MIDCOURSE NAVIGATION FOR THE EUROPEAN COMET HALLEY MISSION

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ABSTRACT

The European Comet Halley mission involves a sequence of mid-course orbit corrections to remove deviations from the nominal flight path, as indicated by orbit determination. As a function of the spacecraft design, several manoeuvre execution modes are studied, and the corresponding optimization problems are discussed. For on-board propellant estimation, a simple sequential algorithm is employed. The use of these optimization and propellant budgeting methods in operations is considered.

Keywords: Mid-course navigation, Fuel estimation, Manoeuvre optimization, GIOTTO mission.

1. INTRODUCTION

1.1 The GIOTTO mission

After the joint NASA/ESA cometary mission was abandoned for budgetary reasons in early 1980, ESA came up with a European project in order not to miss the unique opportunity of the appearance of Halley's comet in 1986.

Within the financial and scheduling envelope, the spacecraft design has mainly been based on existing technology. The spinning GIOTTO spacecraft will meet the scientific objective,

> to approach the cometary nucleus close enough to transmit high resolution images and to analyse gas and dust in situ.

In the first half of July, 1985, GIOTTO will be launched by ARIANE from Kourou into a transfer orbit, together with a geostationary satellite. After a few revolutions around the earth, a solid motor will provide a velocity increment of about 1400 m/s to inject the spacecraft into an excess hyperbola into its interplanetary trajectory (Fig. 1).

The encounter with the comet is scheduled after about 250 days, close to midnight March 13th/14th 1986. In the last 4 hours before encounter, nominally 40 Kbits/s of scientific data will be transmitted in X-band to the 64m radio-astronomy antenna in Parkes, Australia. Survival of the probe after fly-by is not envisaged.

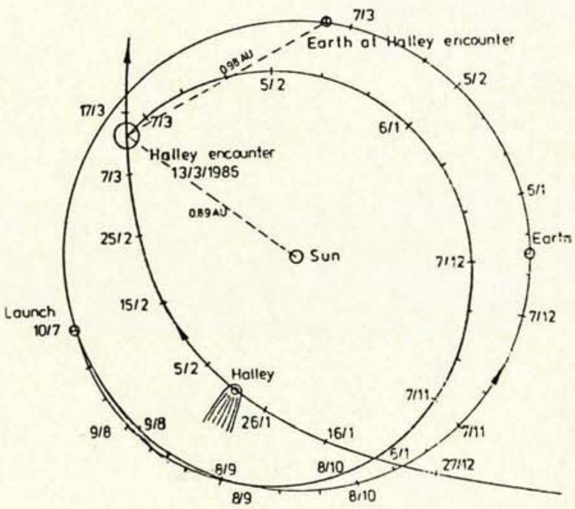

Figure 1

1.2 Mid-course navigation

The unavoidable injection error, detected from orbit determination, necessitates correction manoeuvres in order to compensate for deviations from the target at the fixed time of arrival. Unfortunately, the motion of this target can only be uncertainly predicted because of non-gravitational accelerations acting on the comet during its perihelion passage which cannot be modelled precisely. Observation data from the scientific camera on the dual spin spacecraft are unlikely to provide useful information for terminal navigation. New information on the target position will be derived from earth-based astronometric observations in November/December, 1985. Therefore, at this time during the cruise, re-targeting has to be planned. Eventually, information from other cometary missions can be utilized a few days before encounter for re-targeting. Apparently, the navigation analysis has to consider a variety of mission-specific features, however the dominating portion of propellant will be consumed correcting for the execution error in the perigee kick. This first orbit correction manoeuvre will take place as soon as sufficient ranging data are

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available, preferably within the first two days after injection, when the spacecraft is less than 600,000km from the earth. Within this distance communication can be maintained by means of the low gain cardoid antenna in S-band which offers the opportunity of turning an axial thruster, i.e. the spacecraft spin axis, into almost any direction required for the manoeuvre. We call this manner of manoeuvre execution an omni-directional mode (OM).

Further away, the high-gain despun paraboloid antenna has to be kept pointing towards the earth. The angle of the antenna beam with respect to the spacecraft spin axis is determined by the encounter geometry. The side of the spacecraft opposite to the antenna is covered by a dust protection shield which is turned into the approach direction towards the comet during the terminal mission phase. This mounting of the despun high gain antenna, together with thermal and power conditions, constrains the spin axis directions at any time during the cruise to a small arc on about a 45° cone around the earth's direction. Orbit manoeuvres in the cruise phase therefore have to consider the prescribed thruster configuration. We call this type of manoeuvre execution a combined mode (CM). More details on these mission specific correction modes and combinations will be given in section 2.1.

