin\alpha \end{bmatrix}$$

$$\bar{e}_T = A(\xi, \nu, \zeta)^T \begin{bmatrix} 1 \\ 0 \\ 0 \end{bmatrix}$$

where the superscript T denotes transpose. The choice of the three angles at the start of each arc is completely unconstrained; however, once the choice is made, the angles are held constant throughout the arc.

B. Constrained Mode

A problem of particular interest is that of optimizing the motion of the spacecraft subject to the constraint that the antenna, which is fixed relative to the spacecraft, be continuously pointing in the general direction of the Earth. Other constrained problems of interest include thrusting at a fixed angle to the sun line and thrusting along the velocity vector. All of these problems may be considered within the framework of the following general formulation.

Consider the body-fixed unit vector $\bar{s}$, given by

$$\bar{s} = \cos \delta \cos \epsilon \, I + \cos \delta \sin \epsilon \, J + \sin \delta \, K$$

where, as stated previously, δ and ϵ are specified angles representing spacecraft design factors. $\bar{s}$ is an arbitrary vector that may be used for different purposes in different problems. As an example, it may represent the antenna pointing direction. In the most general constrained case, we require that the spacecraft be continuously oriented such that $\bar{s}$ lies within a cone of specified half angle about a known unit vector $\bar{x}$ (t). This latter vector will be determined internally in the program according to an input option flag. The input options will permit defining $\bar{x}$ (t) as the direction from the spacecraft to any planet, the sun, or a star, or as the direction of the heliocentric velocity vector. The position of $\bar{s}$ in the cone about $\bar{x}$ (t), which requires two angles for specification, may be specified or optimized. Also the orientation of the spacecraft about $\bar{s}$ is available as a parameter for optimization.

To determine the inertial components of certain body-fixed vectors, e.g., $\bar{e}_T$ and $\bar{n}$, it is necessary to define a series of rigid body rotations and the transformations associated with each. To this end, consider two arbitrary unit vectors $\bar{a}$ and $\bar{b}$ and suppose $\bar{a}$ is rotated about $\bar{b}$ through an angle α to a position $\bar{c}$. In general, $\bar{c}$ is given by

$$\bar{c} = A(\bar{b}, \alpha)\, \bar{a}$$

where $A(\bar{b}, \alpha)$ is a transformation matrix of the form (See Reference 1)

$$A(\bar{b}, \alpha) = U \cos \alpha + B (1 - \cos \alpha) + C \sin \alpha$$

with U being the 3 x 3 identity matrix,

$$B = \begin{bmatrix} b_1 b_1 & b_1 b_2 & b_1 b_3 \\ b_2 b_1 & b_2 b_2 & b_2 b_3 \\ b_3 b_1 & b_3 b_2 & b_3 b_3 \end{bmatrix}$$

and

$$C = \begin{bmatrix} 0 & -b_3 & b_2 \\ b_3 & 0 & -b_1 \\ -b_2 & b_1 & 0 \end{bmatrix}$$

where b_1, b_2, b_3 are the components of $\bar{b}$.

At any instant in time, the spacecraft orientation in inertial space may be defined through the following sequence of rotations. First, consider the body axis system IJK aligned with the inertial frame such that the components of $\bar{s}$ are identical in the body and the inertial coordinate systems. Then rotate the IJK system through an angle σ such that $\bar{s}$ is aligned with $\bar{x}$ (t). The angle σ is given by

$$\sigma = \cos^{-1}(\bar{s} \cdot \bar{x}) \qquad (0 \le \sigma \le \pi)$$

and the rotation is about a unit vector $\bar{m}$ which is normal to both $\bar{s}$ and $\bar{x}$, i.e.,

$$\bar{m} = (\bar{s} \times \bar{x}) / \sin \sigma$$

The transformation for this rotation is, of course, $A(\bar{m}, \sigma)$ for which the general form is given above. We now seek two rotations which together uniquely define the location of $\bar{s}$ within the cone about $\bar{x}$. For reasons which will become clear subsequently, we choose first a rotation about $\bar{x}$ through an arbitrary angle ϕ followed by a rotation about a vector normal to $\bar{x}$ through an angle ψ. The particular normal vector about which this latter rotation takes place is entirely arbitrary and we pick, for convenience, one which is easily attainable and which is known to be normal to $\bar{x}$ without checking. This vector is the one which $\bar{m}$ would occupy after rotating through the angle ϕ about $\bar{x}$. Denote this vector $\bar{q}$, then

$$\bar{q} = A\ (\bar{x},\ \phi)\ \bar{m}$$

and the transformation matrix for the third rotation is $A\,(\bar{q},\ \psi)$. This particular choice for the second and third rotations was made because it results in a single control parameter, namely ψ, through which the constraint that $\bar{s}$ lie within a specified cone about $\bar{x}$ may be imposed. Letting ψ_{max} represent the specified half-cone angle, then the constraint is satisfied simply by forcing

$$\psi \leq \psi_{max}$$

If the value of ψ to be used at a particular time satisfies the inequality constraint, then the solution proceeds as if there were no constraint. If, however, the inequality constraint would be violated, then ψ is set equal to ψ_{max}. Denoting as $\bar{s}_I$ the vector $\bar{s}$ in the inertial system, we then have

$$\bar{s}_I = A(\bar{q}, \psi)\,\bar{x}$$

The final rotation required is one about $\bar{s}_I$ through an angle θ, for which the transformation matrix is $A(\bar{s}_I, \theta)$. The angle θ defines the roll orientation of the spacecraft about the vector $\bar{s}$ (or $\bar{s}_I$), and is an independent parameter for optimization.

The application of the above-defined transformations permits one to obtain the inertial components of any unit vector, say $\bar{p}$, that is fixed in the IJK system and has the known components p_1, p_2, p_3. Denoting the inertial components of $\bar{p}$ as $p_{I_1}, p_{I_2}, p_{I_3}$ then

$$\begin{bmatrix} p_{I_1} \\ p_{I_2} \\ p_{I_3} \end{bmatrix} = A(\bar{s}_I, \theta)\, A(\bar{q}, \psi)\, A(\bar{x}, \phi)\, A(\bar{m}, \sigma) \begin{bmatrix} p_1 \\ p_2 \\ p_3 \end{bmatrix}$$

Of course, this transformation is valid for both $\bar{p} = \bar{n}$ and for $\bar{p} = \bar{e}_T$.

Two special cases of the above formulation for the constrained mode warrant mention. The first is that by setting ψ_{max} equal to a large number, one may effectively run unconstrained trajectories with the same control parameters used in the constrained mode. The second is that $\bar{s}$ may be forced to coincide with $\bar{x}$ exactly by requiring that

$$\psi = \psi_{max} = 0$$

However, if this is done, then ϕ and θ are not independent parameters and it is convenient to set $\phi = 0$. If this is done, then both $A(\bar{x}, \phi)$ and $A(\bar{q}, \psi)$ degenerate to the identity matrix, and θ is the only spacecraft orientation angle remaining to be optimized.

The angles ϕ, ψ, and θ are held constant over each arc of the trajectory, although they are permitted to vary from arc to arc. Also the input angle ψ_{max} will be held constant over each arc, but will be permitted to change at the start of each arc. This will allow the study of cases in which $\bar{s}$ is either little or not at all constrained over some arcs and very tightly constrained over other arcs.

Optimization Parameters

At the user's option, the following parameters will be made available for optimization.

Initial conditions

i, Ω, ω - inclination, longitude of ascending node, and argument of perigee of the launch parking orbit,

x_{p_o}, y_{p_o}, z_{p_o} - initial planetocentric position components, and

$\dot{x}_{p_o}$, $\dot{y}_{p_o}$, $\dot{z}_{p_o}$ - initial planetocentric velocity components; or

t_o - time at departure of parking orbit, and

$|V_{p_o}|$ - speed at departure of parking orbit

Propulsion system parameters

p_o - reference power

c - jet exhaust speed

Spacecraft design parameters

α, β —— direction angles of solar array normal vector

δ, ϵ —— direction angles of constraint vector (e.g., antenna pointing vector)

Arc end times

t_1, t_2, - - -, t_n —— times at which arcs terminate

Thrust angles (independent set for each arc)

Unconstrained mode

ξ, ν, ζ

Constrained mode

ϕ, ψ, θ

<u>Initial Conditions</u>

Provisions are made for inputting the initial position and velocity of the spacecraft at departure of the launch parking orbit. These inputs are in the form of the geocentric ecliptic Cartesian coordinates of the position and velocity. However, because appropriate values for the initial position and velocity may be difficult to estimate, an option is provided to assist in generating reasonable first guesses. When this option is employed, the solution is begun by generating a ballistic patched-conic trajectory between the appropriate planets, using the specified launch date, t_ℓ, and flight time, Δt. This trajectory is comprised of planetocentric hyperbolic trajectories at each end which are patched to a connecting heliocentric trajectory at the spheres of influence. An iteration is performed such that the planetocentric trajectories satisfy certain conditions such as specified periplanet height and compatibility with a given launch site. Once this iteration has converged, the conditions at perigee are employed to fix the elements of the circular launch parking orbit, including the ephemeris of the spacecraft in the orbit. Denote the geocentric position vector and velocity vector at perigee of the resulting ballistic hyperbolic escape trajectory as R_{p_b} and V_{p_b}, respectively. Then the launch energy, c_{3_b}, of the escape trajectory is

$$c_{3_b} = |V_{p_b}|^2 - 2\mu_p / |R_{p_b}|$$

where μ_p is the gravitational constant of the launch planet (Earth). Now suppose that the launch energy of the escape hyperbola for the low thrust trajectory is a specified fraction, say k_c, of the ballistic launch energy. Also suppose that the point of departure from the circular parking orbit for the reduced energy trajectory is chosen such that the resulting hyperbolic excess velocity is in the same direction as that of the ballistic trajectory and that the hyperbolic orbit is tangent to the parking orbit at the perigee point. Under these assumptions, the launch point is rotated around the parking orbit in a direction opposite the direction of motion (assuming $k_c < 1$) through an angle $\Delta\phi$ defined by

$$\Delta\phi = \cos^{-1}\left[\frac{1}{e_b e_p}\left(1 + \sqrt{(e_p^2 - 1)(e_b^2 - 1)}\right)\right]$$

where $$e_b = 1 + \frac{|R_{p_b}|\, c_{3_b}}{\mu_p} \quad ; \quad e_p = 1 + \frac{|R_{p_b}|\, k_c\, c_{3_b}}{\mu_p}$$

are the eccentricities of the ballistic and reduced energy hyperbolic trajectories, respectively. Thus, the initial geocentric position and velocity, R_{p_o} and V_{p_o}, are obtained simply by rotating the R_{p_b} and V_{p_b} through the angle $(-\Delta\phi)$ and setting the initial velocity magnitude to

$$|V_{p_o}| = \sqrt{k_c\, c_{3_b} + 2\,\mu_p / |R_{p_b}|}$$

Of course, the new launch time, referenced to the ballistic launch date t_ℓ becomes

$$t_o = -\,\Delta\phi \sqrt{|R_{p_b}|^3 / \mu_p}$$

It is expected that the above procedure to obtain initial conditions will only be employed when there exists virtually no information from which reasonable starting conditions can be obtained. If a reasonable set of initial conditions are available, they may be input directly in terms of the Cartesian coordinates x_{p_o}, y_{p_o}, z_{p_o}, $\dot{x}_{p_o}$, $\dot{y}_{p_o}$, $\dot{z}_{p_o}$ with launch time being prescribed through the input launch date.

Once the initial position, velocity, and time are defined, the motion of the spacecraft is integrated forward along a coasting arc in the n-body force field (no solar radiation pressure) to an input time t_1, which represents the time at which the electric propulsion system is turned on and solar radiation pressure is included in the perturbations. Typically, t_1 will be several hours past t_o. The introduction of the time interval between the high-thrust phase and the starting of the low-thrust phase serves two purposes. First, it includes in the simulation a phase that will be necessary in a mission to erect solar arrays and/or make other preparations, and check out systems before commencing the interplanetary phase of the mission. Secondly, the time interval permits the vehicle to recede from the proximity of a planet which, inherently, is a region of high non-linearities and which frequently cause considerable difficulty in the trajectory and mission analysis.

Two options are available for optimizing the point of departure from the parking orbit. In one option, the spacecraft is assumed to have a prescribed ephemeris in the given orbit such that the variation of the launch time results in the variation of the point of departure as defined by two-body motion in the parking orbit. In this option the independent parameters are the speed at departure $|V_{po}|$ and the initial time t_o. Partial derivatives of the state with respect to these parameters are obtained analytically rather than by finite differences over the first arc. These partials are obtained:

$$\frac{\partial R_{p_o}}{\partial |V_{p_o}|} = 0 \quad ; \quad \frac{\partial \dot{R}_{p_o}}{\partial |V_{p_o}|} = \frac{V_{p_o}}{|V_{p_o}|}$$

$$\frac{\partial R_{p_o}}{\partial t_o} = \sqrt{\frac{\mu_p}{|R_{p_o}|}} \; \frac{V_{p_o}}{|V_{p_o}|} \quad ; \quad \frac{\partial \dot{R}_{p_o}}{\partial t_o} = - \frac{\mu_p}{|R_{p_o}|^3} \; R_{p_o}$$

$$\frac{\partial m_o}{\partial |V_{p_o}|} = - \frac{a_1}{a_2} \, e^{-|V_{p_o}|/a_2} \quad ; \quad \frac{\partial m_o}{\partial t_o} = 0$$

where R_{p_o}, $\dot{R}_{p_o}$ are the planetocentric position and velocity at departure of the launch parking orbit. The partials at the end of the first (two-body) arc are then immediately obtained by multiplying these partials by the state transition matrix for that arc. The partials of the initial mass m_o are obtained by differentiating the equation

$$m_o = a_1 \, e^{-|V_{p_o}|/a_2} - a_3$$

where a_1, a_2 and a_3 are input constants describing the performance capability of a specific launch vehicle. The second option permits variations in initial position and velocity through variations in the circular launch parking orbit orientation and the speed of departure. As in the previous option, the departure point is assumed to coincide with the periapse of the departure hyperbola. From the input planetocentric position and velocity vectors R_{p_o}, $\dot{R}_{p_o}$, compute the angular momentum

$$H = R_{p_o} \times \dot{R}_{p_o}$$

and define $$\bar{h} = H / |H| \quad ; \quad \bar{\ell} = (\bar{k} \times \bar{h}) / \sin i$$

where $\bar{k}$ is a unit vector along the North Pole and i is the inclination of the parking orbit relative to the equator. The unit vector $\bar{\ell}$ lies along the ascending node of the parking orbit on the equator. The inclination is obtained from the equation

$$\cos i = \bar{h} \cdot \bar{k}$$

Letting the independent parameters for this option be the inclination i, the longitude of node Ω, the argument of perigee ω, and the departure speed $|V_{p_o}|$, the desired partials are written

$$\frac{\partial X}{\partial i} = \bar{\ell} \times X$$

$$\frac{\partial X}{\partial \Omega} = \bar{k} \times X$$

$$\frac{\partial X}{\partial \omega} = \bar{h} \times X$$

$$\frac{\partial m_o}{\partial i} = \frac{\partial m_o}{\partial \Omega} = \frac{\partial m_o}{\partial \omega} = 0$$

where X denotes either R_{p_o} or $\dot{R}_{p_o}$. The partials with respect to $|V_{p_o}|$ remain the same as for the first option above. Again, the partials at the end of the first arc are obtained by multiplying the above by the state transition matrix.

State Transition Matrix

The state transition matrix is generated by dividing the numerical differences in the state on perturbed and nominal trajectories by the magnitude of the perturbation. The desired partial derivatives are those of the final position, velocity, and mass with respect to each independent parameter which is to be optimized. With these available, the partials of most functions and end conditions of interest can be formed. Note that this definition is a generalization of the usual definition of state transition matrix. As commonly defined, state transition matrix includes partials of state with respect only to state at an earlier time. Here partials with respect to all independent parameters are included.

A straightforward approach to the evaluation of the transition matrix involves the integration of a perturbation trajectory for each independent parameter to be optimized. For a specific case involving n independent parameters, as many as $n + 1$ trajectories (n perturbed plus 1 nominal) could be required to produce the desired matrix. Because it will, in general, be necessary to generate several matrices before convergence is attained, it is quite clear that a large number of trajectories must be integrated to yield a solution. It so happens, however, that because of the particular form of the problem, it is possible to generate the transition matrix with fewer than the standard number of perturbation trajectories. This circumstance arises because the orientation angles for each arc of the trajectory are treated as individual control parameters which directly affect the motion of the spacecraft only over a portion of the trajectory. For example, consider a trajectory consisting of two arcs.

Let y_i, $i = 0, 1, 2$, represent the state at the end of the i^{th} arc (0 denotes the start of the first arc) which occurs at time t_i, let α_1 and α_2 represent the control over the first and second arc, respectively, and let β represent the propulsion system and spacecraft design parameters. The particular matrix that we seek may be written

$$\Phi(0,2) = \left[\frac{\partial y_2}{\partial y_0} \,\vdots\, \frac{\partial y_2}{\partial \beta} \,\vdots\, \frac{\partial y_2}{\partial \alpha_1} \,\vdots\, \frac{\partial y_2}{\partial \alpha_2} \right]$$

Here y_0 is meant to include the time t_0 the speed $|V_{p_o}|$ or any of the launch parking orbit orientation angles i, Ω, or ω that are declared as independent variables. Alternatively, it will include only initial state variables that are not fixed. To obtain this matrix, we begin by integrating the nominal and perturbation trajectories over the first arc, stopping the integration at t_1 and forming the transition matrix for that arc, i.e.,

$$\Phi(0,1) = \left[\frac{\partial y_1}{\partial y_0} \,\vdots\, \frac{\partial y_1}{\partial \beta} \,\vdots\, \frac{\partial y_1}{\partial \alpha_1} \right]$$

where the partials are formed by simple finite differencing. This matrix is stored in the locations allotted for the final matrix $\Phi(0,2)$, and the integration of the nominal trajectory is continued over the next arc. The perturbation trajectories are not continued, however. Rather, a new set is begun at time t_1 and integrated over the interval t_1 to t_2. There is a perturbation trajectory over the second arc for each state variable, for each propulsion system and spacecraft design parameter, and for each control parameter in effect over that arc, but none for any control parameters of a previous arc. At the end of the arc, one forms the partials

$$\frac{\partial y_2}{\partial y_1}\ , \quad \left(\frac{\partial y_2}{\partial \beta}\right)^* , \quad \text{and} \quad \frac{\partial y_2}{\partial \alpha_2}$$

where $(\partial y_2/\partial \beta)^*$ are the partials in the state at t_2 due to variations in the propulsion system and spacecraft design parameters over the second arc only. Then, form $\Phi(0,2)$ through the operations

$$\Phi(0,2)^* = \left[\ \frac{\partial y_2}{\partial y_1}\ \Phi(0,1) \ \vdots\ \frac{\partial y_2}{\partial \alpha_2}\ \right]$$

$$\Phi(0,2) = \Phi(0,2)^* + \left[\ 0 \ \vdots\ \left(\frac{\partial y_2}{\partial \beta}\right)^* \ \vdots\ 0 \ \vdots\ 0\ \right]$$

$$= \left[\ \frac{\partial y_2}{\partial y_0} \ \vdots\ \frac{\partial y_2}{\partial \beta} \ \vdots\ \frac{\partial y_2}{\partial \alpha_1} \ \vdots\ \frac{\partial y_2}{\partial \alpha_2}\ \right]$$

Of course, the extension to more arcs is straightforward. In general, we will seek the matrix $\Phi(0,i)$ given by

$$\Phi(0,i) = \left[\ \frac{\partial y_i}{\partial y_0} \ \vdots\ \frac{\partial y_i}{\partial \beta} \ \vdots\ \frac{\partial y_i}{\partial \alpha_1} \ \vdots\ \frac{\partial y_i}{\partial \alpha_2} \ \vdots\ - - - - \ \vdots\ \frac{\partial y_i}{\partial \alpha_i}\ \right]$$

$$= \Phi(0,i)^* + \left[\ 0 \ \vdots\ \left(\frac{\partial y_i}{\partial \beta}\right)^* \ \vdots\ 0 \ \vdots\ 0 \ \vdots\ - - - \ \vdots\ 0\ \right]$$

with $$\Phi(0,i)^* = \left[\ \frac{\partial y_i}{\partial y_{i-1}}\ \Phi(0,\ i-1) \ \vdots\ \frac{\partial y_i}{\partial \alpha_i}\ \right]$$

Once the transition matrix is formed for the first arc, each subsequent arc will add to the matrix a number of columns equal to the number of control parameters in effect over that arc. The number of rows remains constant and is equal to the number of state variables.

It should be noted at this point that over the first arc of the trajectory, perturbation trajectories need be run only for those particular state variables for which the initial values will be permitted to vary independently and for any other independent parameters. On all subsequent arcs, however, it is necessary to generate perturbation trajectories for the current independent parameters and for all state variables. Thus, since there are seven state variables, the matrix $(\partial y_i/\partial y_{i-1})$ will be 7x7 for $i \geq 2$ and will be $7 \times m$ for $i=1$ where m is the number of independent variations permitted in the state at the start of the first arc. If any propulsion system or spacecraft design parameters are to be optimized, but do not enter into the problem in the first arc, it is convenient to enter a column of zeroes for each such parameter in the transition matrix so that the general form of the matrix is established immediately, and the above formula for $\Phi(0, i)$ is always applicable for $i \geq 2$. As an example, consider the problem of a solar-electric propulsion mission commencing from the vicinity of a low altitude orbit about Earth. Suppose the initial geocentric speed is to be optimized as are the power level and jet exhaust speed of the low-thrust propulsion system and three spacecraft orientation angles over each arc. The first arc consists of coasting on the geocentric hyperbolic trajectory to a specified time t_1 at which the low-thrust system is turned on. The only independent parameter for optimization over this first arc is the geocentric speed at the start of the arc. Consequently, one perturbation trajectory over the first arc must be generated, and the transition matrix will be 7×3. The first column is associated with the initial geocentric speed, and the last two columns are zeroes which are entered in anticipation of future effects of variations in the propulsion system parameters, power and jet exhaust speed. It may be argued that the initial mass is also varying due to its dependence on the velocity, but it is precisely because the variation is not independent that a perturbation trajectory for the initial mass is not required. Over the second arc, a total of twelve perturbation trajectories are required -- seven for the state, two for the propulsion system parameters, power and jet exhaust speed, and three for the orientation angles. Following the procedure indicated above, three columns (associated with the three orientation angles) will be added to the transition matrix such that, at the end of the second arc, the matrix is 7×6. The continuation of this procedure to subsequent arcs is then clear.

It was implicitly assumed above that the time at the end of each arc is fixed. If an arc end time is left open for optimization, then a column must be added to the transition matrix. It is <u>not</u> necessary to integrate a perturbation trajectory for this arc end time as the partials are available in closed form. For example, denote t_i as the end time of the i^{th} arc and let y_i represent the state at t_i. Then, if t_i is to be optimized, form the column matrix

$$\frac{\partial y_i}{\partial t_i} = \dot{y}_i^- - \dot{y}_i^+$$

where the minus and plus signs indicate the limits of the time derivative of y_i evaluated just before and just after the end of the arc, respectively. If i is the last arc, the $\dot{y}_i^+ = 0$. This column is added to the transition matrix, and subsequently is treated in the identical manner as the partials with respect to the control parameters, α_i, for the i^{th} arc. Note that if the time derivative of a state variable (such as position) is continuous from one arc to the next, the partial is zero. However, for velocity and mass, the switching on or off of an engine, or the change of a thrust angle, can result in a finite partial derivative.

Final Conditions

The particular iterator being used to perform the parameter optimization, and to solve the boundary value problem, permits the selection of the performance index of the problem from the same list of functions available for specification as final conditions. Regardless of whether a function is specified as the performance index or a final condition that is to be satisfied, the iterator requires as inputs the value of the function on the current nominal trajectory and the partial derivatives of the function with respect to all the independent parameters of the problem.

We permit any specific final condition, f, to be, in its most general form, an explicit function of the final state, y_f, the final time, t_f, the spacecraft design or propulsion system parameters, β, and/or the initial state, y_o

$$f = f(y_f, t_f, \beta, y_o)$$

Then, denoting the independent parameters of the problem as γ, the desired partial derivatives of f are

$$\frac{\partial f}{\partial \gamma} = \frac{\partial f}{\partial y_f} \Phi(o, \ell) + \frac{\partial f}{\partial t_f} \frac{\partial t_f}{\partial \gamma} + \frac{\partial f}{\partial \beta} \frac{\partial \beta}{\partial \gamma} + \frac{\partial f}{\partial y_o} \frac{\partial y_o}{\partial \gamma}$$

where $\Phi(o, \ell)$ is the transition matrix over the entire trajectory. Clearly the partials $\partial t_f/\partial \gamma$, $\partial \beta/\partial \gamma$, and $\partial y_o/\partial \gamma$ will be either zero or one since t_f, β, and y_o are potential independent parameters. Consequently, the only derivatives that are as yet undefined are those of f with respect to y_f, t_f, β, and y_o. Of course, these depend on the specific form of f and, in order to write them down, it is imperative to list the final conditions to be made available for optional use. This list of conditions, along with the concomitant partial derivatives, follows.

Mass: The mass-related functions that shall be available for end conditions include the final mass, m_f, and the net spacecraft mass, m_n (frequently referred to as payload). For the purpose of this program, the total spacecraft mass, m_o, is assumed to consist of the power plant m_{pp}, propellant m_p, tankage m_t, structure m_{st}, net spacecraft m_n, and when applicable, retro stage propellant m_{rp} and structure m_{rt}; i.e.,

$$m_o = m_{pp} + m_p + m_t + m_{st} + m_{rp} + m_{rt} + m_n$$

where $$m_{pp} = \alpha_{ps} P_o$$

$$m_t = k_t m_p$$

$$m_{st} = k_{st} m_o$$

$$m_{rp} = m(t_f)(1 - e^{-\Delta v / c_r})$$

$$m_{rt} = k_r m_{pr}$$

with α_{ps} being the specific mass of the propulsion system; k_t, k_{st} and k_r are structural scaling factors; Δv is the retro maneuver velocity increment; and c_r is the jet exhaust speed of the retro stage engine. Of course, $m(t_f)$ is the final spacecraft mass just prior to any retro maneuver. For capture missions, impulsive thrust is assumed for the retro maneuver. Of course, there is no retro maneuver for flyby missions. Capture orbits will be defined in terms of the perifocal and apofocal distances, r_{Tp} and r_{Ta}, relative to the target planet, and the retro maneuver will be assumed to take place at perifocus. Consequently, the incremental velocity, Δv, of the retro maneuver is equal to the difference between the speed at perifocus of the hyperbolic trajectory on which the spacecraft approaches the target planet and at perifocus of the desired elliptic orbit. That is, we define

$$\Delta v = v_{T_p} - \left(\frac{2\mu_T r_{Ta}}{r_{Tp}(r_{Tp} + r_{Ta})} \right)^{\frac{1}{2}}$$

$$v_{T_p} = \left[\mu_T \left(\frac{2}{r_{T_p}} - \frac{1}{a_T} \right) \right]^{\frac{1}{2}}$$

where μ_T is the gravitational constant of the target planet and a_T is the semi-major axis of the planetocentric hyperbolic approach trajectory as evaluated at the final time. Of course, on any specific trial trajectory in the iteration, the planetocentric distance at the end of the trajectory may not be the perifocal distance desired. In fact, depending on the specific condition used to terminate the trajectory, the final position may not even be a perifocal point. Thus, prior to convergence, the Δv will not represent a meaningful quantity, but as the iteration proceeds to the solution, Δv will take on the value desired. The important points are that the definition of Δv remains consistent throughout the iteration, is representative of each trial trajectory, and converges to the desired quantity when the specified end conditions are satisfied.

The final mass is evaluated as follows:

$$m_f = m(t_f) - m_{rp}$$

or

$$m_f = m(t_f)\, e^{-\Delta v/c_r}$$

and the required partial derivatives are

$$\frac{\partial m_f}{\partial y_f} = \frac{\partial m(t_f)}{\partial y_f}\, e^{-\Delta v/c_r} + \frac{m(t_f)}{c_r}\, e^{-\Delta v/c_r}\, \frac{\partial |\dot{R}_T(t_f)|}{\partial y_f}$$

$$\frac{\partial m_f}{\partial t_f} = \frac{m(t_f)}{c_r}\, e^{-\Delta v/c_r}\, \frac{\partial |\dot{R}_T(t_f)|}{\partial t_f}$$

$$\frac{\partial m_f}{\partial \beta} = \frac{\partial m_f}{\partial y_o} = 0$$

Since $m(t_f)$ is the final value of one of the state variables, $\partial m(t_f)/\partial y_f$ is a matrix of seven elements consisting of 6 zeroes and a one. The partials of $|\dot{R}_T(t_f)|$ will be derived in a subsequent paragraph.

The net spacecraft mass is evaluated

$$m_n = m_f - m_{pp} - m_t - m_{st} - m_{rt}$$

Rewriting in a more suitable form for differentiating

$$m_n = \left[(1+k_r)\, e^{-\Delta v/c_r} + k_t - k_r \right] m(t_f) - (k_t + k_{st})\, m_o - \alpha_{ps}\, p_o$$

then
$$\frac{\partial m_n}{\partial y_f} = \left[(1+k_r)\, e^{-\Delta v/c_r} + k_t - k_r \right] \frac{\partial m(t_f)}{\partial y_f} + \frac{m(t_f)}{c_r} (1+k_r)\, e^{-\Delta v/c_r} \frac{\partial |\dot{R}_T(t_f)|}{\partial y_f}$$

$$\frac{\partial m_n}{\partial t_f} = \frac{m(t_f)}{c_r} (1+k_r)\, e^{-\Delta v/c_r} \frac{\partial |\dot{R}_T(t_f)|}{\partial t_f}$$

$$\frac{\partial m_n}{\partial \beta} = -\alpha_{ps} \frac{\partial p_o}{\partial \beta}$$

$$\frac{\partial m_n}{\partial y_o} = (k_t + k_{st}) \frac{(m_o + a_3)}{a_2} \frac{\partial |V_{p_o}|}{\partial y_o}$$

where a_2 and a_3 are as defined in the preceding section, and where $|V_{p_o}|$ may be one of the y_o's directly or related to the y_o's through the equation

$$|V_{p_o}|^2 = \dot{x}_{p_o}^{\ 2} + \dot{y}_{p_o}^{\ 2} + \dot{z}_{p_o}^{\ 2}$$

Of course, p_o is one of the β's and $\dot{x}_{p_o}$, $\dot{y}_{p_o}$, $\dot{z}_{p_o}$ are directly related to the initial state such that their partials are zeroes and ones.

Thrust: To permit the study of specific thruster systems, it may be desirable to fix thrust level while allowing power level and specific impulse to be optimized. Thus, thrust is placed in the list of available end conditions and is evaluated by the equation

$$|T|_o = 2\ \eta p_o / c$$

where $|T|_o$ represents a reference thrust level, which is the actual thrust for a nuclear electric propulsion system but is a maximum thrust at 1 AU for a solar electric system. The required partial derivatives are

$$\frac{\partial |T|_o}{\partial y_f} = \frac{\partial |T|_o}{\partial t_f} = \frac{\partial |T|_o}{\partial y_o} = 0$$

$$\frac{\partial |T|_o}{\partial \beta} = 2\,\frac{\eta}{c}\,\frac{\partial p_o}{\partial \beta} + 2\,\frac{p_o}{c}\left(\frac{d\eta}{dc} - \frac{1}{c}\right)\frac{\partial c}{\partial \beta}$$

with $$\frac{d\eta}{dc} = 2\,\frac{\eta}{c}\left(1 - \frac{\eta}{b}\right)$$

The derivative of η with respect to c comes from the assumed relationship between η and c stated in the section, Propulsion System Characteristics.

Heliocentric Position and Velocity: The final heliocentric position and velocity may be specified in terms of the Cartesian components of the two vectors or in terms of such position and velocity-related parameters as radial distance r, speed v, flight path angle γ, semi-major axis a, eccentricity e, aphelion distance r_a, and/or perihelion distance r_p. The Cartesian components of position and velocity are the state variables of the problem and are immediately available. The partial derivatives of the components are also immediately available from the matrix $\Phi(o, t)$. The other parameters and their derivatives are formed from the state variables as follows:

$$r = (R \cdot R)^{\frac{1}{2}}$$

$$\frac{\partial r}{\partial u} = \frac{u}{r} \qquad (u = x,\ y,\ z)$$

$$v = (\dot{R} \cdot \dot{R})^{\frac{1}{2}}$$

$$\frac{\partial v}{\partial \dot{u}} = \frac{\dot{u}}{v} \qquad (\dot{u} = \dot{x}, \dot{y}, \dot{z})$$

$$a = \frac{r}{2 - rv^2/\mu}$$

$$\frac{\partial a}{\partial u} = \frac{au}{r^2}\left(1 + \frac{av^2}{\mu}\right) ; \frac{\partial a}{\partial \dot{u}} = \frac{2a^2\dot{u}}{\mu} \quad (u = x, y, z ; \dot{u} = \dot{x}, \dot{y}, \dot{z})$$

$$\gamma = \sin^{-1}\left(\frac{R \cdot \dot{R}}{rv}\right)$$

$$\left.\begin{aligned}\frac{\partial \gamma}{\partial u} &= \frac{1}{r\cos\gamma}\left(\frac{\dot{u}}{v} - \frac{u}{r}\sin\gamma\right)\\ \frac{\partial \gamma}{\partial \dot{u}} &= \frac{1}{v\cos\gamma}\left(\frac{u}{r} - \frac{\dot{u}}{v}\sin\gamma\right)\end{aligned}\right\}(u = x, y, z;\ \dot{u} = \dot{x}, \dot{y}, \dot{z})$$

$$e = \left(1 - \frac{r}{a}\left(2 - \frac{r}{a}\right)\cos^2\gamma\right)^{\frac{1}{2}}$$

$$\left.\begin{aligned}\frac{\partial e}{\partial u} &= \frac{1}{e}\left[\frac{u}{r}\ \frac{v^2}{\mu}\left(1 - \frac{r}{a}\right)\cos^2\gamma + \frac{\sin\gamma}{a}\left(2 - \frac{r}{a}\right)\left(\frac{\dot{u}}{v} - \frac{u}{r}\sin\gamma\right)\right]\\ \frac{\partial e}{\partial \dot{u}} &= \frac{1}{e}\left[\frac{2r\dot{u}}{\mu}\left(1 - \frac{r}{a}\right)\cos^2\gamma + \frac{r\sin\gamma}{av}\left(2 - \frac{r}{a}\right)\left(\frac{u}{r} - \frac{\dot{u}}{v}\sin\gamma\right)\right]\end{aligned}\right\}\begin{pmatrix}u = x, y, z\\ \dot{u} = \dot{x}, \dot{y}, \dot{z}\end{pmatrix}$$

$$r_a = a\,(1 + e)$$

$$\frac{\partial r_a}{\partial u} = a\,\frac{\partial e}{\partial u} + (1 + e)\,\frac{\partial a}{\partial u} \qquad (u = x, y, z;\ \dot{x}, \dot{y}, \dot{z})$$

$$r_p = a\,(1 - e)$$

$$\frac{\partial r_p}{\partial u} = -\,a\,\frac{\partial e}{\partial u} + (1 - e)\,\frac{\partial a}{\partial u} \qquad (u = x, y, z;\ \dot{x}, \dot{y}, \dot{z})$$

The partials with respect to the independent parameters are then formed according to the general equation given at the start of this section, with u and $\dot{u}$ representing a portion of y_f. The partials with respect to t_f, β, and y_o are zero.

Planetocentric Position and Velocity: By input option, several planetocentric position and velocity-related parameters are available as end conditions. These include the Cartesian position coordinates x_T, y_T, and z_T, the distance, r_T, speed v_T, flight path angle γ_T, semi-major axis a_T, eccentricity e_T, apocenter distance r_{T_a}, and pericenter distance r_{T_p}. Let P_T, $\dot{P}_T$ be the heliocentric position and velocity of the target planet at the final time. Then the planetocentric position and velocity of the spacecraft at the final time are

$$R_T = R - P_T$$

$$\dot{R}_T = \dot{R} - \dot{P}_T$$

with components x_T, y_T, z_T and $\dot{x}_T$, $\dot{y}_T$, $\dot{z}_T$, respectively. Since both P_T and $\dot{P}_T$ are functions only of final time, the partial derivatives of R_T and $\dot{R}_T$ with respect to any independent parameter α are

$$\frac{\partial R_T}{\partial \alpha} = \frac{\partial R}{\partial \alpha} - \dot{P}_T \frac{\partial t_f}{\partial \alpha}$$

and

$$\frac{\partial \dot{R}_T}{\partial \alpha} = \frac{\partial \dot{R}}{\partial \alpha} - \ddot{P}_T \frac{\partial t_f}{\partial \alpha}$$

Consequently, the partials of $\dot{R}_T$ with respect to α will equal those of $\dot{R}$ except when α is the final time; for this latter case, an additional term is added to the heliocentric partials to obtain the planetocentric partials.

Then, employing the standard definitions

$$r_T = (R_T \cdot R_T)^{\frac{1}{2}}$$

$$v_T = (\dot{R}_T \cdot \dot{R}_T)^{\frac{1}{2}} = |\dot{R}_T(t_f)|$$

$$\gamma_T = \sin^{-1}\left(\frac{R_T \cdot \dot{R}_T}{r_T\, v_T}\right)$$

$$a_T = \frac{r_T}{(2 - r_T v_T^{\,2} / \mu_T)}$$

$$e_T = \left(1 - \frac{r_T}{a_T}\left(2 - \frac{r_T}{a_T}\right)\cos^2\gamma_T\right)^{\frac{1}{2}}$$

$$r_{T_a} = a_T(1 + e_T)$$

$$r_{T_p} = a_T(1 - e_T)$$

The partial derivatives with respect to x_T, y_T, z_T, $\dot{x}_T$, $\dot{y}_T$, $\dot{z}_T$ are identical to those for the heliocentric parameters except a subscript T is added to all symbols.

Due to the large sensitivities inherent in targeting on the planetocentric periapse distance r_{T_p}, it is advisable to transform any problem with this end condition to one with an alternate set of end conditions which are less sensitive. One such transformation is derived here.

Suppose that the trajectory integration is to be terminated at a specified time and that the spacecraft is to be at a specified distance r_T from the target planet on the approach leg of the planetocentric hyperbola with conditions consistent with a specified periapse distance. It is assumed that r_T is within the sphere of influence, but well-removed from the periapse point. A crucial aspect of the transformation presented below is that any trajectory neighboring the nominal and terminating at the same time and in the vicinity of the target planet will have nearly the same final planetocentric velocity as the nominal. That is, even for relatively large dispersions in the final planetocentric position, the final velocity remains nearly invariant. Thus, one may use the velocity from the nominal trajectory to calculate a new desired final position which will satisfy the periapse distance constraint as well as the constraint on r_T. Because one degree of freedom remains, this position vector is not unique. Since the effect of the choice of the specific position vector on performance is very small, an arbitrary choice is made which results in a posigrade trajectory with an ecliptic inclination of the approach hyperbola equal to the declination of the approach asymptote. Under these assumptions, the target position vector becomes

$$R_T = \frac{d}{v_T^2} \dot{R}_T - \frac{h}{v_T \left|\bar{k} \times \dot{R}_T\right|} (\bar{k} \times \dot{R}_T)$$

where

$$h = \sqrt{r_{T_p}^2 v_T^2 - 2\mu_T r_{T_p} \left(1 - \frac{r_{T_p}}{r_T}\right)}$$

$$d = -\sqrt{r_T^2 v_T^2 - h^2}$$

$$v_T = \left|\dot{R}_T\right|$$

and $\mathring{R}_T$ is the final planetocentric velocity on the current nominal trajectory, $\bar{k}$ is a unit vector in the direction of the ecliptic North Pole, and μ_T is the gravitational constant of the target planet. When employing this transformation, it is advisable to first target on a trajectory that terminates on the approach to the target planet, although it should not be necessary to terminate initially within the sphere of influence. It is recommended that the specified value of r_T be in the range $0.5\ r_{T_s} \leq r_T \leq r_{T_s}$ where r_{T_s} is the radius of the sphere of influence.