The fuel optimum correction sequences in any combination of manoeuvre execution modes involve, at most, three impulses. The optimization problem and solution methods are treated in 2.2 and 2.3. The fast numerical methods have been applied in Monte Carlo calculations of the statistical properties of the injection error correction (3.3). However, the statistical results based on very simple one-impulse strategies compared, in most cases, quite well with the optimum three-impulse solutions (3.4).

In the overall propellant estimation, therefore, only one-impulse corrections were considered; this means at any time during the cruise, a manoeuvre is calculated and decomposed such that the error, as far as it is known at that time, is immediately removed. With this approach, manoeuvre execution errors, dynamical system noise, the orbit determination process and the target observations process and the sequence of fine corrections later in the cruise can easily be modelled and included (3.1, 3.2).

From the experience and the results of these studies, some consequences on the operational flight dynamics support can be drawn (4). They can be used in the context of:

- stochastic optimization of orbit injection and mid-course corrections considering orbit determination accuracies of probe and comet;
- fuel budgeting after the perigee kick;
- calculations of mission success probabilities in trade-offs during operations.

In particular, the results show the importance for mission success of manoeuvre optimization, orbit determination and cometary ephemeris estimation relative to each other.

2. COMPUTATION OF MID-COURSE CORRECTIONS

2.1 The Correction Modes

Each of the two manoeuvre modes OM (= omni-directional mode) and CM (= combined mode) provides a single velocity increment. Table 1* shows the five different possibilities to correct a target-miss-vector of GIOTTO by up to three maneouvres in either of these modes. More than three manoeuvres need not appear in an absolute fuel minimum solution (Ref. 3). In the next paragraphs we will deal with the problem of computing optimum solutions for these five mid-course correction modes.

* for Table 1 please see following page.

2.2 The Propellant Minimisation Problem

Mid-course correction is a typical example of trajectory controls, which only comprise manoeuvres $(t_i, \underline{v}_i)$ with small, almost impulsive velocity increments $\underline{v}_i = (v_i^{(1)}, v_i^{(2)}, v_i^{(3)})$ at the manoeuvre times t_i, i = 1,....N.

The set $\{t_i\}$ of switch-times emerges from a proper discretisation of the time interval $[t_o, t_N]$ between the departure time t_o and the arrival time t_N. In interplanetary flights, the t_i may typically be representative of daily time slots with ground station contact.

If we assume linear equations of motion or equations of motion linearisable around a reference trajectory, the total effect $\Delta\underline{x}_N$ of the N manoeuvres $(t_i, \underline{v}_i)$ on the n-dimensional ($n \leq 6$) target state $\underline{x}_N$ in the target space $R^n(t_N)$ at t_N can be written as

$$\Delta\underline{x}_N = \sum_{i=1}^{N} \sum_{\rho=1}^{3} \underline{b}^{(\rho)}(t_i)\, v_i^{(\rho)} \quad (1)$$

The time dependent n-vectors $\underline{b}^{(\rho)}(t_i)$ are well defined along the reference trajectory. They are the partial derivatives of the state $\underline{x}_N$ with respect to $v_i^{(\rho)}$ at t_i. The column vectors $\underline{b}^{(\rho)}(t_i)$ form the n x 3 transition matrix $B(t_N, t_i)$ from $\underline{v}_i$ to $\underline{x}_N$. In the case of interplanetary flights $B(t_N, t_i)$ may be computed by techniques which take account of perturbations during the departure from the earth-moon system or during the approach to the target planet (Ref. 4).

We introduce at each t_i a suitable reference frame for the velocity increments $\underline{v}_i$, for instance a frame with one of its axes pointing in the direction of the sun or in the direction of the actual spin-axis. This procedure enables us to incorporate some satellite specific manoeuvre modes into the optimisation process by pure rotations of the transition vectors $\underline{b}^{(\rho)}(t_i)$.

For the above linear and impulsive controls the following rendezvous problem is to be solved:

Determine a permissible sequence of manoeuvres $\{t_i^*, \underline{v}_i\}$ with minimum fuel consumption $t_i^* \in \{t_i\}$, $t_i \in [t_o, t_N]$ which provides a prescribed target increment of $\Delta\underline{x}_N$ at the fixed arrival time t_N according to the rendezvous conditions (1).