PART II

ARBITRARY SPACE TRAJECTORY OPTIMIZATION PROGRAM

(ASTOP)

USER'S MANUAL

Introduction

The effort under contract NAS5-11193 consisted of developing computer programs for the design, optimization and error analysis of low thrust interplanetary trajectories. Three computer programs were developed:

ASTEA - for the open-loop error analysis of arbitrary space trajectories (tailored for low thrust trajectories).

SWINGBY - for the optimization of low thrust swingby trajectories in the patched conic mode.

ASTOP - for n-body, low thrust trajectory parameter optimization.

These programs are documented in the final reports prepared for the present contract by Analytical Mechanics Associates, Inc. Part II of this report describes the ASTOP computer program and contains its basic operating instructions.

The ASTOP Computer Program

ASTOP is an acronym for Arbitrary Space Trajectory Optimization Program. ASTOP is designed to generate precision optimum interplanetary low-thrust trajectories incorporating realistic operational and hardware constraints. It is a derivative of the basic ITEM n-body interplanetary trajectory program.

ASTOP General Program Description

The ASTOP computer program generates optimum, precision low thrust interplanetary trajectories with provisions for realistic operational and hardware constraints. It is a derivative of the ITEM program which has been developed by Analytical Mechanics Associates, Inc. for the Goddard Space Flight Center over the past several years. For the most part, ASTOP was developed simply by adding features to the basic ITEM program; hence, ASTOP retains nearly all features available in ITEM. For details of the features and capabilities of ITEM the reader is referred to [2].

The ITEM program is a precision, n-body, numerical integration trajectory simulation program. It employs an Encke formulation of the equations of motion and integrates using a self-starting variable - order backwards-difference predictor-corrector technique. The independent variable of integration is the universal anomaly β defined implicitly by the equation

$$\beta = \sqrt{\mu}/r$$

where μ is the gravitational constant of the attracting body associated with the reference frame in which the integration is taking place and r is the distance of the spacecraft from that body. The program employs a sphere of influence concept to decide when to switch reference frames. The program employs an analytic planetary ephemeris routine.

To provide the desired features of ASTOP it was necessary to add a low thrust propulsion system model, modify the solar pressure computations to account for large solar cell arrays, incorporate selected spacecraft design parameters, add certain types of optional end conditions, introduce selected optional operational procedures and constraints and hardware constraints, and modify the iterator to include parameter optimization capability. The basic propulsion system model is consistent with the assumption of limited power with constant jet exhaust speed. The power output by the powerplant is permitted to vary with solar distance according to the equation

$$\gamma = \frac{1}{r^2} \sum_{i=0}^{4} a_i \, r^{-i/2}$$

where γ is the ratio of power at a distance r to power at 1 AU and a_i represents a set of input coefficients. The option of specifying power to be a constant is also provided.

The basic philosophy upon which ASTOP is built is that the performance that may be achieved by fully optimizing a trajectory subject to specified constraints is generally insensitive to the control variables in the vicinity of the optimum. Pursuing this thought, one might concede that a slowly varying control variable might be replaced over a time arc by a constant value with only a small loss in performance and, by choosing a sufficiently small time interval, even fairly large control variable rates might be accomodated with small performance penalties. The basic premise of ASTOP is that, for the low thrust interplanetary missions of current interest, the number of constant control arcs which are required to yield a performance near that of the variational optimum is quite small, possibly on the order of five or six. When one further considers the complexity of the hardware required to continually vary the thrust direction relative to the spacecraft, the simplicity of the approximate optimal constant control arcs becomes increasingly more interesting.

ASTOP is designed such that a complete mission trajectory is comprised of a series of constant control arcs. The number of arcs and their durations are input. The spacecraft orientation angles are the control parameters and, along with the arc end times, certain propulsion system and spacecraft design parameters, and selected initial conditions, are available for optimization. The spacecraft design assumes that the solar arrays (if any) are fixed relative to the thrust vector; consequently, a specified spacecraft orientation affects the performance in three ways: (1) the direction of the thrust vector; (2) the direction and magnitude of the solar pressure; and (3) the magnitude of the power generated due to the incidence angle of the sun's rays on the solar arrays.

Operational constraints are provided through an option which permits one to force the spacecraft orientation to follow a specified vector. This constraint may be employed to satisfy a requirement that an antenna point toward the Earth or that a sensor point toward a specified star or the sun. This feature is termed the Constrained Mode. There are two primary hardware constraints incorporated in the program. One is represented by the fact that the orientation of the arrays is expressly related to spacecraft orientation and, therefore, to thrust direction. This, of course, couples the effects of choice of thrust direction and the power generated as a function of array planform area exposed to the sun. The second hardware constraint is the option of imposing the condition that power input to a thruster be held constant even though the power generated by the powerplant may be variable. This is accomplished by assuming that individual thrusters are turned off as needed and that the fraction of a power unit left over is simply radiated in the form of heat.

To obtain reasonable initial conditions for launch from Earth orbit, an optional program feature is included in which the program commences by solving for a patched conic ballistic trajectory from Earth orbit to the prescribed destination. For this ballistic trajectory the orientation of and point of departure from the circular Earth parking orbit is determined, along with the launch energy requirement, c_{3b}. Holding the orientation of the parking orbit fixed, the point of departure from the orbit and the departure time is altered such that, for a given launch energy $c_3 < c_{3b}$, the resultant hyperbolic excess velocity is in the same direction as for the ballistic trajectory. In essence, the ballistic trajectory defines the ephemeris of the spacecraft in the launch parking orbit such that on each iteration the choice of a launch time results in a specific departure point from the parking orbit.

The program is designed such that thrusting is not permitted during the first arc following departure of the launch parking orbit. Solar pressure forces are also neglected during this arc; consequently, the spacecraft orientation has no effect on the motion and no provisions are made for orientation angle inputs on this first arc. The duration of the arc is input and may be made as long or short as desired.

The program will presently accommodate a maximum of 13 arcs, including the first in which no thrust is permitted. The maximum number of independent parameters presently is 19 while the maximum number of dependent parameters is 30. These limitations may be extended in the future by increasing the dimensions of certain arrays.

ASTOP Program Capabilities

The basic ASTOP program features and options, including the propulsion system model, constraint options, spacecraft design features, and available dependent and independent parameters are described in detail in Part I of this report. The inputs required for invoking and using these options and features are explained in the following section.

For the purpose of improving the parameter optimization capability, the iterator of the ITEM program was replaced with one very similar to that of the SWINGBY program discussed in Reference 3. With this iterator comes the capability of declaring any dependent parameter as the performance index. Although not yet activated, this iterator also possesses the capability of imposing interval constraints on any of the available dependent parameters.

ASTOP Input and Output

INPUT

Inputs to ASTOP are given through the namelist feature of the IBM Fortran IV programming language. The input namelist is named MINPUT, and every input required or used in the program is declared by the name in the list. The general form for assigning an input value to a quantity is, simply

NAME = VALUE

where NAME is the name assigned to the variable and is included in the namelist, and VALUE is a numerical or logical quantity consistent in form (i.e., logical, integer, or real) with NAME. Unless otherwise specified, all MINPUT names commencing with the letters I-N represent integers, whereas all names commencing with the letters A-H or O-Z are double precision floating point numbers. All input data sets must begin with the characters

&MINPUT

commencing in card column 2 and followed by at least one blank, and end with the characters

&END

preceded by at least one blank if data is contained on the same card. Card column 1 is ignored on all input cards. Multiple data assignments on a single card is permissible if separated by commas. A comma following the last VALUE on a card is optional. The order of the input data assignments is arbitrary; i.e., they need not be in the same order as listed in the namelist. In fact, there is no requirement that any specific input parameter be represented in the input data set. If no value is included in the inputs for a particular parameter, the default value, if any, is used. If there is no default value, the value used is that which happened to be in the particular core location assigned to the parameter at the time of execution. For other details regarding the namelist feature, the reader is referred to the IBM System/360 Fortran IV Language Manual.

After reading inputs, the routine MODIF (currently a dummy routine) is called and can be used for additional modifications.

NAME	DIMENSION	DESCRIPTION
NOPT	(72)	This variable contains many of the program option flags. Certain options pertain only to the ASTOP program. Other option flags of this array are identical to and compatible with the ITEM program and are dis-

NAME	DIMENSION	DESCRIPTION
NOPT (cont.)		cussed in [6]. Among these available options from ITEM are 2-16 and 40-46. Of these NOPT (2)-NOPT(13) and NOPT(40) - NOPT(46) permit the printing of various position, velocity and acceleration vector components and magnitudes in selected reference frames as follows:

2-Spacecraft position relative to current integration reference body, in km.
3-Spacecraft velocity relative to current integration reference body, in km/sec.
4-Spacecraft position relative to Earth, in km.
5-Spacecraft position relative to Moon, in km.
6-Moon position relative to Earth, in km.
7-Spacecraft position relative to Sun, in km.
8-Spacecraft position relative to Venus, in km.
9-Spacecraft position relative to Mars, in km.
10-Spacecraft position relative to Jupiter, in km.
11-Spacecraft position deviation from reference two body conic, in km.
12-Spacecraft velocity deviation from reference two body conic, in km/sec.
13-Non-two body accelerations acting on spacecraft, in km/sec^2.
40&41-Spacecraft position relative to Saturn, in km.
40&42-Spacecraft position relative to Uranus, in km.
40&43-Spacecraft position relative to Neptune, in km.
40&44-Spacecraft position relative to Pluto, in km.
40&45-Spacecraft position relative to Earth-Moon barycenter, in km.
40&46-Spacecraft position relative to Mercury, in km.

Setting the flag to 1 results in the corresponding vector being printed at initial time and at the end of each arc as specified by the ARCDTA and NARCS input parameters. Setting the flag to zero causes the print to be suppressed. Note that both NOPT(40) and the appropriate flag NOPT(41) - NOPT(46) must be set to 1 to obtain a printout of any of the last six position vectors. NOPT(14) and (15) are not used. NOPT(16) set to one deletes a message print after each rectification. The message printed for NOPT(16) set to zero includes the reason for rectification and the associated time and reference. Element 65 must contain

NAME	DIMENSION	DESCRIPTION
NOPT(cont.)		either the integer 1 or the integer 2. The integer 1 signifies the unconstrained flight mode; 2 signifies the constrained flight mode. If 1 is input in 65, then 63 and 64 are ignored. If 2 is input in 65, then 64 is expected to contain one of the integers 1, 2, or 3. The integer 1 in 64 signifies that the body-fixed $\bar{s}$-vector (Part I of this report) is constrained to continuously point toward a prescribed body in the solar system; 2 signifies that $\bar{s}$ is to be directed along an input vector; 3 signifies that $\bar{s}$ is to be directed along the heliocentric velocity vector. The integer in 63 indicates the body that the $\bar{s}$-vector is pointed to and is used only if a 1 is input in 64. The integers associated with the various bodies in the solar system are as follows:

1- Earth
2- Moon (not presently available)
3- Sun
4- Venus
5- Mars
6- Jupiter
7- Saturn
8- Uranus
9- Neptune
10- Pluto
11- Mercury (not presently available)

Element 66 must contain either the integer 0 signifying that power varies with solar distance or the integer 1 signifying the power is constant.

The following option flags when set to 1 trigger the associated conditions:

57- include retro maneuver at end. RTA and RTP must be input.

58- MINMX3 iterator starts in the optimize mode.

56- signifies there is an arbitrary celestial body designated as the target planet. For this option, the data TDATEX to EMUDD as described below must be input.

NAME	DIMENSION	DESCRIPTION
NOPT (cont.)		60- the ITEM printout for every trajectory will be given. This trigger is automatically set internally for the final trajectory. 68- Force the arrays to be normal to the sun-spacecraft line throughout the trajectory. i.e. ($\bar{e}_r \cdot \hat{n} = 1.$).
ER	(3)	This vector is included only if a 2 is input in element 64 of NOPT. It contains the x, y, and z components, respectively, of the unit vector along which $\bar{s}$ is to be directed.
IREFNB IREFNT		These two fields are the planet numbers of the launch and target planets, respectively. The code numbers for the individual planets are as follows: 1 - Earth 2 - Moon (not presently available) 3 - Sun 4 - Venus 5 - Mars 6 - Jupiter 7 - Saturn 8 - Uranus 9 - Neptune 10 - Pluto 11 - Mercury (not presently available)
STA	(3)	The three fields contain, respectively, the longitude in degrees, the latitude in degrees, and the altitude in kilometers of the launch pad. This launch site is used in conjunction with a specified launch profile (see XOPT below) in the patched conic ballistic trajectory iteration such that the final launch parking orbit is consistent with the input launch site and launch trajectory profile.
POS VEL	(3) (3)	The three fields of the first vector contain the geocentric x, y, and z components, respectively, of the spacecraft position at departure from the

NAME	DIMENSION	DESCRIPTION
VEL(cont.)	(3)	Earth's parking orbit in Earth radii. The second vector contains the geocentric x, y, and z components of the spacecraft velocity at departure of the parking orbit in Earth radii per hour. Both vectors are referred to the geocentric ecliptic coordinate system. These inputs are necessary only if the patched-conic ballistic trajectory routine, which yields initial conditions, is bypassed (see XOPT below).
XOPT		This field contains an option flag as follows: = 1 - Initial conditions, in the form of the definition of a launch parking orbit and epoch of that orbit, are to be obtained from a patched-conic ballistic trajectory to the desired destination given a launch date and flight time. May not be used unless the destination is a planet. = 2 - Initial conditions are input in the form of initial position and velocity vector POS and VEL.
XJLD		This field contains the Julian date (with leading 244 omitted) which, if the option flag, XOPT, is set to one, represents the launch date for the ballistic patched-conic trajectory or, if the option is set to two, represents the launch date corresponding to the initial conditions input in POS and VEL.
XINJ		This is the flight time of the ballistic trajectory in days and is used only if the option flag, XOPT, is set to one.
RBRE		This field contains the radius of the Earth's sphere of influence in Earth radii.
XKC		k_c = fraction of the ballistic c_3 to be used as the estimate of c_3 into the first low thrust trajectory. Used only if the option flag, XOPT, is set to one.

NAME	DIMENSION	DESCRIPTION
AL1 AL2 AL3		a_1, a_2, a_3 - coefficients representing the performance of a particular launch vehicle and used in evaluating the initial spacecraft mass as a function of the departure speed. a_1 and a_3 are in kilograms; a_2 is in meters per second.
RLP		$\lvert R_{po} \rvert$ - radius of the launch parking orbit, in Earth radii. Used only if the option flag, XOPT, is set to one.
TL, DL		t_L, θ_L - time in hours and travel angle in degrees, respectively, of the launch trajectory from the launch pad to insertion in the circular parking orbit. Used only if the option flag XOPT is set to one.
TIMEL		Total time interval in hours from XJLD over which ephemeris table for all planets in simulation is created at start of run.
ENPLAN		The flag defining the number of attracting bodies to be included in the simulation. Referring to the list of planet numbers in the description of IREFNB, the input number for this flag means that all bodies in the list whose planet number is less than or equal to this flag are included in the simulation. Thus, the input value of this flag should be greater than or equal to the target planet number. If the target is not a planet, the flag must be greater than or equal to 3.
RTA RTP		Apocenter and pericenter distances, respectively, of capture orbit about target planet, in planetary radii.
ARRAY	(8, 3, 12)	A table array of program integration intervals as a function of radial distance from current reference body. There can be up to 7 integration intervals as specified by the 8 radial points. Define the array as:

NAME	DIMENSION	DESCRIPTION
ARRAY(cont.)	(8,3,12)	ARRAY(J,K,L) L=1, 12 - corresponding planet reference. K=1 - radial value, in Earth radii when referring to Earth reference; in AU otherwise. =2 - integration interval, in (Earth radii)$^{\frac{1}{2}}$ when referring to Earth reference; in (AU)$^{\frac{1}{2}}$ otherwise. =3 - not used. J=1, 8 - up to 8 radii or 7 intervals. The integration interval ARRAY (J,2,L) applies between the two radial distances defined by ARRAY (J,1,L) and ARRAY (J+1,1,L).
POSRCS VELRCS		Tolerance values for the ratio of the Encke perturbations to the two body position and velocity beyond which rectification of the nominal two body orbit is performed before continuing the integration of the trajectory.
THTS		Maximum change permitted in the osculating eccentric anomaly beyond which rectification of the nominal two body orbit is performed before continuing the integration of the trajectory.
TSCL REKM XMDKM		Output constants to convert position and velocity to arbitrary output units. The internal units of position are earth radii for geocentric distance and AU for all other distances. The internal units of velocity are earth radii/hour for geocentric velocity and AU/hour for all other velocities. TSCL converts the time factor for the velocity. REKM and XMDKM convert earth radii and AU respectively.
ITMAX		Maximum number of iterations in each of the two modes, select and optimize, of the iterator. Variables TDATEX through EMUDD represent the osculating orbital elements and the gravitational constant of an artibtrary celestial body. They are used only if NOPT(56) is set to 1. The only restrictions on the orbital elements is that the eccentricity must be less than 1 (i.e., an elliptical orbit).

NAME	DIMENSION	DESCRIPTION
TDATEX		Date of perihelion passage (days) with leading 244 omitted.
SAI		Semi-major axis (AU)
ECI		Eccentricity
CNI		Inclination (degrees)
OMI		Longitude of node (degrees)
SOI		Argument of perihelion (degrees)
EMUDD		Gravitational constant (m^3/sec^2)
XDAR	(7)	Perturbations of state variables x, y, z, $\dot{x}$, $\dot{y}$, $\dot{z}$ and m, respectively, at the start of each arc except first. Used to evaluate state transition matrix for each arc. The appropriate units are AU for position, AU/hr for velocity and kilograms for mass. Only heliocentric position and velocity are used as state variables.
		This group contains propulsion system and spacecraft design parameters which are held fixed throughout the simulation. These parameters are:
NPWR		Number of coefficients used in polynomial expression for power as a function of solar distance. Not used if constant power option is in effect (NOPT(66) = 1)
AO	(10)	NPWR coefficients used in the polynomial for the power variation with solar distance. Not used if constant power option is in effect (NOPT(66)=1)
CA		c_a - fraction of photons incident on the solar arrays that are absorbed ($0 \le c_a \le 1$). The remainders of the photons are assumed to be reflected at an angle equal to the incidence angle.
BL DSQ		b, d^2 - coefficients used in the expression for electric propulsion system efficiency. b is dimensionless and d^2 has units of m^2/sec^2.

NAME	DIMENSION	DESCRIPTION	
CR		c_r -	jet exhaust speed of the retro stage in m/sec.
XKR		k_r -	tankage factor of the retro stage.
XKT		k_t -	tankage factor of the electric propulsion system.
XKST		k_{st} -	structure factor of the spacecraft.
APS		α_{ps} -	Specific propulsion system and power-plant mass in kilograms per watt.
XKS		k_s -	solar pressure acting on flat plate at a distance of 1 AU from the sun assuming all photons are absorbed, in newtons/m^2.
XKP		k_p -	surface area of solar arrays per unit reference power, in m^2/watt.
DP		Δa -	portion of spacecraft area independent of the reference power, in m^2.
DELP		Δp -	power of the individual thrusters, in watts. A non-zero value implies discrete steps in power consumed by electric propulsion system as distance from the sun varies.
NARCS		number of arcs comprising the complete trajectory, including the initial arc commencing at departure from the launch parking orbit during which no thrust is permitted.	

Certain information pertaining to the individual independent parameters is contained in a single array named BX.

BX(I, J) I = 1, ---, 4 J = 1, ---, 100	For each independent parameter, four pieces of information are required by the iterator. The subscript I relates to these four items; J relates to the individual independent parameters. Default values of all elements of BX are zero.

NAME	DIMENSION	DESCRIPTION
BX(I, J) (cont.)	BX(1, J)	Trigger indicating whether Jth parameter is to be an independent parameter in boundary value problem. BX(1, J) = 0. Not an independent parameter. = 1. Used as independent parameter.
	BX(2, J)	Perturbation increment used to compute partial derivatives. Used only if trigger is on. Units are same as that of the parameter.
	BX(3, J)	Maximum change to Jth parameter permitted in a single iteration. Should be a positive quantity. Used only if trigger is on. Units are same as that of the parameter.
	BX(4, J)	Weighting factor. Should be a positive quantity. A guideline for selecting these weights is to estimate the uncertainty in how well you think you know a given independent variable. Then set the weighting factor equal to the inverse square of the uncertainty, where the uncertainty is expressed in the same units as the input units of the variable. The smaller the value of the weighting factor, the more the importance given to the associated variable by the iterator.

The initial values for all independent parameters are input separately as described below. The specific independent parameters associated with the various values of J in the BX array are as follows:

NAME	DIMENSION	DESCRIPTION
BX	J	
	1	t_o - time of departure from Earth parking orbit in hours from the input Julian launch date (XJLD). The input value is assumed. Since analytic partials with respect to t_o, are employed, BX(2, 1), the increment, is ignored.
	2-5	delta increments in the parameters argument of pericenter (ω), longitude of ascending node

NAME	DIMENSION	DESCRIPTION
BX (cont.)	J 2-5 (cont.)	(Ω), inclination (i) and speed at pericenter (V_{po}), respectively. The units are degrees for the angles and Earth radii per hour for V_{po}. These are increments to the nominal values obtained from the position and velocity vectors and are therefore 0 on all nominal trajectories. No input value is required for these parameters. Since analytic partials are employed for these parameters, the perturbation increments are ignored. These parameters may not be used in conjunction with either J=1 or 6.
	6	$\lvert V_{po} \rvert$ - speed at departure from Earth parking orbit in Earth radii per hour. This parameter is not input but is computed from the VEL vector or the solution of the Lambert Problem. Not used if BX(1,5) is equal to one.
	7-9	x, y, and z components, respectively, of the geocentric position at departure from the Earth parking orbit, in Earth radii. May not be used as independent parameters if independent parameter 1 - 6 are used.
	10-12	x, y, and z components, respectively, of the geocentric velocity at departure from the Earth parking orbit, in Earth radii per hour. May not be used as independent variables if independent parameters 1 - 6 are used.
	14	Jet exhaust speed c of low thrust propulsion system in m/sec.
	15	Reference power p_o of low thrust propulsion system in watts.
	16-17	Angles α and β, respectively, in degrees, defining the direction of the unit vector $\bar{n}$, the normal to the solar cell array, relative to the body fixed coordinate system OIJK.
	18-19	Angles δ and ϵ, respectively, in degrees, defining the direction of the $\bar{s}$ vector relative to the body fixed coordinate system OIJK.

NAME	DIMENSION	DESCRIPTION
BX (cont.)	J (cont.)	
	20	End time of the first arc in hours.
		The remaining parameters are in groups of four for each arc commencing with the second arc. i pertains to the arc number where $i \geq 2$.
	4i+13 4i+14 4i+15	The first three parameters are the spacecraft orientation angles, in degrees. The orientation angles ξ, ν, ζ (unconstrained mode) or θ, ψ, ϕ (constrained mode) have identification numbers 4i+13, 4i+14, 4i+15, respectively. The mode is determined by option NOPT (65).
	4i+16	The last parameter for each arc is the time at the end of the arc in hours.

All input information pertinent to the individual dependent parameters is also contained in a single array named BY.

BY(K, L) K=1, ---, 5 L=1, ---, 50		For each dependent parameter, corresponding to a specific value of L, the iterator requires up to four input quantities corresponding to the four locations indicated by the subscript K. These inputs are:
	BY(1, L)	Trigger. If off (i.e., equal to zero), the parameter is ignored and is not considered a dependent parameter. Then the other three inputs pertaining to the Lth parameter need not be input. If trigger is on (i.e., not equal to zero), the Lth parameter is considered to be a dependent parameter or constraint.
	BY(2, L)	Desired value of the dependent parameter.
	BY(3, L)	Tolerance of convergence to desired value.
	BY(4, L)	Constraint type indicator, as follows:

NAME	DIMENSION	DESCRIPTION	
BY(K, L) (cont.)	BY(4, L) (cont.)	BY(4, L)=1.	Point, or normal, constraint. Iterator attempts to drive parameter to the center of the tolerance specified.
		=0.	Interval constraint. Once within the specified interval, the constraint is ignored unless the boundaries are subsequently violated.
		=-1.	Denotes performance index to be used in optimize mode. Only one such variable permitted in a case. Maximization or minimization is achieved by setting the desired value to an unattainably high (maximize) or low (minimize) value.
	BY(5, L)	not used	

The default values of all elements of the BY array are zero. Because it is frequently desirable to know the value of a function that is available as a dependent parameter, even though the function may not be constrained on a particular case, a complete set of dependent parameters is evaluated on all trajectories regardless of the trigger setting. That is, the trigger setting is used only to indicate to the iterator whether a particular dependent parameter is to be constrained, not whether it is to be evaluated.

The specific dependent parameters associated with the several possible values of L are as follows:

BY	L		
		1	Initial mass less low thrust propellant and retro propellant if any, in kilograms.
		2	Net spacecraft mass m_n in kilograms.
		3	Reference thrust $\|T\|_o$ in newtons.
		4	Heliocentric distance r in AU.

NAME	DIMENSION	DESCRIPTION	
BY (cont.)	L (cont.)	5	Heliocentric speed v in AU/hr.
		6	Helio. semi-major axis a in AU.
		7	Helio. flight path angle γ in degrees.
		8	Helio. eccentricity e.
		9	Helio. apocenter distance r_a in AU.
		10	Helio. pericenter distance r_p in AU.
		11	Planetocentric distance r_T in AU.
		12	Planetocentric speed v_T in AU/hr.
		13	Planeto. semi-major axis a_T in AU.
		14	Planeto. flight path angle γ_T in deg.
		15	Planeto. eccentricity e_T.
		16	Planeto. apocenter distance r_{Ta} in AU.
		17*	Planeto. pericenter distance T_p in AU.
		18 19 20	$\left.\begin{array}{l} x \\ y \\ z \end{array}\right\}$ Cartesian coordinates of vehicle with respect to target in AU.
		21-50	Not used.

The notation employed above for the end conditions is consistent with that used in Part I of this report. The reader is referred there for a mathematical description of each end condition.

The following group of inputs contain the independent parameters requiring initial values as well as the values of parameters which are permitted to change at the start of each arc.

---

* When this parameter is designated as an end condition, its trigger is automatically reset to zero as is that of r_T (L=11), and the triggers of the Cartesian coordinates (L=18-20) are set to one. The desired values of r_T and the tolerances and constraint type indicators of the Cartesian coordinates must be input. A flag is set internally to re-evaluate the desired values of the Cartesian coordinate on each trajectory.

NAME	DIMENSION	DESCRIPTION
CE		c - jet exhaust speed in meters per second.
P0		p_o - reference power in watts.
ALPHA BTA		α, β - direction angles of $\bar{n}$ in degrees.
DLTA EPSLON		δ, ϵ - direction angles of $\bar{s}$ in degrees.
		Following these six quantities are sets of data for each arc up to NARCS sets. There are seven parameters for each arc. They are stored in an array ARCDTA (7,20) for up to 20 arcs of data. The subscript i and corresponding index I refer to data for the i^{th} arc.
ARCDTA(1,I)		t_i - time at end of i^{th} arc in hours from the input Julian date.
ARCDTA(2,I)		T_i - trigger with possible settings of ± 1. A positive setting indicates that the i^{th} arc will be a thrusting arc, whereas a negative value indicates a coasting arc.
ARCDTA(3,I) ARCDTA(4,I) ARCDTA(5,I)		ξ, ν, ζ, or θ, ψ, ϕ - the appropriate set of three spacecraft orientation angles in degrees.
ARCDTA(6,I)		ψ_{max} - maximum permissible value of the angle ψ, in degrees. Used only if operating in the constrained flight mode.
ARCDTA(7,I)		p_x - increment of power, in watts, generated by the power source but not available for primary propulsion.
		For the first arc, only t_1 is needed.

Defaults

Following are the default values of all program inputs.

NAME	DIMENSION	VALUES
NOPT	(72)	72*0
ER	(3)	0., 0., 1.
IREFNB		1
IREFNT		6
STA	(3)	80.5, 28.5, 0.
POS	(3)	1.025, 0., 0.
VEL	(3)	0., 7., 0.
XOPT		2.
XJLD		0.
XINJ		600.
RBRE		123.4
XKC		.5
AL 1		138726.52
AL 2		3776.8656
AL 3		1999.2024
RLP		1.025
TL		.3333
DL		12.0
TIMEL		15000.
ENPLAN		6
RTA		38.
RTP		2.

NAME	DIMENSION	VALUES
ARRAY	(8, 3, 12)	ARRAY (1, 1, L) = 0. ARRAY (2, 1, L) = 400. ARRAY (1, 2, L) = .00390625 for L = 1 - 12
POSRCS		.0001
VELRCS		.0001
THTS		1.5
TSCL		3600.
REKM		6378.165
XMDKM		1.49598D8
ITMAX		50
TDATEX		0.
SAI		1.
ECI CNI OMI SOI EMUDD		0.
XDAR	(7)	3*.1D-8, 3*.1D-9, .1D-2
NPWR		5
AO		.627, 5.3054, -10.0376, 7.1073, -2.0021, 5*0.D0
CA		0.
BL		.769
DSQ		2.0449D8
CR		2941.995

NAME	DIMENSION	VALUES
XKR		.1111111
XKT		.03
XKST		0.
APS		.03
XKS		0.
XKP		0.
DP		0.
DELP		0.
NARCS		1
BX		400*0.
BY		250*0.
CE		3.D4
P0		5.D4
ALPHA		0.
BTA		0.
DLTA		0.
EPSLON		0.
ARCDTA		140*0.

OUTPUT

The printout of ASTOP may be divided conveniently into four groups as follows: (1) the namelist inputs, (2) the iteration summary, (3) the final trajectory summary, and (4) the case summary. The iteration summary is written for each nominal trajectory and contains independent and dependent parameter information, a spacecraft mass breakdown, and the partial derivative matrix. The final trajectory summary includes position and velocity data at the initial time and at the end of each arc. The case summary is a collection of trajectory, spacecraft and mission data that will usually be of interest to the mission analyst and is intended to serve as a brief, one-page description of the salient features of the case. A more detailed description of the ASTOP output is given in the following paragraphs.

To provide a means of verifying the inputs to the program, the input namelist MINPUT is printed at the start of each case. The format is standard namelist format as discussed above just preceding the definitions of inputs.

For each trajectory selected as a nominal and for which a partial derivative matrix is evaluated, the iteration summary is printed starting at the top of a new page for each iteration. Following a statement of the iteration number is written a block of parameters preceded with the heading "INDEPENDENT PARAMETERS." The block contains all parameters, each appropriately titled, that are available for selection as independent parameters of the boundary value problem regardless of whether they are actually so designated. Those which are employed as independent parameters in the specific case being considered are indicated with a double asterisk beside the title. The length of the block is variable because that depends on the number of trajectory arcs in the problem. The first line of the block contains seven parameters representing the initial state of the spacecraft (three components each of the initial position and velocity plus the initial mass). The second row contains the jet exhaust speed, c, and reference power p_0, the angles α and β defining the orientation of $\bar{n}$, the angles δ and ϵ defining the orientation of $\bar{s}$, the initial time t_0, and the time of the end of the first (coasting) arc. The next line(s) contains the three thrust angles and the end time for each of the arcs (exclusive of the first) comprising the trajectory. Each line contains information for two arcs. Following the arc data are the four parameters defining changes in the initial orientation of the launch parking orbit and the speed of departure from that orbit. These parameters are title "DLOMS", "DLOML", "DLINC", and "DLVPO" and represent changes from the previous nominal in argument of perigee, longitude of ascending node, inclination to the equator, and speed at periapse of the launch hyperbola, respectively. The final parameter in the independent parameters block is the speed at periapse of the launch hyperbola.

The next block of data consists of the 20 parameters which are presently available as end conditions. These parameters are all appropriately titled and are preceded by the heading "DEPENDENT PARAMETERS." Again, those parameters which are specified as end conditions of the case are indicated with a double asterisk beside the title. The parameters comprising the 20 available end conditions are the final spacecraft mass; the net spacecraft mass; the reference thrust; the final heliocentric distance and speed; the final heliocentric osculating semi-major axis, flight path angle, eccentricity, apocenter distance and perihelion distance; the target centered final distance and speed; the final planetocentric osculating semi-major axis, flight path angle, eccentricity, apocenter distance and pericenter distance; and the three final planetocentric ecliptic Cartesian components of distance.

Following the block of dependent parameters is a line stating the reference system in which the integration terminated, the value of the inhibitor used by the iterator, and a trajectory counter. This counter is the cumulative number of trajectories integrated exclusive of the nominal trajectories. This is followed by a spacecraft mass breakdown. This includes the initial mass, the low thrust propulsion, propellant, and tankage masses, the structural mass, the retro propellant and structure, and the net spacecraft mass. Finally, the iteration summary print is concluded with the partial derivative matrix. Each row of this matrix represents the partials of one of the specified end conditions with respect to all of the indicated independent parameters. The order of the partials reading across a given row is the same as that in which independent parameters appear in the first block of data of the iteration summary. Likewise, the order reading down the matrix is that in which the dependent parameters as they appear in the second data block. Additional self-explanatory messages from the iterator may follow the partial derivative matrix.

Any particular case may terminate for one of several reasons such as (1) the case is converged, (2) the case will not converge, (3) the maximum number of iterations specified is exceeded, or (4) the case times out. Regardless of the cause of the termination, a final trajectory summary is then printed. This summary consists of standard ITEM program print for the initial time and for the final times of the several arcs comprising the trajectory. This includes at each point the position and velocity data as requested through the input flags NOPT(2) - NOPT(16) plus a variety of osculating elements evaluated at the same times. The osculating elements are the true anomaly, semimajor axis, eccentricity, periapse distance, apoapse distance, inclination, argument of periapse, period, mean motion, right ascension of ascending node, mean anomaly, eccentric anomaly, and time since periapse passage. Each data point also includes the ecliptic Cartesian coordinate of a unit vector along the periapse direction and of another unit vector along the angular momentum.

After indicating the launch planet and target of the case mission, the case summary contains a statement as to whether the program is operating in the constrained or unconstrained mode for the case. If in the constrained mode, a message is printed specifying the reference to which $\bar{s}$ is constrained. For the case in which the reference is a star, the input unit vector defining the location of the star is printed. This is followed with a line of data specifying several of the input coefficients used for the case. Included are the coefficients describing the capability of the launch vehicle, the low thrust propulsion system efficiency coefficients, the photon absorption coefficient c_a, and the specific array area k_p. A spacecraft mass breakdown similar to that printed in the iteration summary is then written, followed by the angles α, β, δ, and ϵ which define the orientation of the $\bar{n}$ and $\bar{s}$ vectors in the body fixed coordinate system. The next line of data contains a number of low thrust propulsion system parameters including the reference power, the reference thrust, the jet exhaust speed, the efficiency, the unit thruster power Δp, and the array area. A trajectory schedule is then printed, giving pertinent information from the ARCDTA array relative to each arc. The first line of the block contains the end times of all arcs. Below this are written six lines of data containing the three thrust angles, the power diverted for non-propulsive uses p_x, the maximum allowable half-cone angle ψ_{max} (applicable only for constrained mode cases), and the thrust mode indicator "ON" or "OFF" defining the operational status of the propulsion system for the arc. These six quantities are positioned in separate columns for each arc, and the columns are located below and midway between the two times defining the start and end of the arc. If these are more than eight arcs, the format is repeated until all arcs are accounted for. No information is printed for the first arc since no thrust is permitted on the arc. After the trajectory schedule are written the date, the hyperbolic excess speed, and the energy parameter c_3 at departure of the launch planet and upon arrival at the target. If no retro maneuver is required, the departure and arrival conditions complete the case summary printout. If a retro maneuver is performed, however, then related parameters are printed on a single line. These include the retro stage propellant and structure, the periapse and apoapse distances of the final orbit, the specific impulse of the retro stage, the orbital velocity at periapse of the capture orbit, and the incremental velocity imparted by the retro stage. All units are explicitly indicated on the case summary page. Examples of all the above types of printout available with ASTOP are given in the section, <u>Example Problem</u>.

Brief Subroutine Descriptions

Excluding the IBM System/360 library routines, the ASTOP program is comprised of 56 subroutines and function subprograms, plus a BLOCK DATA subprogram. The latter is an IBM System/360 feature which permits the assignment of values to variables in common arrays through the use of DATA initialization statements. Many of the subroutines listed below are actually entry points within other subroutines. A brief description of the 56 routines is given below.

Identification	Purpose
AMAINT	Integration routine for second order differential equations. Uses a Runge-Kutta scheme to build up the tables and then a sixth order Lagrange interpolation scheme.
AMAL	Generates a matrix which will rotate any vector through a given angle around a given vector.
CONTRL	Control routine for trajectory. It tests for rectification, reference switching, and end of arcs. Calls GENMA at the end of each arc.
CROSP	General routine to take cross product of two vectors.
DATE1	Computes a Julian date given a calendar date.
DCUBIC	Computes roots of cubic equation. Called by SAMM.
DERIV	Derivative routine. Computes derivatives of the state variables.
DETDT	Find the proper integration interval from ARRAY block. Calls INTVEC.
DIAG1	Prints out trajectory summary output and through ENTRY SUMMRY prints out case summary output.
DOTP	Computes the dot product of two vectors.
ECLCOO	Computes the matrix which rotates from the equatorial coordinate system to the ecliptic coordinate system.
ELCO	Finds and prints the osculating elements of a trajectory for printout.

Identification	Purpose
ENDT	Fixes the time for switching references at earth and target.
EPH	Computes the analytical ephemeris for position and velocity of all pertinent bodies.
EPHEM1	Initializes ephemeris tables for solving the Lambert problem.
EPHEM	Builds up the ephemeris tables for the trajectory calculations.
FCOMP	Evaluates the Stumpf function for the two body solution.
FINDXB	Finds the body-fixed $\bar{s}$ vector when in the constrained mode.
FNCDT	Given a Julian date finds t_o for the trajectory.
FNMAT	Generates the matrix of partials for the iterator and computes the dependent parameters. Calls DIAG 1.
FNPRNT	Calls TRAJL to print arc data on final trajectory.
GENMA	Generates intermediate matrices to be used by FNMAT for the matrix of partials.
INIT	Initializes various things and calls SETI for additional initialization. Called by TRAJL.
INPUT	Reads in namelist data and converts to internal units. Calls LAMBRT if Lambert problem.
INT	Interpolates an ephemeris tables with respect to time.
INTVEC	Finds integration interval being called by DETDT.
INT1	Calls the ephemeris routine for solving the Lambert problem.
ITMAT	Initializes parameters for the iterator MINMX3 and calls MINMX3.
LAMBRT	Control routine for the Lambert problem. Calls LAMB1.
LAMB1	Computes the solution to the Lambert problem.

Identification	Purpose
MAIN	Driver routine. Calls INPUT and then calls ITMAT.
MIIP1	Prints arc summary for each trajectory.
MIIP2	Called if NC(17) = 1; gives intermediate, diagnostic printouts.
MINMX3	Generalized iterator routine. Basically, all decisions regarding the solution of the two point boundary value problem are made in this routine, including the calling of the trajectory generator routine, the evaluation of the partial derivative matrix, the selection of the changes in the independent parameters, and the printing of selected information on each iteration if requested.
MODIF	Dummy routine which can be used to modify input.
MTMT	Matrix - matrix multiply routine.
MTVT	Matrix - vector multiply routine.
PDATE	Given a Julian date, finds a calendar date.
POLCAR	Finds launch station coordinates given longitude, latitude, and altitude. Used in the Lambert problem.
PREEC	Finds the precession matrix.
PSICO	Used in the Lambert option. Computes the earth's rotation.
RADII	Computes the magnitude and its square and cube of a vector.
RECT1	Computes the proper masses called from TRAJL.
RECT	Calculates the new rectification parameters and initializes to continue integration.
REDUCE	Reduces an angle to lie between 0 and 2π radians.
REFSW	Reference switch routine used during Lambert problem.
SAMM	Computes a two-body solution and computes first partials of the state variables.