Correction Mode	Manoeuvres No.	type	mode	Spin-axis	Purpose
1	3	1AC +2AC or +2RP or +1AC,1RP	OM CM CM CM	arbitrary, permitted fixed fixed fixed	Nominal, fuel optimum 3-impulse solution
2	1	AC	OM	arbitrary, permitted	one-impulse solution in early orbit phase; not necessarily fuel optimum but simple
3	3	3AC (*1) or 2AC,1RP or 1AC,2RP or 3RP (*2)	CM CM CM	fixed " " "	fuel optimum solution for fixed spin axis *1) if radial thruster failed *2) if axial thruster failed
4	1	1AC,1RP	CM	fixed	one-manoeuvre solution for fixed spin axis
5	2	2AC(*) or 1AC,1RP or 2RP(*)	CM CM CM	fixed " "	two-manoeuvre solution for continuation of 3-manoeuvre solution

TABLE 1: Mid-course correction modes

AC = axial continuous burn
RP = radial pulsed burn
OM = omni-directional mode
CM = combined mode
* = possible contingency modes in case of thruster failure

The fuel used by a single manoeuvre depends on the thruster configuration and on the thrust mode. We only consider thruster systems with constant exhaust velocities. The fuel consumption formula that are specific for GIOTTO will be given in the next paragraph.

In operations, constraints on the manoeuvre times and the spacecraft attitude must be taken into account. Fortunately, any constraints on the t_i can easily be incorporated by removing forbidden times from the set $\{t_i\}$.

Constraints on the attitude and, hence on the v_i-directions are assumed to have the following form:

$$\frac{\underline{v}_i}{|\underline{v}_i|}\cdot\hat{\underline{s}}_i^{(j)} \leqslant \cos\phi_i^{(j)}, j=1,..,J, i=1,..,N \qquad (2)$$

The $\hat{\underline{s}}_i^{(j)}$ are unit vectors. The $\phi_i^{(j)}$ are the half-cone angles of forbidden cones about the $\hat{\underline{s}}_i^{(j)}$.

For GIOTTO we have to investigate a "fixed time of arrival" problem. The dimension n of $\Delta\underline{x}_N$ is equal to 3 as the arrival velocity is not controlled.

In the omni-directional mode for the GIOTTO mission, the $\underline{v}_i$ may be subjected to sun-aspect angle constraints. The $\underline{v}_i$ may also be constrained to small cones around the satellite-earth-line. The corresponding manoeuvres could then be well calibrated from range-rate information.

2.3 The Propellant Consumption

In the case of GIOTTO we have:

a) the omni-directional mode: $\underline{v}_i$ is generated by a continuous burn after an axial thruster has been rotated into the (permissible) direction of $-\underline{v}_i$. The fuel consumption is proportional to

$$||\underline{v}_i||_3 = +\sqrt{v_i^{(1)2} + v_i^{(2)2} + v_i^{(3)2}} \qquad (3)$$

b) the combined mode: v_i can be generated by three thrusters, two axial thrusters pointing in the direction of $+v_i^{(3)}$ and $-v_i^{(3)}$ respectively, and a radial thruster perpendicular to the axial ones.

With $\| \underline{v}_i \|_2 = + \sqrt{v_i^{(1)^2} + v_i^{(2)^2}}$ the fuel consumption is proportional to

$$| \underline{v}_i^{(3)} | + g_i \| \underline{v}_i \|_2 \qquad (4)$$

The different efficiency of the axial continuous burn and the radial burns (pulsed) is represented by a positive weight g_i.

The fuel consumption for other thruster combinations and spacecraft stabilisations can often be modelled in a similar way, e.g. for 3-axes stabilised satellites we either have a purely omni-directional mode or fixed thrust directions.

If there are k=1,...., K manoeuvres in the omni-directional mode and ℓ=K+1,...., N manoeuvres in the combined mode, the total fuel consumption is proportional to

$$Z(\underline{v}_1,\ldots,\underline{v}_N) = \sum_{k=1}^{K} \| \underline{v}_k \|_3 + \sum_{\ell=K+1}^{N} (| \underline{v}_\ell^{(3)} | + g_{\ell-K} \| \underline{v}_\ell \|_2) \qquad (5)$$

The cost function (5) is a convex function in the 3xN unknowns $\underline{v}_i^{(\rho)}$, k=1,...,K; ρ =1,2,3. Our rendez-vous problem therefore becomes a <u>convex optimisation problem</u> for the unknown impulses. The cost function (5) has to be minimised under the linear rendez-vous conditions (1) and - in the omni-directional mode - under the convex constraints (2). In general, this can only be accomplished numerically.

2.4 <u>The Discretisation Method</u>

If we can prescribe at each switch point t_i, a sufficiently dense set of <u>permissible</u> directions $\underline{p}_{ij}$, j=1,...,M of velocity increments then the convex problem degenerates into a <u>linear optimisation problem</u> in which the unknowns are NxM absolute values V_{ij} of velocity increments (see also Ref. 3).

When solving a problem with constrained velocity directions one has to bear in mind that at some t_i the region with permissible thrust directions, i.e.the permissible domain $\Omega(t_i)$ for the control function is not convex. Hence the images $D_i = B(t_N,t_i)\,\Omega(t_i)$ of $\Omega\,(t_i)$ in the target space are not convex either. Their union $D = \bigcup_{i=1}^{N} D_i$ is the reachable domain introduced by Contensou and Edelbaum (Ref. 1, page 39). Solutions of the rendez-vous problem comprising <u>any permissible combination</u> of vectors of D may <u>not</u> be <u>feasible.</u> They may contain more than one manoeuvre at a single switch time t_i.

Therefore we have to seek for solutions of the linear problem, which do not contain more than one non-vanishing V_{ij} at each t_i. This yields a <u>special linear optimisation problem</u>.

It shows one more peculiarity; all coefficients of its cost function (5) are positive. This indicates the application of special numerical techniques. As has been shown in Ref. 3, replacing the classical simplex algorithm by a gradient projection method may reduce the computation time by a factor two.

We finally recall that most of the classical solution methods for the above problems, as for instance the analysis of the primer introduced by Lawden(Ref. 2) either tacitly assume convexity of the permissible region or become very complex.

2.5 <u>One and Two Manoeuvre Solutions</u>

In case of our special 'fixed time of arrival' problem (dimension of $\Delta \underline{x}_N$=3) the fuel optimum solution normally contains 3 non-vanishing impulses and hence possibly 3 manoeuvres. During operations not necessarily globally optimum correction modes with only 1 or 2 manoeuvres have to be considered.

As is well known the optimum one manoeuvre solution can immediately be derived from the following algorithm:

a) invert for i=1,...,N the rendez-vous condition

$$\Delta \underline{x}_N = B(t_N\,,\,t_i)\underline{v}_i \qquad (6)$$

This yields a velocity increment $\underline{v}_i$, with $|\underline{v}_i| \leqslant \infty$.

b) If i $\leqslant$ K check whether $\underline{v}_i$ points into a permissible direction.

c) Compute for permissible $\underline{v}_i$ and for i $\leqslant$K the fuel consumption $z(\underline{v}_i)$ from formula (3),
however for
i $>$K the fuel consumption from formula (4).

d) Select the optimum time t_i from the condition that $z(\underline{v}_i)$ should assume its minimum value.

If one of the thrusters failed, the one impulse solution might not be feasible.

The computation of the 2-manoeuvre solutions can be based on the method presented in 2.4. Computing the fuel optimum solution for all $\binom{N}{2}$ combinations of 2 permissible times t_ℓ, at least one optimum 2-manoeuvre solution is to be found among these $\binom{N}{2}$ cases. This appears to be a rather tedious procedure, however, it will only be applied a few times during the mid-course navigation.

3. FUEL STATISTICS

3.1 <u>The Kalman Type Algorithm</u>

As indicated in 2.2, any deviation from a reference trajectory will be sufficiently small to allow linearization. Restraining to one impulse correction and assuming white Gaussian zero mean noise processes and initial errors, this leads to a simple sequential algorithmic description of the navigation process by means of well known concepts of linear guidance and filtering. The probability densities of the moduli of the velocity increments ($\approx$ fuel statistics) can be derived analytically (ref. 4).

3.1.1 State Propagation

The random variable state of the system is represented by the n - vector $\underline{x}_i$ at times $\{t_i\}$ of events, namely measurement or correction times. The stochastic processes $\{\underline{x}_i\}$ consists of the state deviations of the probe, the comet and eventually some parameters (n = 6, 12 or more) from nominal. The propagation between events is described by the linear equation

$$x_i = \Phi_{i,i-1}\underline{x}_{i-1} + \underline{w}_{i-1} \quad (7)$$

where $\Phi_{i,i-1}$ denotes the state transition matrix from event i-1 to event i and $\{\underline{w}_i\}$ denotes a white Gaussian noise process of unmodelled dynamical disturbances with $E(\underline{w}_i\underline{w}_i^T) = Q_i$;the Q_i can be approximated from the original continuous process.