Identification	Purpose
SETI	Called by INIT. Takes solutions from MINMX3 and stores the new independent variables. Computes analytic partials. Calls SAMM.
SMQINT	Matrix inversion routine called by MINMX3.
SOLENG	Calculates the thrust term of the acceleration and calculates the mass derivative.
SUBFG	Called by SAMM and TBDP. Computes the G terms for the two body solution.
TBDP	Solves the two body problem. Called by DERIV.
TLKUP	Calls the interpolation routine, INT, to find the positions of the various planets.
TRAJL	Controls the integration for a single trajectory.
UNVEC	Unitizes a vector.
XYZ	Evaluates desired final planetocentric position vector for a specified final distance and periapse distance, given the final planetocentric velocity.

ASTOP Program Machine Requirements

When compiled by the IBM 360/Model 91 computer at the Goddard Space Flight Center under their Fortran H, Level 18 compiler with compiler optimization level equal to two, the ASTOP program occupies about 70000 (hexidecimal) bytes in core. This includes the core requirements for the following IBM library subroutines which must be accessible to the program:

IHCLASCN	IHCEFNTH
IHCLATN2	IHCLEXP
IHCLSCN	IHCLLOG
IHCLSQRT	IHCEFIOS
IHCFDXPD	IHCERRM
IHCNAMEL	IHCUOPT
IHCECOMH	IHCETRCH
IHCCOMH2	IHCUATBL
IHCFCVTH	REMTIM

The program is written entirely in double precision Fortran IV using the non-standard Fortran statement IMPLICIT REAL *8 (A-H, O-Z). This results in the assignment of an 8-byte word location to each real variable name commencing with the letters A-H or O-Z, unless the name is specifically declared to be of another type. An 8-byte word contains 15 hexadecimal digits. As in standard Fortran IV, names commencing with the letters I-N represent integer variables of 4-byte word length.

The only peripheral equipment referenced by the ASTOP program are the card reader, assigned to UNIT 5, the high-speed printer, assigned to UNIT 6, and and arbitrary output device, assigned to UNIT 11, used for remote terminal output at GSFC. No magnetic tapes are employed by the program for either input or output. Of course, temporary data storage assignments are made as required on the disk and drum storage areas. The load and execution step are approximately 500,000 bytes.

Example Case

The example case presented here is a 600 day (approximate) Earth-Jupiter flyby mission launched from a 1.025 Earth radii circular parking orbit inclined 28.5 degrees to the equator at 12 noon on November 26, 1980, using the SLV3C/Centaur/TE364-4 launch vehicle. A small solar electric propulsion system with a reference power of 1500 watts and a jet exhaust speed of 24000 m/sec is assumed. In addition, a tankage factor of 0.03 and a specific propulsion system mass of 0.03 kg/watt are specified. The constrained mode option is invoked such that the solar array is continuously oriented normal to the sun-spacecraft line with the thrust vector normal to the heliocentric radius vector (i.e. circumferential thrust). The trajectory is divided into three arcs - a 12-hour coasting arc following the assumed impulsive launch vehicle Earth departure maneuver, a long duration low thrust arc, and a final coast arc to the target. The required end conditions are that, at 13650 hours into the mission (568.75 days), the distance of the spacecraft from Jupiter must be 0.2 AU and the predicted (two-body) perijove distance at that time is to be 0.01 AU (about 21.4 Jupiter radii). A total of five parameters are left unspecified. These include the longitude of ascending node of the launch parking orbit, the argument of perigee of the escape hyperbola, the speed at departure of the launch parking orbit, the angle between the thrust vector and its projection in the ecliptic plane, and the end time of the low thrust arc. The performance index of the problem is net spacecraft mass.

The use of an iterative trajectory optimization program such as ASTOP is facilitated by good first guesses of the independent parameters. To obtain such guesses for this sample case, an optimum 600 day Earth-Jupiter flyby trajectory was generated using an available heliocentric two-body low thrust trajectory optimization program. This latter program has the capability of generating optimal trajectories subject to the fixed thrust angle constraint; therefore, the heliocentric two-body low thrust may be expected to possess most of the salient features of the n-body solution. The two-body program solution yields directly the burn time, a (variable) out-of-plane thrust angle, and the hyperbolic excess velocity. The burn time from this solution was about 5993 hours; therefore, the estimated input burn time for the check case was 6,000 hours. The optimal out-of-plane thrust angle varied over the range of -2 degrees at the initial time to -39 degrees at engine shutdown. The estimated constant value required by ASTOP was obtained simply by "eyeballing" a time-average value of the variable angle and slightly biasing that value to a smaller absolute value to account for the greater propulsive force at the smaller heliocentric distances. The value chosen for input was -10 degrees. The geocentric position and velocity at launch was computed using the hyperbolic excess velocity from the heliocentric two-body solution and the specified launch parking orbit radius and inclination, assuming the departure point coincides with perigee of the escape hyperbola. To do this requires the use of the vis-viva integral and conic equation for a hyperbolic trajectory in conjunction with standard spherical trigonometry equations. There will generally exist two equally valid solutions, one with a northerly heading and one with a southerly heading. One of these was arbitrarily selected and is represented by the two vectors POS and VEL contained in the input data set presented below. Using these inputs, a trajectory was integrated with ASTOP for the 600 day flight. This trajectory terminated surprisingly close to

Jupiter, slightly past the perijove point. The perijove distance was about 0.023 AU (the sphere of influence radius is 0.3 AU) compared to the desired value of 0.01 AU. Since the problem was to terminate at about 0.2 AU from Jupiter, the flight time was reduced from 600 days by an amount equal to the time spent on this trajectory from a distance of 0.2 AU on the approach leg to the final time. This time interval was found to be about 750 hours. Hence, the final flight time selected is 13650 hours. Note that this is not the time to closest approach.

The control cards and namelist data set required to generate the example case on the GSFC IBM 360/91 computer are reproduced below. The control cards cause the program to be loaded into the computer from magnetic tape no. TD 8090 where it resides in object module form. The data set contains all input data, exclusive of the default data, that are required to run the example case. The necessary control information for the independent parameters is contained in the BX array and for the dependent parameters in the BY array. The weights for the independent parameters were obtained as suggested in the earlier Inputs section using uncertainties of 5 degrees each for the longitude of node and argument of perigee, 0.01 km/sec for departure speed, 10 degrees for out-of-plane thrust angle and 1000 hours for burn time. Of the five independent parameters, only one - the out-of-plane thrust angle - uses the specified perturbation step size to form partial derivatives. The other four employ analytic partials in conjunction with the state transition matrix which is formed by finite differences using the perturbation step sizes input through XDAR. Note that the position of the BY array pertaining to the final planetocentric Cartesian position coordinates are initialized in anticipation of the automatic transformation of end conditions from the periapse distance to the final position vector. The settings of BTA, EPSLON, and NOPT (63) - NOPT (65) force the program to operate in the constrained mode with the arrays continually facing the sun squarely and the thrusters pointing normal to the sun-spacecraft line.

From the output, it is seen that the first nominal trajectory results in a perijove distance of about .023 AU which is very close to the desired value considering the approximate nature of the method used to obtain the first guesses. After five iterations, the message "ITERATOR IS NOW IN THE OPTIMIZE MODE" is printed. This signifies that a trajectory satisfying the end conditions to the specified tolerances has been found. At this point the iterator commences trying to improve the performance index. After sixteen more iterations, the converged case is printed. The machine time (CPU) required to generate this example case was 20 minutes on the IBM 360/91.

JOB CONTROL AND NAMELIST INPUT DATA
FOR JUPITER FLYBY MISSION

```
// JOB CARD
//GOSH  EXEC  LOADER,PARM='MAP,SIZE=490K,EP=MAIN',REGION=500K
//GO.SYSLIN    DD DSNAME=OBJECT,UNIT=2400-9,LABEL=(,NL),                X
//             DISP=(OLD,KEEP),DCB=(RECFM=FB,BLKSIZE=3200,LRECL=80),   X
//             VOL=SER=8090
//GO.FT11F001 DD SYSOUT=R,DCB=(RECFM=FB,LRECL=80,BLKSIZE=240)
//GO.DATA5 DD *
 &MINPUT
 BX(1,2)=3*1.D0,4.D-2,BX(1,3)=3*1.D0,4.D-2,BX(1,5)=2*1.D0,.1D0,1.D4
 BX(1,21)=1.D0,1.D-3,3.D0,1.D-2,BX(1,24)=1.D0,1.D0,2.4D2,1.D-6
 BY(1,2)=1.D0,5.D2,1.D0,-1.D9,BY(1,11)=1.D0,.2D0,1.D-6,1.D0
 BY(1,17)=1.D0,1.D-2,1.D-6,1.D0,BY(1,18)=0.D0,.1D0,1.D-8,1.D0
 BY(1,19)=0.D0,.1D0,1.D-8,1.D0,BY(1,20)=0.D0,.1D0,1.D-8,1.D0
 ARCDTA(1,1)=12.D0,ARCDTA(1,0)=6.D3,1.D0,-1.D1,ARCDTA(1,3)=1.365D4,-1.D0
 POS= .6469314278D0, .3806301922D0,-.69801531D0
 VEL=-5.54040599D0, 5.14887785D0,-2.3272332D0
 XDAR=3*1.D-9,3*1.D-10,1.D-3,CE=2.4D4,P0=1.5D3,BTA=270.D0,EPSLON=90.D0
 ARRAY(2,1,3)=1.D1,ARRAY(2,1,6)=2.D0
 AL1=153208.05D0,AL2=2385.1258D0,AL3=53.34D0,XJLD=4570.D0
 NOPT=2,1,1,NOPT(6)=1,NOPT(10)=1,1,1,NOPT(16)=1,NOPT(63)=3,1,2,NARCS=3
 &END
```

```
SMINPUT
BX= 0.0                     , 0.0                     , 0.0                     , 0.0                      , 1.000000000000000       ,
  1.000000000000000         , 1.000000000000000       , 0.3999999999999999D-01, 1.000000000000000          , 1.000000000000000       ,
  1.000000000000000         , 0.3999999999999999D-01, 0.0                     , 0.0                        , 0.0                     ,
 0.0                        , 1.000000000000000       , 1.000000000000000       , 0.1000000000000000D 00, 10000.00000000000       ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
  1.000000000000000         , 0.9999999999999999D-03, 3.000000000000000         , 0.9999999999999999D-02, 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 1.000000000000000       , 1.000000000000000          , 240.0000000000000       ,
 0.9999999999999999D-06, 0.0                          , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
 0.0                        , 0.0                     , 0.0                     , 0.0                      , 0.0                     ,
```

0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	BY=
0.0	0.0	0.0	0.0	0.0	
1.000000000000000	500.0000000000000	1.000000000000000	-100000000.000000	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
1.000000000000000	0.2000000000000000	0.9999999999999999D-06	1.000000000000000	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
1.000000000000000	0.9999999999999999D-02	0.9999999999999999D-06	1.000000000000000	0.0	
0.0	0.1000000000000000D 00	0.1000000000000000D-07	1.000000000000000	0.0	
0.0	0.1000000000000000D 00	0.1000000000000000D-07	1.000000000000000	0.0	
0.0	0.1000000000000000D 00	0.1000000000000000D-07	1.000000000000000	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	
0.0	0.0	0.0	0.0	0.0	

```
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,NC=
         0.        0.        0.        0.        0.        0.        0.        0.        0.        0.        0.
         0.        0.        0.        0.        0.        0.        0.        0.        0.        0.        0.
         0.        0.        0.        0.        0.        0.        0.        0.        0.        0.        0.
         0.        0.        0.        0.        0.        0.        0.        0.        0.        0.        0.
         0.        0.        0.        0.        0.        0.        0.        0.        0.        0.        0.
         0.        0.        0.        0.        0.        0.        0.        0.        0.        0.        0.
         0.        0.        0.        0.        0.        0.NOPT=        2.        1.        1.        0.
         0.        1.        0.        0.        0.        1.        1.        1.        0.        0.        0.
         1.        0.        0.        0.        0.        0.        0.        0.        0.        0.        0.
         0.        0.        0.        0.        0.        0.        0.        0.        0.        0.        0.
         0.        0.        0.        0.        0.        0.        0.        0.        0.        0.        0.
         0.        0.        0.        0.        0.        0.        0.        0.        0.        0.        0.
         0.        0.        0.        3.        1.        2.        0.        0.        0.        0.        0.
         0.        0.RTA=  38.00000000000000    ,RTP=  2.000000000000000    ,ER= 0.0                  , 0.0                      ,
 1.000000000000000       ,IREFNB=          1,IREFNT=          6,STA=  80.50000000000000    ,  28.50000000000000    ,
0.0                      ,POS= 0.6469314278000000    , 0.3806301922000000    ,-0.6980153100000000    ,VEL= -5.540405990000000    ,
 5.146877850000000       , -2.327233203000000    ,XJLD=  4570.000000000000    ,XINJ=  690.0000000000000    ,XOPT=
 2.000000000000000       ,RBPE=  123.4000000000000    ,XKC= 0.5000000000000000    ,AL1=  153208.0500000000    ,AL2=
 2365.125800000000       ,AL3=  53.34000000000000    ,RLP=  1.025000000000000    ,TL= 0.3333000000000000    ,DL=
 12.00000000000000       ,TIMEL=  15000.00000000000    ,ENPLAN=  5.000000000000000    ,RACDE= 0.0                  ,ARRAY=
0.0                      ,  400.0000000000000    , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.3906250000000000D-02, 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
 400.0000000000000       , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.3906250000000000D-02, 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      ,  10.00000000000000    ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.3906250000000000D-02, 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      ,  400.0000000000000    , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.3906250000000000D-02, 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      ,  400.0000000000000    , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.3906250000000000D-02,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      ,  2.000000000000000    , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.3906250000000000D-02, 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
 400.0000000000000       , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.3906250000000000D-02, 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      ,  400.0000000000000    ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.3906250000000000D-02, 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      ,  400.0000000000000    , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.3906250000000000D-02, 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.0                      ,
0.0                      , 0.0                      ,  400.0000000000000    , 0.0                      , 0.0                      ,
0.0                      , 0.0                      , 0.0                      , 0.0                      , 0.3906250000000000D-02,
```

```
0.0                  . 0.0                    . 0.0                    . 0.0                     . 0.0                    .
0.0                  . 0.0                    . 0.0                    . 0.0                     . 0.0                    .
0.0                  . 0.0                    . 0.0                    . 0.0                     . 0.0                    .
0.0                  .  400.000000000000      . 0.0                    . 0.0                     . 0.0                    .
0.0                  . 0.0                    . 0.0                    . 0.390625000000000D-02, 0.0                    .
0.0                  . 0.0                    . 0.0                    . 0.0                     . 0.0                    .
0.0                  . 0.0                    . 0.0                    . 0.0                     . 0.0                    .
0.0                  . 0.0                    . 0.0                    . 0.0                     . 0.0                    .
 400.000000000000    . 0.0                    . 0.0                    . 0.0                     . 0.0                    .
0.0                  . 0.0                    . 0.390625000000000D-02, 0.0                     . 0.0                    .
0.0                  . 0.0                    . 0.0                    . 0.0                     . 0.0                    .
0.0                  . 0.0                    . 0.0                    . 0.0                     . 0.0                    .
0.0                  . 0.0                    . 0.0                    .POSRCS= 0.999999999999999D-04,VELRCS=
0.399999999999999D-04,THTS=  1.50000000000000     ,TSCL=  3600.00000000000     ,REKM=  6378.16500000000     ,XMDKM=
 149598000.000000    ,XDAR= 0.100000000000000D-08, 0.100000000000000D-08, 0.100000000000000D-08, 0.100000000000000D-09,
0.100000000000000D-09, 0.100000000000000D-09, 0.999999999999999D-03,AD= 0.627000000000000     ,  5.30540000000000     ,
-10.0376000000000    ,  7.10730000000000      , -2.90210000000000      , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    ,CA= 0.0                  ,BL= 0.769000000000000     ,DSO=
 20449000.0000000    ,CR=  2941.99500000000     ,XKR= 0.111111000000000     ,XKT= 0.299999999999999D-01,XKST=
0.0                  ,APS= 0.299999999999999D-01,XKS= 0.0                  ,XKP= 0.0                         ,DP=
0.0                  ,DELP= 0.0                        ,TDATEX= 0.0                  ,SAI=  1.00000000000000     ,ECI=
0.0                  ,CNI= 0.0                        ,OMI= 0.0                  ,SOI= 0.0                         ,TPI=
0.0                  ,EMUDD= 0.0                      ,RADDCD= 0.0                  ,NPWR=              5,ITMAX=           50,NARCS=
          3,CE=  24000.0000000000     ,P0=  1500.00000000000     ,ALPHA= 0.0                     ,BTA=  270.000000000000     ,DLTA=
0.0                  ,EPSLON=  90.0000000000000     ,ARCDTA=  12.0000000000000     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
 6000.00000000000    ,  1.00000000000000      , -10.0000000000000      , 0.0                     , 0.0                    ,
0.0                  , 0.0                    ,  13650.0000000000      , -1.00000000000000       , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0                    , 0.0                     , 0.0                    ,
0.0                  , 0.0                    , 0.0
&END
```

FLAGS NC(1-72)

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	31	32	33	34	35	36
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
37	38	39	40	41	42	43	44	45	46	47	48	49	50	51	52	53	54	55	56	57	58	59	60	61	62	63	64	65	66	67	68	69	70	71	72
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0

OPTIONS NOPT(I)

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	31	32	33	34	35	36
2	1	1	0	0	1	0	0	0	1	1	1	0	0	0	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
37	38	39	40	41	42	43	44	45	46	47	48	49	50	51	52	53	54	55	56	57	58	59	60	61	62	63	64	65	66	67	68	69	70	71	72
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	3	1	2	0	0	0	0	0	0	0	0	0

TRAJECTORY SUMMARY ITERATION NO. 1

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS, USED IN THE ITERATICN

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.469314280-01	3.806301920-01	-6.980153100-01	-5.54040595D 00	5.14887785D 00	-2.32723320D 00	3.75587569D 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.40000000D 04	1.50000000D 03	0.0	0.0	2.70000000D 02	0.0	9.00000000D 01	1.20000000D 01
THETA **	**PSI**	**PHI**	**T2 ****	**THETA**	**PSI**	**PHI**	**T3**
-1.00000000D C1	C.0	0.0	6.00000000D 03	0.0	0.0	0.0	1.36500000D 04
DL ONS **	**DL OML ****	**DL INC**	**DL VPO ****	**VPOC**			
0.0	0.0	0.0	0.0	7.91347307D 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.45593553D 02	2.99685493D 02	7.09400505D-02	5.25530318D 00	3.03418253D-04	4.96608427D 00	5.21794081D 01	7.90741499D-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.89297319D 00	1.C39195350 00	2.01922322D-01	2.66538926D-C4	-7.41094887D-03	-8.19094602D 01	4.09678051D 00	-3.77719798D-02
PCENT.TRGT	**X TARGET ****	**Y TARGET ****	**Z TARGET ****				
2.29509920D-02	1.97583491D-01	3.25587105D-02	-2.59499370D-02				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 5.82076609D-11   TRAJ. COUNTER IS 1

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.75587569D 02	4.50000000D 01	3.00020158D 01	9.00060474D-01	0.0	0.0	0.0	2.99685493D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-4.07008D 00	-5.01896D 00	-3.22179D 02	1.75667D-01	-1.75829D-03
1.57257D 00	1.60522D 00	-3.64650D 00	-1.59808D-01	3.91029D-C6
-1.33112D 00	-1.94124D 00	4.83149D-01	-4.92996D-02	-1.95362D-05
3.97971D-01	-2.24544D-02	3.37744D-02	-3.07417D-01	3.72727D-06

TRAJECTORY SUMMARY ITERATION NO. 2

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS. USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.500396250-01	3.727521410-01	-6.993778060-01	-5.475374000 00	5.228217700 00	-2.302590550 00	3.757315740 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.400000000 04	1.500000000 03	0.0	0.0	2.700000000 02	0.0	9.000000000 01	1.200000000 01
THETA **	**PSI**	**PHI**	**T2 ****	**THETA**	**PSI**	**PHI**	**T3**
-1.242224790 01	0.0	0.0	5.971300920 03	0.0	0.0	0.0	1.365000000 04
DL OMS **	**DL OML ****	**DL INC**	**DL VPO ****	**VPOC**			
2.603476990-01	-9.658593890-01	0.0	-4.518967600-04	7.913021170 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.458313170 02	2.999343100 02	7.094005050-02	5.244378310 00	3.030826550-04	4.936933790 00	5.194566640 01	7.883616670-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.829023150 00	1.044844440 00	1.976515450-01	2.667828950-04	-7.408135500-03	-8.614271240 01	2.112554200 00	-2.305822330-02
PCENT.TRGT	**X TARGET ****	**Y TARGET ****	**Z TARGET ****				
8.241952270-03	1.873979850-01	6.186448160-02	-1.099611080-02				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 2.500000000-01   TRAJ. COUNTER IS 3

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.757315740 02	4.500003000 01	2.990025710 01	8.970077140-01	0.0	0.0	0.0	2.999343100 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.986520 00	-4.863390 00	-3.223320 02	2.112700-01	-1.770290-03
1.524450 00	1.504080 00	-3.628230 00	-1.941940-01	3.688910-06
-1.323190 00	-1.927610 00	4.971830-01	-5.992770-02	-1.958750-05
3.998250-01	-2.082700-02	4.705610-02	-3.023460-01	4.639780-06

TRAJECTORY SUMMARY ITERATION NO. 3

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS. USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.494853110D-01	3.723471760D-01	-7.001081340D-01	-5.471096600D 00	5.238659740D 00	-2.289355960D 00	3.756973290D 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.400000000D 04	1.500000000D 03	0.0	0.0	2.700000000D 02	0.0	9.000000000D 01	1.200000000D 01
THETA **	**PSI**	**PHI**	**T2 ****	**THETA**	**PSI**	**PHI**	**T3**
-1.327644680D 01	0.0	0.0	5.978968330D 03	0.0	0.0	0.0	1.365000000D 04
DL OMS **	**DL OML ****	**DL INC**	**DL VPO ****	**VPOC**			
1.407002640D-01	-1.825029220D-01	0.0	1.074492410D-04	7.913128620D 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.457868750D 02	2.998895610D 02	7.094005050D-02	5.240861360D 00	3.028199120D-04	4.923181860D 00	5.186232410D 01	7.875379250D-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.830374290D 00	1.045989430D 00	2.001289730D-01	2.663672710D-04	-7.426097920D-03	-8.562514680D 01	2.353655470D 00	-2.490457390D-02
PCENT.TRGT	**X TARGET ****	**Y TARGET ****	**Z TARGET ****				
1.005237600D-02	1.829968480D-01	6.558872030D-02	-5.400077190D-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 3.906250000D-03   TRAJ. COUNTER IS 5

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.756973290D 02	4.500000000D 01	2.991045420D 01	8.973136260D-01	0.0	0.0	0.0	2.998895610D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.09303D 00	-4.84923D 00	-3.22301D 02	2.24869D-01	-1.76816D-03
1.52843D 00	1.49265D 00	-3.62496D 00	-2.06410D-01	3.68293D-06
-1.32262D 00	-1.92572D 00	4.99116D-01	-6.38873D-02	-1.94715D-05
3.98808D-01	-2.10283D-02	5.23979D-02	-3.00916D-01	4.93738D-06

```
                              TRAJECTORY SUMMARY ITERATION NO.  4

                                  INDEPENDENT PARAMETERS

      ** NAME APPLIES TO PARAMETERS. USED IN THE ITERATION
 X (TO)          Y (TO)          Z (TO)          XDOT            YDOT            ZDOT            M (TO)
 6.495051190-01  3.723527530-01 -7.000867920-01 -5.47119704D 00  5.23842978D 00 -2.28975430D 00  3.75686984D 02

 C               PO              ALPHA           TO              BETA            DELTA           EPSLON          T1
 2.40000000D 04  1.50000000D 03  0.0             0.0             2.70000000D 02  0.0             9.00000000D 01  1.20000000D 01

 THETA     **    PSI             PHI             T2        **    THETA           PSI             PHI             T3
-1.32732917D 01  0.0             0.0             5.98063541D 03  0.0             0.0             0.0             1.36500000D 04

 DL OMS    **    DL OML    **    DL INC          DL VPO    **    VPOO
-4.12317913D-03  4.79962363D-03  0.0             3.24614130D-05  7.91316108D 00

                                  DEPENDENT PARAMETERS

 FINAL MASS      NET S/C   **    THRUST          DST.FR.SUN      VL.WRT.SUN      SM.AXIS SN      FL.P.A.SUN      ECCENT.SUN
 3.45773763D 02  2.99876356D 02  7.09400505D-02  5.24099868D 00  3.02834051D-04  4.92382831D 00  5.18653576D 01  7.87566950D-01

 APCEN. SUN      PRICEN.SUN      DST.FR.TRG      VL.WRT.TRG      SM.AX.TRGT      FL.P.A.TGT      ECCENT.TRG      APCNT.TRGT
 8.80167275D 00  1.04598387D 00  1.99999692D-01  2.66388139D-04 -7.42520438D-03 -8.56394460D 01  2.34639011D 00 -2.48476305D-02

 PCENT.TRGT      X TARGET  **    Y TARGET  **    Z TARGET  **
 9.99722171D-03  1.88690704D-01  6.55027835D-02 -5.43726955D-03

   TRAJECTORY TERMINATED IN JUPITER REFERENCE      INHIBITOR IS  6.10351563D-05     TRAJ. COUNTER IS     7

                                SPACECRAFT MASS BREAKDOWN (KG)

  INITIAL        PROPULSION      PROPELLANT      TANKAGE         STRUCTURE      RETRO PROP       RETRO STR        NET S/C
 3.75686984D 02  4.50000000D 01  2.99132310D 01  8.97396930D-01  0.0              0.0              0.0            2.99876356D 02

                                PARTIAL DERIVATIVE MATRIX BY ROWS


-3.99313D 00 -4.84975D 00 -3.22292D 02  2.24912D-01 -1.76749D-03

 1.52831D 00  1.49388D 00 -3.62498D 00 -2.06362D-01  3.68559D-06

-1.32262D 00 -1.92579D 00  4.99063D-01 -6.38770D-02 -1.94600D-05

 3.99851D-01 -2.10064D-02  5.23393D-02 -3.00970D-01  4.93338D-06
```

TRAJECTORY SUMMARY ITERATION NO. 5

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.495055800-01	3.723518740-01	-7.000868320-01	-5.47118957D 00	5.23843726D 00	-2.28975344D 00	3.75687131D 02	
C	P0	ALPHA	TO	BETA	DELTA	EPSLON	T1
2.40000000D 04	1.50000000D 03	0.0	0.0	2.70000000D 02	0.0	9.00000000D 01	1.20000000D 01
THETA **	PSI	PHI	T2 **	THETA	PSI	PHI	T3
-1.327322630 01	0.0	0.0	5.98061028D 03	0.0	0.0	0.0	1.36500000D 04
DL OMS **	DL OML **	DL INC	DL VPO **	VPOO			
7.678438630-06	-8.556051940-05	0.0	-4.602725810-07	7.91316062D 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.45773947D 02	2.99876551D 02	7.09400505D-02	5.24099734D 00	3.02833958D-04	4.92382326D 00	5.18653276D 01	7.87566654D-01
APCEN. SUN	PRICEN.SUN	DST.FR.TRG	VL.WRT.TRG	SM.AX.TRGT	FL.P.A.TGT	ECCENT.TRG	APCNT.TRGT
8.83166227D 00	1.04598425D 00	2.00000003D-01	2.66388091D-04	-7.42520632D-03	-8.56385823D 01	2.34677213D 00	-2.48504735D-02
PCENT.TRGT	X TARGET **	Y TARGET **	Z TARGET **				
1.00000609D-02	1.88890020D-01	6.55056738D-02	-5.43767698D-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE INHIBITOR IS 9.53674316D-07 TRAJ. COUNTER IS 9

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.75687131D 02	4.56000000D 01	2.99131836D 01	8.97395513D-01	0.0	0.0	0.0	2.99876551D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.99312D 00	-4.84973D 00	-3.22292D 02	2.24909D-01	-1.76750D-03
1.52530D 00	1.49387D 00	-3.62498D 00	-2.06361D-01	3.68553D-06
-1.32261D 00	-1.92878D 00	4.99064D-01	-6.38766D-02	-1.94601D-05
3.98852D-01	-2.10064D-02	5.23390D-02	-3.00970D-01	4.93340D-06

ITERATOR IS NOW IN OPTIMIZE MODE.

```
                                   TRAJECTORY SUMMARY ITERATION NO.  6

                                        INDEPENDENT PARAMETERS

         ** NAME APPLIES TO PARAMETERS. USED IN THE ITERATION
 X (TO)           Y (TO)           Z (TO)           XDOT             YDOT             ZDOT             M (TO)
 6.495055680-01  3.723518940-01 -7.000868330-01 -6.471189740 00  5.238437100 00 -2.289753440 00  3.756871270 02

 C                PO               ALPHA            TO               BETA             DELTA            EPSLON           T1
 2.400000000 04  1.500000000 03  0.0              0.0              2.700000000 02  0.0              9.000000000 01  1.200000000 01

 THETA    **      PSI              PHI              T2       **      THETA            PSI              PHI              T3
-1.327323000 01  0.0              0.0              5.980610950 03  0.0              0.0              0.0              1.365000000 04

 DL OMS   **      DL OML   **      DL INC           DL VPO   **      VPOO
 9.191787930-08  1.706269000-06  0.0              1.217583730-08  7.913160630 00

                                        DEPENDENT PARAMETERS

 FINAL MASS       NET S/C  **      THRUST           DST.FR.SUN       VL.WRT.SUN       SM.AXIS SN       FL.P.A.SUN       ECCENT.SUN
 3.457739420 02  2.998765460 02  7.094005050-02  5.240997360 00  3.028339600-04  4.923823360 00  5.186532820 01  7.875666600-01

 APCEN. SUN       PRICEN.SUN       DST.FR.TRG       VL.WRT.TRG       SM.AX.TRGT       FL.P.A.TGT       ECCENT.TRG       APCNT.TRGT
 8.801662470 00  1.045984240 00  2.000000000-01  2.663280910-04 -7.425206300-03 -8.563860130 01  2.346763740 00 -2.485041120-02

 PCENT.TRGT       X TARGET   **    Y TARGET   **    Z TARGET   **
 9.999998620-03  1.888500390-01  6.550561140-02 -5.437653320-03

      TRAJECTORY TERMINATED IN JUPITER REFERENCE      INHIBITOR IS  9.536743160-07     TRAJ. COUNTER IS   10

                                        SPACECRAFT MASS BREAKDOWN (KG)

  INITIAL         PROPULSION       PROPELLANT       TANKAGE          STRUCTURE       RETRO PROP       RETRO STR        NET S/C
 3.756871270 02  4.500000000 01  2.991318500 01  8.973955500-01  0.0               0.0              0.0              2.998765460 02

                                     PARTIAL DERIVATIVE MATRIX BY ROWS


-3.993120 00 -4.849730 00 -3.222920 02  2.249090-01 -1.767500-03

 1.528300 00  1.493870 00 -3.624980 00 -2.063610-01  3.685530-06

-1.322610 00 -1.925780 00  4.990640-01 -6.387660-02 -1.946010-05

 3.988520-01 -2.100640-02  5.233900-02 -3.009700-01  4.933400-06
```

TRAJECTORY SUMMARY ITERATION NO. 7

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.520368850-01	3.712295690-01	-6.983276420-01	-5.464546010 00	5.231070190 00	-2.321485710 00	3.757567440 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.400000000 04	1.500000000 03	0.0	0.0	2.700000000 02	0.0	9.000000000 01	1.200000000 01
THETA **	**PSI**	**PHI**	**T2 ****	**THETA**	**PSI**	**PHI**	**T3**
-1.374904920 01	0.0	0.0	5.964364190 03	0.0	0.0	0.0	1.365000000 04
DL OMS **	**DL OML ****	**DL INC**	**DL VP0 ****	**VP00**			
-3.374944600-01	2.528404360-01	0.0	-2.184296010-04	7.912942210 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.458703330 02	2.999737410 02	7.094005050-02	5.240888990 00	3.027761820-04	4.921981470 00	5.186568930 01	7.875823000-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.798446970 00	1.045515980 00	2.001233200-01	2.563623670-04	-7.426407240-03	-8.563259990 01	2.350246500 00	-2.488029480-02
PCENT.TRGT	**X TARGET ****	**Y TARGET ****	**Z TARGET ****				
1.002748040-02	1.890113990-01	6.552842460-02	-5.482684410-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE INHIBITOR IS 3.125000000-02 TRAJ. COUNTER IS 12

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.757567440 02	4.500000000 01	2.988641040 01	8.965923120-01	0.0	0.0	0.0	2.999737410 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.956300 00	-4.851860 00	-3.223630 02	2.313880-01	-1.773660-03
1.508500 00	1.495880 00	-3.623090 00	-2.122000-01	3.653720-06
-1.318180 00	-1.926340 00	4.983580-01	-6.544680-02	-1.952730-05
4.003580-01	-1.873460-02	5.083560-02	-3.002370-01	5.113170-06

TRAJECTORY SUMMARY ITERATION NO. 8

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS. USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.54184631D-01	3.70291653D-01	-6.96815290D-01	-5.45882112D 00	5.22480122D 00	-2.34836496D 00	3.75808319D 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.40000000D 04	1.50000000D 03	0.0	0.0	2.70000000D 02	0.0	9.00000000D 01	1.20000000D 01
THETA **	**PSI**	**PHI**	**T2 ****	**THETA**	**PSI**	**PHI**	**T3**
-1.41068975D 01	0.0	0.0	5.95224486D 03	0.0	0.0	0.0	1.36500000D 04
DL OMS **	**DL OML ****	**DL INC**	**DL VPO ****	**VPOO**			
-2.86491162D-01	2.14779587D-01	0.0	-1.61798870D-04	7.91278041D 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.45941922D 02	3.00045930D 02	7.09400505D-02	5.24093651D 00	3.02743195D-04	4.92112352D 00	5.18690024D 01	7.87623946D-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.79711824D 00	1.04512880D 00	2.00098905D-01	2.66362155D-04	-7.42648718D-03	-8.56431577D 01	2.34535757D 00	-2.48442551D-02
PCENT.TRGT	**X TARGET ****	**Y TARGET ****	**Z TARGET ****				
9.99128074D-03	1.89601924D-01	6.54575709D-02	-5.75766749D-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 3.12500000D-02   TRAJ. COUNTER IS 14

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.75808319D 02	4.50000000D 01	2.98663962D 01	8.95991885D-01	0.0	0.0	0.0	3.00045930D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.92430D 00	-4.85293D 00	-3.22415D 02	2.36251D-01	-1.77824D-03
1.49072D 00	1.49698D 00	-3.62180D 00	-2.16533D-01	3.63088D-06
-1.31449D 00	-1.92677D 00	4.97719D-01	-6.65803D-02	-1.95798D-05
4.01573D-01	-1.67662D-02	4.94123D-02	-2.99723D-01	5.24965D-06

TRAJECTORY SUMMARY ITERATION NO. 9

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.55942652D-01	3.69519767D-01	-6.95571253D-01	-5.45397484D 00	5.21985875D 00	-2.37021548D 00	3.75836629D 02	
C	P0	ALPHA	T0	BETA	DELTA	EPSLON	T1
2.40000000D 04	1.50000000D 03	0.0	0.0	2.70000000D 02	0.0	9.00000000D 01	1.20000000D 01
THETA **	PSI	PHI	T2 **	THETA	PSI	PHI	T3
-1.44218014D 01	0.0	0.0	5.94485253D 03	0.0	0.0	0.0	1.36500000D 04
DL OMS **	DL OML **	DL INC	DL VP0 **	VP00			
-2.33226070D-01	1.73340930D-01	0.0	-8.88050286D-05	7.91269160D 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.45982423D 02	3.00086796D 02	7.09400505D-02	5.24095483D 00	3.02718560D-04	4.92045282D 00	5.18709233D 01	7.87649221D-01
APCEN. SUN	PRICEN.SUN	DST.FR.TRG	VL.WRT.TRG	SM.AX.TRGT	FL.P.A.TGT	ECCENT.TRG	APCNT.TRGT
8.79604365D 00	1.04486199D 00	2.00084855D-01	2.66361211D-04	-7.42658245D-03	-8.56410019D 01	2.34615138D 00	-2.48504691D-02
PCENT.TRG.	X TARGET **	Y TARGET **	Z TARGET **				
9.93730418D-03	1.88990411D-01	6.54401620D-02	-5.84456572D-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 3.12500000D-02   TRAJ. COUNTER IS 16

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.75836629D 02	4.50000000D 01	2.98542061D 01	8.95626182D-01	0.0	0.0	0.0	3.00086796D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.89760D 00	-4.85301D 00	-3.22446D 02	2.40716D-01	-1.78104D-03
1.47540D 00	1.49711D 00	-3.62094D 00	-2.20409D-01	3.61319D-06
-1.31148D 00	-1.92712D 00	4.97225D-01	-6.76160D-02	-1.95983D-05
4.02507D-01	-1.50893D-02	4.84133D-02	-2.99320D-01	5.36232D-06

TRAJECTORY SUMMARY ITERATION NO. 10

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS. USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.58776315D-01	3.68285092D-01	-6.93545129D-01	-5.44597865D 00	5.21198149D 00	-2.40530371D 00	3.75872246D 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.40000000D 04	1.50000000D 03	0.0	0.0	2.70000000D 02	0.0	9.00000000D 01	1.20000000D 01
THETA **	**PSI**	**PHI**	**T2 ****	**THETA**	**PSI**	**PHI**	**T3**
-1.49389059D 01	0.0	0.0	5.93454127D 03	0.0	0.0	0.0	1.36500000D 04
DL OMS **	**DL OML ****	**DL INC**	**DL VPO ****	**VPOO**			
-3.75313112D-01	2.78151706D-01	0.0	-1.11718605D-04	7.91257988D 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.46034879D 02	3.00139758D 02	7.09400505D-02	5.24090918D 00	3.02674128D-04	4.91910504D 00	5.18720644D 01	7.87670376D-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.79373836D 00	1.04447173D 00	2.00149560D-01	2.66346015D-04	-7.42731452D-03	-8.56432567D 01	2.34557562D 00	-2.48486498D-02
PCENT.TRGT	**X TARGET ****	**Y TARGET ****	**Z TARGET ****				
9.99402075D-03	1.89062313D-01	6.54233230D-02	-5.92260277D-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 1.56250000D-02   TRAJ. COUNTER IS 19

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.75872246D 02	4.50000000D 01	2.98373672D 01	8.95121015D-01	0.0	0.0	0.0	3.00139758D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.85244D 00	-4.05113D 00	-3.22484D 02	2.48171D-01	-1.78497D-03
1.44881D 00	1.49573D 00	-3.62009D 00	-2.26825D-01	3.59029D-06
-1.30657D 00	-1.92755D 00	4.96429D-01	-6.93312D-02	-1.96228D-05
4.03794D-01	-1.22465D-02	4.69880D-02	-2.98675D-01	5.54904D-06

TRAJECTORY SUMMARY ITERATION NO. 11

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.605045230-01	3.675375330-01	-6.922968480-01	-5.440990980 00	5.207235740 00	-2.426624630 00	3.758868810 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.400000000 04	1.500000000 03	0.0	0.0	2.700000000 02	0.0	9.000000000 01	1.200000000 01
THETA **	**PSI**	**PHI**	**T2 ****	**THETA**	**PSI**	**PHI**	**T3**
-1.522219510 01	0.0	0.0	5.928587750 03	0.0	0.0	0.0	1.365000000 04
DL ONS **	**DL ONL ****	**DL INC**	**DL VPO ****	**VPOO**			
-2.283265680-01	1.688660620-01	0.0	-4.590097470-05	7.912533980 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.460593550 02	3.001645300 02	7.094005050-02	5.240976900 00	3.026576610-04	4.918754890 00	5.187489830 01	7.877035800-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.793275730 00	1.044234050 00	2.000851290-01	2.663540610-04	-7.427010010-03	-8.564301650 01	2.345161690 00	-2.484454940-02
PCENT.TRGT	**X TARGET ****	**Y TARGET ****	**Z TARGET ****				
9.990529340-03	1.890006660-01	6.538503820-02	-6.132379360-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 1.562500000-02   TRAJ. COUNTER IS 21

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.758868810 02	4.500000000 01	2.982752530 01	8.948257600-01	0.0	0.0	0.0	3.001645300 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.823120 00	-4.848160 00	-3.225010 02	2.522720-01	-1.787210-03
1.431080 00	1.493540 00	-3.620290 00	-2.303280-01	3.574490-06
-1.303660 00	-1.927820 00	4.959560-01	-7.023650-02	-1.963340-05
4.044810-01	-1.043120-02	4.607310-02	-2.983410-01	5.649910-06

TRAJECTORY SUMMARY ITERATION NO. 12

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.62464707D-01	3.66707632D-01	-6.90862514D-01	-5.43536391D 00	5.20176321D 00	-2.45086463D 00	3.75885575D 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.40000000D 04	1.50000000D 03	0.0	0.0	2.70000000D 02	0.0	9.00000000D 01	1.20000000D 01
THETA **	**PSI**	**PHI**	**T2 ****	**THETA**	**PSI**	**PHI**	**T3**
-1.55848943D (	0.0	0.0	5.92150106D 03	0.0	0.0	0.0	1.36500000D 04
DL OMS **	**DL OML ****	**DL INC**	**DL VPO ****	**VPOO**			
-2.60134388D-01	1.93250442D-01	0.0	4.09391898D-06	7.91253808D 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.46069715D 02	3.00175241D 02	7.09400505D-02	5.24096527D 00	3.02629234D-04	4.91792420D 00	5.18761383D 01	7.87722504D-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.79188378D 00	1.04396463D 00	2.00100019D-01	2.66348582D-04	-7.42729722D-03	-8.56384279D 01	2.34724502D 00	-2.48609836D-02
PCENT.TRGT	**X TARGET ****	**Y TARGET ****	**Z TARGET ****				
1.00063892D-02	1.89015762D-01	6.53843399D-02	-6.16015792D-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 7.81250000D-03   TRAJ. COUNTER IS 24

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.75885575D 02	4.50000000D 01	2.98158589D 01	8.94475768D-01	0.0	0.0	0.0	3.00175241D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.78796D 00	-4.84287D 00	-3.22506D 02	2.57594D-01	-1.78992D-03
1.40960D 00	1.48960D 00	-3.62115D 00	-2.34927D-01	3.55824D-06
-1.30027D 00	-1.92806D 00	4.95418D-01	-7.14407D-02	-1.96494D-05
4.05052D-01	-8.24551D-03	4.53182D-02	-2.97894D-01	5.78180D-06

TRAJECTORY SUMMARY ITERATION NO. 13

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)
6.62479995D-01	3.66701178D-01	-6.90851280D-01	-5.43531962D 00	5.20172083D 00	-2.45105339D 00	3.75885519D 02

C	PG	ALPHA	TO	BETA	DELTA	EPSLON	T1
2.40000000D 04	1.56000000D 03	0.0	0.0	2.70000000D 02	0.0	9.00000000D 01	1.20000000D 01

THETA **	PSI	PHI	T2 **	THETA	PSI	PHI	T3
-1.35759135D 01	0.0	0.0	5.92145287D 03	0.0	0.0	0.0	1.36500000D 04

DL OMS **	DL OML **	DL .NC	DL VPO **	VPOO
-2.02730521D-03	1.50384017D-03	0.0	1.76495883D-07	7.91253825D 00

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.46069791D 02	3.00175319D 02	7.09400505D-02	5.24101064D 00	3.02633639D-04	4.91812964D 00	5.18771923D 01	7.87732675D-01

APCEN. SUN	PRICEN.SUN	DST.FR.TRG	VL.WRT.TRG	SM.AX.TRGT	FL.P.A.TGT	ECCENT.TRG	APCNT.TRGT
6.79230106D 00	1.04395822D 00	2.00051325D-01	2.60356547D-04	-7.42695426D-03	-8.56380704D 01	2.34703643D 00	-2.48582864D-02

PCENT.TRGT	X TARGET **	Y TARGET **	Z TARGET **
1.00043770D-02	1.88967305D-01	6.53695633D-02	-6.22178227D-03

TRAJECTORY TERMINATED IN JUPITER REFERENCE INHIBITOR IS 3.20000000D 01 TRAJ. COUNTER IS 30

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.75885519D 02	4.50000000D 01	2.98157283D 01	8.94471848D-01	0.0	0.0	0.0	3.00175319D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.78766D 00	-4.84277D 00	-3.22506D 02	2.57454D-01	-1.78993D-03
1.40940D 00	1.48954D 00	-3.62120D 00	-2.34799D-01	3.55816D-06
-1.30026D 00	-1.92807D 00	4.95409D-01	-7.13985D-02	-1.96506D-05
4.05066D-01	-8.23465D-03	4.52591D-02	-2.97902D-01	5.77858D-06

TRAJECTORY SUMMARY ITERATION NO. 14

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.624759610-01	3.666991300-01	-6.908562350-01	-5.435303370 00	5.201782210 00	-2.450971330 00	3.758843150 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.400000000 04	1.500000000 03	0.0	0.0	2.700000000 02	0.0	9.000000000 01	1.200000000 01
THETA **	**PSI**	**PHI**	**T2 ****	**THETA**	**PSI**	**PHI**	**T3**
-1.558197450 01	0.0	0.0	5.921668540 03	0.0	0.0	0.0	1.365000000 04
DL OMS **	**DL OML ****	**DL INC**	**DL VPO ****	**VPOC**			
8.941021280-04	-1.086509230-03	0.0	3.776598300-06	7.912542030 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN *	FL.P.A.SUN	ECCENT.SUN
3.460682430 02	3.001737610 02	7.094005050-02	5.240996750 00	3.026329550-04	4.918085660 00	5.187683990 01	7.877291130-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.792294910 00	1.043966410 00	2.000586740-01	2.663553300-04	-7.427006920-03	-8.563271120 01	2.349447320 00	-2.487636840-02
PCENT.TRGT	**X TARGET ****	**Y TARGET ****	**Z TARGET ****				
1.002235460-02	1.889691680-01	6.538996890-02	-6.186979590-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 8.000000000 00   TRAJ. COUNTER IS 33

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.758843150 02	4.500000000 01	2.981607200 01	8.944821600-01	0.0	0.0	0.0	3.001737610 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.787760 00	-4.842750 00	-3.225050 02	2.576450-01	-1.789850-03
1.409450 00	1.489510 00	-3.621170 00	-2.348720-01	3.558390-06
-1.300260 00	-1.928060 00	4.954190-01	-7.142320-02	-1.964860-05
4.050650-01	-8.237610-03	4.529030-02	-2.978970-01	5.780060-06

TRAJECTORY SUMMARY ITERATION NO. 15

INDEPENDENT PARAMETERS

```
          ** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION
 X (TO)           Y (TO)           Z (TO)           XDOT             YDOT             ZDOT             M (TO)
 6.624732210-01   3.667014340-01  -6.908576400-01  -5.435324320 00   5.201778730 00  -2.450949370 00   3.758826250 02

 C                PO               ALPHA            TO               BETA             DELTA            EPSLON           T1
 2.400000000 04   1.500000000 03   0.0              0.0              2.700000000 02   0.0              9.000000000 01   1.200000000 01

 THETA     **     PSI              PHI              T2        **     THETA            PSI              PHI              T3
-1.558287960 01   0.0              0.0              5.921929400 03   0.0              0.0              0.0              1.365000000 04

 DL OMS    **     DL OML    **     DL INC           DL VPO    **     VPOC
 2.535955700-04  -5.870227520-05   0.0              5.299924350-06   7.912547330 00
```

DEPENDENT PARAMETERS

```
 FINAL MASS       NET S/C   **     THRUST           DST.FR.SUN       VL.WRT.SUN       SM.AXIS SN       FL.P.A.SUN       ECCENT.SUN
 3.450661110 02   3.001716150 02   7.094005050-02   5.241004320 00   3.026339510-04   4.918127400 00   5.187698970 01   7.877304800-01

 APCEN. SUN       PRICEN.SUN       DST.FR.TRG       VL.WRT.TRG       SM.AX.TRGT       FL.P.A.TGT       ECCENT.TRG       APCNT.TRGT
 8.792286250 00   1.043963540 00   2.000511080-01   2.663565350-04  -7.426955540-03  -8.563333000 01   2.349117890 00  -2.487374970-02

 PCENT.TRGT       X TARGET  **     Y TARGET  **     Z TARGET  **
 1.001983860-02   1.889629710-01   6.538595500-02  -6.174018120-03

    TRAJECTORY TERMINATED IN JUPITER REFERENCE      INHIBITOR IS  2.000000000 00     TRAJ. COUNTER IS   36
```

SPACECRAFT MASS BREAKDOWN (KG)

```
 INITIAL          PROPULSION       PROPELLANT       TANKAGE          STRUCTURE       RETRO PROP       RETRO STR        NET S/C
 3.758826250 02   4.500000000 01   2.981651430 01   8.944954290-01   0.0              0.0              0.0             3.001716150 02
```

PARTIAL DERIVATIVE MATRIX BY ROWS

```
-3.787860 00 -4.842800 00 -3.225040 02  2.575900-01 -1.789740-03

 1.409490 00  1.489520 00 -3.621160 00 -2.348990-01  3.558810-06

-1.300270 00 -1.928070 00  4.954170-01 -7.143300-02 -1.964660-05

 4.050640-01 -8.239160-03  4.530070-02 -2.979000-01  5.780230-06
```

TRAJECTORY SUMMARY ITERATION NO. 16

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.624759460D-01	3.667613040D-01	-6.908551920D-01	-5.435326320D 00	5.201760590D 00	-2.450991240D 00	3.758818520D 02	
C	PO	ALPHA	TO	BETA	DELTA	EPSLON	T1
2.400000000D 04	1.500000000D 03	0.0	0.0	2.700000000D 02	0.0	9.000000000D 01	1.200000000D 01
THETA **	PSI	PHI	T2 **	THETA	PSI	PHI	T3
-1.558410750D 01	0.0	0.0	5.921934470D 03	0.0	0.0	0.0	1.365000000D 04
DL OMS **	DL OML **	DL INC	DL VPC **	VPOO			
-4.417829720D-04	4.380672580D-04	0.0	2.424980110D-06	7.912549750D 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.460653300D 02	3.001708340D 02	7.094005050D-02	5.241008180D 00	3.026342330D-04	4.918142240D 00	5.187708260D 01	7.877313970D-01
APCEN. SUN	PRICEN.SUN	DST.FR.TRG	VL.WRT.TRG	SM.AX.TRGT	FL.P.A.TGT	ECCENT.TRG	APCNT.TRGT
8.792317300D 00	1.043967180D 00	2.000472970D-01	2.663571290D-04	-7.426930480D-03	-8.563360020D 01	2.348969990D 00	-2.487256730D-02
PCENT.TRGT	X TARGET **	Y TARGET **	Z TARGET **				
1.001870630D-02	1.889597680D-01	6.538386450D-02	-6.170686220D-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE INHIBITOR IS 5.000000000D-01 TRAJ. COUNTER IS 39

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.758818520D 02	4.500000000D 01	2.981652220D 01	8.944956650D-01	0.0	0.0	0.0	3.001708340D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.787820D 00	-4.842810D 00	-3.225030D 02	2.576100D-01	-1.789740D-03
1.409460D 00	1.489530D 00	-3.621170D 00	-2.349160D-01	3.558820D-06
-1.300260D 00	-1.928070D 00	4.954140D-01	-7.143790D-02	-1.564640D-05
4.050650D-01	-6.235450D-03	4.530210D-02	-2.978990D-01	5.780650D-06

TRAJECTORY SUMMARY ITERATION NO. 17

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.62493341D-01	3.66697384D-01	-6.90843372D-01	-5.43530363D 00	5.20169377D 00	-2.45119044D 00	3.75881156D 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.40000000D 04	1.50000000D 03	0.0	0.0	2.70000000D 02	0.0	9.00000000D 01	1.20000000D 01
THETA **	**PSI**	**PHI**	**T2 ****	**THETA**	**PSI**	**PHI**	**T3**
-1.55874862D 01	0.0	0.0	5.92154481D 03	0.0	0.0	0.0	1.36500000D 04
DL OMS **	**DL OML ****	**DL INC**	**DL VPO ****	**VPCC**			
-2.13291948D-03	1.83038575D-03	0.0	2.19427532D-06	7.91255194D 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.46065295D 02	3.00170819D 02	7.09400505D-02	5.24100855D 00	3.02633474D-04	4.91812125D 00	5.18771461D 01	7.87732225D-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.79228394D 00	1.04395565D 00	2.00046984D-01	2.66357101D-04	-7.42693301D-03	-8.56334796D 01	2.34901950D 00	-2.48729435D-02
PCENT.TRGT	**X TARGET ****	**Y TARGET ****	**Z TARGET ****				
1.00190775D-02	1.86959540D-01	6.53835685D-02	-6.17067337D-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 1.25000000D-01   TRAJ. COUNTER IS 42

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.75881156D 02	4.50000000D 01	2.98158607D 01	8.94475822D-01	0.0	0.0	0.0	3.00170819D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.78750D 00	-4.84276D 00	-3.22503D 02	2.57641D-01	-1.78989D-03
1.40930D 00	1.48951D 00	-3.62118D 00	-2.34960D-01	3.55818D-06
-1.30024D 00	-1.92808D 00	4.95408D-01	-7.14490D-02	-1.96489D-05
4.05066D-01	-8.21792D-03	4.52976D-02	-2.97889D-01	5.78252D-06

TRAJECTORY SUMMARY ITERATION NO. 18

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS. USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.625449890-01	3.666830120-01	-6.907985920-01	-5.435221110 00	5.201436620 00	-2.451944890 00	3.758785980 02	

C	PO	ALPHA	TO	BETA	DELTA	EPSLON	T1
2.40000000D 04	1.50000000D 03	0.0	0.0	2.70000000D 02	0.0	9.00000000D 01	1.20000000D 01

THETA **	PSI	PHI	T2 **	THETA	PSI	PHI	T3
-1.56002035D 01	0.0	0.0	5.91996960D 03	0.0	0.0	0.0	1.36500000D 04

DL OMS **	DL OML **	DL INC	DL VPO **	VPOC
-8.07901540D-03	6.97363813D-03	0.0	8.02223774D-06	7.91255996D 00

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.46065411D 02	3.00171016D 02	7.09400505D-02	5.24100863D 00	3.02630319D-04	4.91803146D 00	5.18773684D 01	7.87735225D-01

APCEN. SUN	PRICEN.SUN	DST.FR.TRG	VL.WRT.TRG	SM.AX.TRGT	FL.P.A.TGT	ECCENT.TRG	APCNT.TRGT
8.79213809D 00	1.04392494D 00	2.00047250D-01	2.66356752D-04	-7.42695322D-03	-8.56330136D 01	2.34922176D 00	-2.48745133D-02

PCENT.TRGT	X TARGET **	Y TARGET **	Z TARGET **
1.00206069D-02	1.88960053D-01	6.53828046D-02	-6.17164780D-03

TRAJECTORY TERMINATED IN JUPITER REFERENCE INHIBITOR IS 3.12500000D-02 TRAJ. COUNTER IS 45

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.75878598D 02	4.50000000D 01	2.98131867D 01	8.94395602D-01	0.0	0.0	0.0	3.00171016D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.78627D 00	-4.84256D 00	-3.22502D 02	2.57752D-01	-1.79050D-03
1.40869D 00	1.48944D 00	-3.62124D 00	-2.35127D-01	3.55563D-06
-1.30014D 00	-1.92810D 00	4.95384D-01	-7.14854D-02	-1.96592D-05
4.05070D-01	-8.15163D-03	4.52801D-02	-2.97846D-01	5.78976D-06

TRAJECTORY SUMMARY ITERATION NO. 19

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION

X (TO)	Y (TC)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.62709189D-01	3.66647839D-01	-6.90659744D-01	-5.43504186D 00	5.20056755D 00	-2.45428426D 00	3.75868747D 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.40000000D 04	1.50000000D 03	0.0	0.0	2.70000000D 02	0.0	9.00000000D 01	1.20000000D 01
THETA **	**PSI**	**PHI**	**T2 ****	**THETA**	**PSI**	**PHI**	**T3**
-1.56407489D 01	0.C	0.0	5.91396364D 03	0.0	0.0	0.0	1.36500000D 04
DL OMS **	**DL OML ****	**DL INC**	**DL VPO ****	**VPOC**			
-2.50345003D-02	2.24131091D-02	0.0	3.08966792D-05	7.91259086D 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.46065772D 02	3.00171683D 02	7.09400505D-02	5.24100868D 00	3.02618600D-04	4.91769764D 00	5.18781961D 01	7.87746393D-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.79159622D 00	1.04379906D 00	2.00048586D-01	2.66355409D-04	-7.42702994D-03	-8.56313481D 01	2.34994717D 00	-2.48801580D-02
PCENT.TRGT	**X TARGET ****	**Y TARGET ****	**Z TARGET ****				
1.00260981D-02	1.88962361D-01	6.53798820D-02	-6.17526470D-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE INHIBITOR IS 7.81250000D-03 TRAJ. COUNTER IS 48

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.75868747D 02	4.50000000D 01	2.98029750D 01	8.94089250D-01	0.0	0.0	0.0	3.00171683D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.78237D 00	-4.84193D 00	-3.22500D 02	2.58048D-01	-1.79283D-03
1.40687D 00	1.48932D 00	-3.62143D 00	-2.35664D-01	3.54595D-06
-1.29985D 00	-1.92820D 00	4.95307D-01	-7.16009D-02	-1.96990D-05
4.05071D-01	-7.94874D-03	4.52295D-02	-2.97691D-01	5.81483D-06

TRAJECTORY SUMMARY ITERATION NO. 20

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS. USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.627047650-01	3.666813460-01	-6.906462000-01	-5.43532934D 00	5.20019450D 00	-2.45451918D 00	3.75860723D 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.40000000D 04	1.50000000D 03	0.0	0.0	2.70000000D 02	0.0	9.00000000D 01	1.20000000D 01
THETA **	**PSI**	**PHI**	**T2 ****	**THETA**	**PSI**	**PHI**	**T3**
-1.56478730D 01	0.0	0.0	5.90908718D 03	0.0	0.0	0.0	1.36500000D 04
DL OMS **	**DL OML ****	**DL INC**	**DL VPO ****	**VPOO**			
-2.44066025D-03	5.37790177D-03	0.0	2.51671623D-05	7.91261602D 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.46066058D 02	3.00172228D 02	7.09400505D-02	5.24101271D 00	3.02610414D-04	4.91747153D 00	5.18789204D 01	7.87755738D-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.79123794D 00	1.04370512D 00	2.00047747D-01	2.66354771D-04	-7.42707051D-03	-8.56319805D 01	2.34965123D 00	-2.48780959D-02
PCENT.TRGT	**X TARGET ****	**Y TARGET ****	**Z TARGET ****				
1.00239549D-02	1.88964208D-01	6.53711709D-02	-6.18377247D-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 7.81250000D-03   TRAJ. COUNTER IS 50

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.75860723D 02	4.50000000D 01	2.97946556D 01	8.93839667D-01	0.0	0.0	0.0	3.00172228D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.78168D 00	-4.84192D 00	-3.22498D 02	2.57879D-01	-1.79473D-03
1.40697D 00	1.48961D 00	-3.62150D 00	-2.35770D-01	3.53846D-06
-1.29985D 00	-1.92828D 00	4.95275D-01	-7.16060D-02	-1.97340D-05
4.05036D-01	-7.94074D-03	4.52311D-02	-2.97589D-01	5.82676D-06

TRAJECTORY SUMMARY ITERATION NO. 21

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.627014440-01	3.666860060-01	-6.906469120-01	-5.435366760 00	5.200163500 00	-2.454507800 00	3.758601510 02	
C	PO	ALPHA	TO	BETA	DELTA	EPSLON	T1
2.400000000 04	1.500000000 03	0.0	0.0	2.700000000 02	0.0	9.000000000 01	1.200000000 01
THETA **	PSI	PHI	T2 **	THETA	PSI	PHI	T3
-1.564778680 01	0.0	0.0	5.909112540 03	0.0	0.0	0.0	1.365000000 04
DL OMS **	DL OML **	DL INC	DL VPO **	VPCO			
1.282980950-04	2.724564190-04	0.0	1.794593460-06	7.912617820 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.460654310 02	3.001715900 02	7.094005050-02	5.241017640 00	3.026106250-04	4.917486230 00	5.187903890 01	7.877569380-01
APCEN. SUN	PRICEN.SUN	DST.FR.TRG	VL.WRT.TRG	SM.AX.TRGT	FL.P.A.TGT	ECCENT.TRG	APCNT.TRGT
8.791270130 00	1.043702330 00	2.000472000-01	2.663548400-04	-7.427067880-03	-8.563555670 01	2.348074690 00	-2.486637800-02
PCENT.TRGT	X TARGET **	Y TARGET **	Z TARGET **				
1.001224230-02	1.889677420-01	6.535934150-02	-6.?33124910-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 5.000000000-01   TRAJ. COUNTER IS 54

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.758601510 02	4.500000000 01	2.979471980 01	8.938415940-01	0.0	0.0	0.0	3.001715900 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.781750 00	-4.841990 00	-3.224970 02	2.578800-01	-1.794710-03
1.407020 00	1.489660 00	-3.621510 00	-2.357700-01	3.539580-06
-1.299860 00	-1.928280 00	4.952690-01	-7.160620-02	-1.973390-05
4.050340-01	-7.943020-03	4.523190-02	-2.975900-01	5.826690-06

```
                         TRAJECTORY SUMMARY ITERATICN NO. 22

                              INDEPENDENT PARAMETERS

      ** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION
X (TO)          Y (TO)          Z (TO)          XDOT            YDOT            ZDOT            M (TO)
6.637026340-01  3.666843530-01 -6.906466480-01 -5.435353830 C0  5.20C174700 00 -2.454512190 00  3.758602020 02

C               PO              ALPHA           TO              BETA            DELTA           EPSLON          T1
2.400000000 04  1.50CC00000 03  0.0             0.0             2.700000C0D 02  0.0             9.000000000 01  1.200000000 01

THETA     **    PSI             PHI             T2        **    THETA           PSI             PHI             T3
-1.564796920 C1 0.0             0.0             5.909026440 03  0.0             0.0             0.0             1.365000000 04

DL OMS    **    DL OML    **    DL INC          DL VPO    **    VPOC
-4.767946550-05 -9.437779460-CS 0.0             -1.598620530-07 7.912617660 00

                              DEPENDENT PARAMETERS

FINAL MASS      NET S/C   **    THRUST          DST.FR.SUN      VL.WRT.SUN      SM.AXIS SN      FL.P.A.SUN      ECCENT.SUN
3.460656420 02  3.0C1718050 02  7.094005050-02  5.241015710 0C  3.02610451D-04  4.917477860 00  5.187900150 01  7.877565820-01

APCEN. SUN      PRICEN.SUN      DST.FR.TRG      VL.WRT.TRG      SM.AX.TRGT      FL.P.A.TGT      ECCENT.TRG      APCNT.TRGT
8.791233410 00  1.043702310 00  2.000467190-01  2.663549140-04 -7.427064780-03 -8.563360710 01  2.348928040 00 -2.487270550-02

PCENT.TRGT      X TARGET  **    Y TARGET  **    Z TARGET  **
1.001857600-02  1.889651230-01  6.536545410-02 -6.183011140-03

   TRAJECTORY TERMINATED IN JUPITER REFERENCE      INHIBITCR IS  5.000000000-C1     TRAJ. COUNTER IS   56

                              SPACECRAFT MASS BREAKDOWN (KG)

 INITIAL        PROPULSION      PROPELLANT      TANKAGE         STRUCTURE       RETRO PROP      RETRO STR       NET S/C
3.758602020 02  4.50CC00000 01  2.979456060 01  8.938368190-01  0.0             0.0             0.0             3.001718050 02

                              PARTIAL DERIVATIVE MATRIX BY ROWS


-3.781710 00 -4.84196C 00 -3.224970 02  2.578770-01 -1.794750-03

 1.407000 00  1.489640 00 -3.621500 00 -2.357720-01  3.538410-06

-1.299860 00 -1.928280 C0  4.932720-01 -7.160640-02 -1.973450-05

 4.050350-01 -7.942200-03  4.523210-02 -2.975880-C1  5.826920-C6
```

TRAJECTORY SUMMARY ITERATION NO. 23

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS, USED IN THE ITERATION

X (TO)	Y (TO)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.627026340-01	3.666843530-01	-6.906466480-01	-5.435353830 00	5.200174760 00	-2.454512190 00	3.758602020 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.400000000D 04	1.500000000D 03	0.0	0.0	2.700000000D 02	0.0	9.000000000D 01	1.200000000D 01
**THETA ** **	**PSI**	**PHI**	**T2 ** **	**THETA**	**PSI**	**PHI**	**T3**
-1.564796920 01	0.0	0.0	5.909026440 03	0.0	0.0	0.0	1.365000000D 04
**DL OMS ** **	**DL OML ** **	**DL INC**	**DL VPO ** **	**VPOO**			
3.984677740-14	5.280507040-14	0.0	-1.238633730-17	7.912617660 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.460656420 02	3.001718050 02	7.094005050-02	5.241015710 00	3.026104510-04	4.917477060 00	5.187900150 01	7.877565820-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.791253410 00	1.043702310 00	2.000457190-01	2.663549140-04	-7.427064780-03	-8.553360710 01	2.348928040 00	-2.487270550-02
PCENT.TRGT	**X TARGET ** **	**Y TARGET ** **	**Z TARGET ** **				
1.001657600-02	1.089651230-01	6.536545410-02	-6.183011140-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 3.43597384D 10   TRAJ. COUNTER IS 70

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.758602020 02	4.500000000D 01	2.979456060 01	8.938368190-01	0.0	0.0	0.0	3.001718050 02

PARTIAL DERIVATIVE MATRIX BY ROWS

-3.781710 00	-4.841660 00	-3.224970 02	2.578770-01	-1.794750-03
1.407000 00	1.489640 00	-3.621500 00	-2.357720-01	3.538410-06
-1.299860 00	-1.928280 00	4.952720-01	-7.160640-02	-1.973450-05
4.050350-01	-7.942190-03	4.523210-02	-2.975880-01	5.826920-06

THIS CASE IS CONVERGED.

141 TRAJECTORIES WITHOUT PARTIAL DERIVATIVES AND 23 TRAJECTORIES WITH PARTIAL DERIVATIVES REQUIRED FOR THIS CASE.
CASE CONVERGED

FINAL TRAJECTORY

TRAJECTORY ARC DATA

```
TIME IN DAYS. HRS.. MINS.. SECS.        0        0        0       0.0
   XR=  4.2268267470F93265D 03    YR=  2.3387737095410430D 03    ZR=  -4.4050582756363720D 03    RR=  6.5376190891701130D 03
 XRDT=  -9.6298843162591140D 00  YRDT=  9.2132145184308160D 00  ZRDT=  -4.3486699215159560D 00  RRDT=  1.4018883613861250D 01
 XVJP=  8.7591022613494920D 08  YVJP=  8.0239625869498380D 07  ZVJP=  -1.7982533906611260D 07  RVJP=  8.7976152044454960D 08
   XI=  0.0                       ETA=  0.0                        ZETA=  0.0                         PERT=  0.0
 XIDT=  0.0                     ETADT=  0.0                      ZETADT=  0.0                        VPERT=  0.0

OSCULATING ELEMENTS AT TIME T=   0.0                 TRUE ANOM=   1.1774557921851830-07
SEM MAJ AXIS=  -9.3767224786388590-01    ECCENT.  =   2.2233268475869700 00   PERICENT.=   6.5376190891701120 03
  APOCENT.  =  -1.7225793980763630 04  INCLINATION=   4.7633064732246600 01 ARG PERIC. =  -1.1472022313715230 02
  PERIOD    =   1.0799833746079590 03  MEAN MOT.  =   5.8176505012362710 00 RA ASC NODE=   1.5342479751636030 02
 M ANOMALY  =   8.8738249645372470-09  E ANOMALY  =   7.2537867352285530-08  T PERIC.  =  -2.6621079572017990-10
UNIT PERICENTER POSITION VECTOR =   6.4553916030687620-01   3.5774083325292860-01  -6.7380161055355190-01
UNIT ANGULAR MOMENTUM VECTOR    =   3.3185959187107960-01   6.6740721184763030-01   6.7064590005579680-01

TIME IN DAYS. HRS.. MINS.. SECS.        0       11       59      60.000
   XR=  -3.5004952336362300 05    YR=  1.7274418740373270 05    ZR=  2.3336796615653690 03    RR=  3.9036002509147240 05
 XRDT=  -7.9154886477043260 00  YRDT=  3.7305040732606440 00  ZRDT=  2.2646352416173450-01  RRDT=  8.7534937602731360 00
 XVJP=  8.7441500811734230 08  YVJP=  8.1492756525559190 07  ZVJP=  -1.7976338620524460 07  RVJP=  8.7838925016111330 08
   XI=  3.8942411731122590 00   ETA=  -1.9327132036171930 00   ZETA=  4.7776927590190530-02   PERT=  4.1852944754492400 00
 XIDT=  4.1666532825663163D-03 ETADT=  -1.6791202433981352D-03ZETADT=  2.1108503247357430-05 VPERT=  4.4923150468936730-03

OSCULATING ELEMENTS AT TIME T=   1.1999999999999990 01 TRUE ANOM=   1.1518426576285150 02
SEM MAJ AXIS=  -8.3794282517019910-01    ECCENT.  =   2.2231510429044350 00   PERICENT.=   6.5372301840484340 03
  APOCENT.  =  -1.7226305333051780 04  INCLINATION=   4.7856679697931240 01 ARG PERIC. =  -1.1472249808072880 02
  PERIOD    =   1.0801204344890730 00  MEAN MOT.  =   5.8171154876559810 00 RA ASC NODE=   1.5342472688134650 02
 M ANOMALY  =   7.9996519215930030 03  E ANOMALY  =   2.4059117028636940 02  T PERIC.  =  -2.9380775559229020-04
UNIT PERICENTER POSITION VECTOR =   6.4654785620861940-01   3.5767592053982480-01  -6.7382772685901410-01
UNIT ANGULAR MOMENTUM VECTOR    =   3.3187033881552460-01   6.6344474402372920-01   6.7059509770425230-01

TIME IN DAYS. HRS.. MINS.. SECS.      246        5        1      35.195
   XR=  -4.3918793273770960 08    YR=  -1.0387283229575150 08    ZR=  7.7877391664304030 06    RR=  4.5137153975144710 08
 XRDT=  -1.1140182882766570 01  YRDT=  -1.6692139945245610 01  ZRDT=  2.7466165172105960-01  RRDT=  2.0071660337987460 01
 XVJP=  3.4916440964942000 08  YVJP=  1.0596292345988650 09  ZVJP=  -1.0772875021593620 07  RVJP=  3.6504647380694460 08
   XI=  2.1353337517596030 04   ETA=  -1.3157874705436360 05   ZETA=  2.9345279347263090 04   PERT=  1.3898554885981520 05
 XIDT=  6.9387108832601570 01 ETADT=  -3.9489967938761040 02ZETADT=  1.1635675223007250 02 VPERT=  4.1805340382663500 02

OSCULATING ELEMENTS AT TIME T=   5.9290254431896080 03 TRUE ANOM=   1.1800655000096270 02
SEM MAJ AXIS=   1.1236527085289290 05    ECCENT.  =   7.7402565535787050-01   PERICENT.=   1.6195787289995270 08
  APOCENT.  =   1.2714616280488030 09  INCLINATION=   1.1114342799804590 00 ARG PERIC. =  -5.5196742855312010 01
  PERIOD    =   9.1924540479131520 04  MEAN MOT.  =   6.3351559257376970-05 RA ASC NODE=   1.3050115572496380 02
 M ANOMALY  =   2.1479147545790570 01  E ANOMALY  =   5.1425989149066030 01  T PERIC.  =   1.6936725369710570 02
UNIT PERICENTER POSITION VECTOR =   2.5356597717904190-01   9.6716659673933139D-01  -1.5927177430117140-02
UNIT ANGULAR MOMENTUM VECTOR    =   1.4749318145835980-02   1.2597622293520070-02   9.9981186106526060-01

TIME IN DAYS. HRS.. MINS.. SECS.      569       17       59      60.000
   XR=  2.8268804439125580 07    YR=  9.7785412078940300 06    ZR=  -9.2496610009140620 05    RR=  2.9926589094952510 07
 XRDT=  -1.0700378922319700 01  YRDT=  -2.8097242747828550 00  ZRDT=  3.4108902925601620-01  RRDT=  1.1068378440738580 01
 XVJP=  2.8268804439125580 07  YVJP=  9.7785412078940300 06  ZVJP=  -9.2496610009140620 05  RVJP=  2.9926589094952510 07
   XI=  1.8666350946585120 04   ETA=  2.2060861517745580 04   ZETA=  -3.7159949070886930 02   PERT=  2.8900732705933020 04
 XIDT=  1.8904623299553530 02 ETADT=  2.2399594898319270 02ZETADT=  -3.7777347462992720 00 VPERT=  2.9313296403798990 02

OSCULATING ELEMENTS AT TIME T=   1.3650000000000000 04 TRUE ANOM=  -1.1075202347148930 02
SEM MAJ AXIS=  -1.7420083890452870 02    ECCENT.  =   2.3489280428903250 00   PERICENT.=   1.4987692545394570 06
  APOCENT.  =  -3.7209226418824530 06  INCLINATION=   1.7711781024571663D 00 ARG PERIC. =   2.0713137117921090 01
  PERIOD    =   1.8158907483697900 02  MEAN MOT.  =   3.4601119663397740-02 RA ASC NODE=   1.0912006272262570 02
 M ANOMALY  =  -1.4134071076177100 03  E ANOMALY  =  -1.8147235.2296089D 02  T PERIC.  =   1.4362942469463320 04
UNIT PERICENTER POSITION VECTOR =  -6.4039511393864370-01   7.6796783415937580-01   1.0931773883208910-02
UNIT ANGULAR MOMENTUM VECTOR    =   2.9202789314794390-02   1.0123825686876170-02   9.9955222384968016D-01
```

TRAJECTORY SUMMARY ITERATION NO. 24

INDEPENDENT PARAMETERS

** NAME APPLIES TO PARAMETERS. USED IN THE ITERATION

X (TO)	Y (TC)	Z (TO)	XDOT	YDOT	ZDOT	M (TO)	
6.62702634D-01	3.66684353D-01	-6.906466480D-01	-5.435353830 00	5.20017470D 00	-2.45451219D 00	3.75860202D 02	
C	**PO**	**ALPHA**	**TO**	**BETA**	**DELTA**	**EPSLON**	**T1**
2.40000000D 04	1.50000000D 03	0.0	0.0	2.70000000D 02	0.0	9.00000000D 01	1.20000000D 01
THETA **	**PSI**	**PHI**	**T2 ****	**THETA**	**PSI**	**PHI**	**T3**
-1.564796920 01	0.0	0.0	5.90902644D 03	0.0	0.0	0.0	1.36500000D 04
DL OMS **	**DL OML ****	**DL INC**	**DL VPO ****	**VPOO**			
3.984677740-14	5.28050704D-14	0.0	-1.238633730-17	7.91261766D 00			

DEPENDENT PARAMETERS

FINAL MASS	NET S/C **	THRUST	DST.FR.SUN	VL.WRT.SUN	SM.AXIS SN	FL.P.A.SUN	ECCENT.SUN
3.450636420 02	3.00171885D 02	7.094005050-02	5.24101571D 00	3.02610451D-04	4.91747786D 00	5.18790015D 01	7.87756582D-01
APCEN. SUN	**PRICEN.SUN**	**DST.FR.TRG**	**VL.WRT.TRG**	**SM.AX.TRGT**	**FL.P.A.TGT**	**ECCENT.TRG**	**APCNT.TRGT**
8.73125341D 00	1.04370231D 00	2.00046719D-01	2.66354914D-04	-7.42706478D-03	-8.56336071D 01	2.34892804D 00	-2.48727055D-02
PCENT.TRGT	**X TARGET ****	**Y TARGET ****	**Z TARGET ****				
1.00185760D-02	1.88965123D-01	6.53654541D-02	-6.18301114D-03				

TRAJECTORY TERMINATED IN JUPITER REFERENCE   INHIBITOR IS 5.65391061D 73   TRAJ. COUNTER IS 141

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO PROP	RETRO STR	NET S/C
3.75860202D 02	4.50000000D 01	2.97945606D 01	8.93836819D-01	0.0	0.0	0.0	3.00171805D 02

PARTIAL DERIVATIVE MATRIX BY ROWS

0.0	0.0	0.0	0.0	0.0
0.0	0.0	0.0	0.0	0.0
0.0	0.0	0.0	0.0	0.0
0.0	0.0	0.0	0.0	0.0

CASE SUMMARY

LAUNCH	TARGET
EARTH	JUPITER

CONSTRAINED MODE
S-VECTOR RELATIVE TO SUN

LAUNCH VEHICLE COEFFICIENTS			EFFICIENCY LAW COEFFICIENTS		PHOTON ABS. COEFF.	SPEC. ARRAY AREA
A1(KG)	A2(M/SEC)	A3(KG)	B	DSQ(M*M/SEC/SEC)	CA	XKP(M*M/W)
153208.05	2385.1258	53.34	0.7690	204490000.0	0.0	0.0

SPACECRAFT MASS BREAKDOWN (KG)

INITIAL	PROPULSION	PROPELLANT	TANKAGE	STRUCTURE	RETRO	NET S/C
3.758602020D 02	4.500000000D 01	2.979456050D 01	8.938368190D-01	0.0	0.0	3.001718050D 02

SPACECRAFT DESIGN FACTORS

N-VECTOR		S-VECTOR	
ALPHA(DEG)	BETA(DEG)	DELTA(DEG)	EPSILON(DEG)
0.0	270.000	0.0	90.000

PROPULSION SYSTEM PARAMETERS

REF POWER(W)	REF THRUST(N)	EXHAUST SPEED(M/SEC)	EFFICIENCY	UNIT THRUSTER POWER(W)	ARRAY AREA(M**2)
1500.00	0.070940	24000.00	0.56752	0.0	0.0

TRAJECTORY SCHEDULE

TIME (DAYS)	0.500	246.209	568.750
THETA OR XI (DEG)	-15.648	0.0	
PSI OR NU (DEG)	0.0	0.0	
PHI OR ZETA (DEG)	0.0	0.0	
POWER DIVERTED (W)	0.0	0.0	
PSI MAX (DEG)	0.0	0.0	
THRUST	ON	OFF	

DEPARTURE AND ARRIVAL CONDITIONS

DATE	EXCESS SPEED(M/SEC)	C3 (KM**2/SEC**2)	DATE	EXCESS SPEED(M/SEC)	C3 (KM**2/SEC**2)
4570.0000	8636.41717	74.58770	5138.7500	10679.00159	114.04107

REFERENCES

[1] Pines, S. and Cockrell, B. F.; "Partial Derivatives of Matrices Representing Rigid Body Rotations," MSC Internal Note No. 68-FM-231, September 20, 1968.

[2] Whitlock, F. H., Wolf, H., Lefton, L., and Levine, N.; "Interplanetary Trajectory Encke Method Fortran Program Manual for the IBM System/360," NASA/GSFC Report X-643-70-330, September 1970.

[3] Horsewood, Jerry L., "A Segmented Two-body Low Thrust Trajectory Optimization Program," Analytical Mechanics Associates, Inc. Report 71-10, March 1971.