3.1.2 Observation Equation

At times t_i measurements

$$\underline{z}_i = H_i\underline{x}_i + \underline{v}_i \quad (8)$$

are taken. The H_i are the n x m matrices of partial measurement/state.$\{\underline{v}_i\}$ is a white Gaussian noise process with $E(\underline{v}_i\underline{v}_i T) = R_i$.

The measurement vector can take the form of arbitrary combinations of ranging of the probe from some ground station, optical observations of the comet's right ascension and declination as seen from the earth or the on-board camera.

3.1.3 Correction Manoeuvres

At correction times t_i no observations are assumed to be taken. Naturally, the portion in the state vector referring to the comet will not be affected. The state after correction $\underline{x}_i^+$ is obtained by incrementing the velocity part of the probe in the state vector before correction $\underline{x}_i^-$ according to

$$\underline{x}_i^+ = \underline{x}_i^- + J(\underline{v}_i + \underline{n}_i) \quad (9)$$

where $\underline{v}_i$ is the computed velocity correction and $\underline{n}_i$ is the zero mean Gaussian noise, modelling the execution error of the manoeuvre with Covariance M_i. J is a 3 x n matrix introduced for the sake of homogeneous notations.

3.1.4 Stochastic evolutions

In the linearized theory all random variables undergo only linear transformations, so they remain Gaussian and are fully described by the evolution of the mean and covariance matrix. In the present application we have to distinguish between two state random variables

- the actual state deviation $\underline{x}$
- the estimated state deviation $\hat{\underline{x}}$

The actual state deviation cannot be known on ground, therefore all manoeuvre calculations will be based on $\hat{\underline{x}}$. The covariance matrices of $\underline{x}$ and $\hat{\underline{x}} - \underline{x}$ will be denoted by

$C = E[\underline{x}\ \underline{x}^T]$ called dispersion
$P = E[(\hat{\underline{x}} - \underline{x})\ (\hat{\underline{x}} - \underline{x}^T)]$ called covariance

The random variables $\underline{x}$ and $\underline{x} - \hat{\underline{x}}$ have zero mean in case of an unbiased estimator by definition of the reference orbit.

The initial values P_o and C_o of covariance and dispersion at orbit injection are obtained from the terminal values of an orbit determination process over the parking orbit starting with the known launcher dispersion. Immediately before the execution of the injection manoeuvre we will usually have $P_o = C_o$.

Measurements will only affect $\hat{\underline{x}}$, whereas manoeuvres influence both velocity parts of the state vectors of the probe due to the mechanization error.

3.1.5 Covariance Propagation and Measurement Update

Between events the dispersion and the covariance matrix naturally propagate according to

$$P_i^- = \Phi_{i,i-1}P_{i-1}\Phi^T_{i,i-1}+Q_{i-1} \quad (10)$$

$$C_i^- = \Phi_{i,i-1}P_{i-1}\Phi^T_{i,i-1}+Q_{i-1} \quad (11)$$

At measurements the dispersion is not influenced, so

$$C_i^+ = C_i^- \quad (12)$$

and by the well known equations

$$P_i^+ = P_i^- + K_iH_iP_i^- \quad (13)$$

with the Kalman gain

$$K_i = P_i^-H^T_i(H_iP_i^-H^T_i+R_i)^{-1}. \quad (14)$$

3.1.6 Correction manoeuvres

For fixed time of arrival guidance, the distance from the target at final time (delivery error) has to be compensated by an impulsive manoeuvre $\underline{v}_i$ according to (6).

Introducing the decomposition

$$\Phi_{N,i} = \begin{bmatrix}\Phi_1 & \Phi_2\\ \Phi_3 & \Phi_4\end{bmatrix} \quad (15)$$

of the transition matrix we obtain

$$\underline{v}_i = \Lambda_i\hat{\underline{x}}_i \quad (16)$$

with the 3 x n Guidance matrix

$$\begin{aligned}\Lambda_i &= [\Lambda_i^c,\ \Lambda_i^p]\\ \Lambda_i &= [\Phi_2{}^{p-1}\ \Phi_2{}^c, \Phi_2{}^{p-1}\Phi_2{}^c]\\ \Lambda_i{}^p &= [-\Phi_2{}^{p-1}\Phi_1{}^p,\ -I_3\ ]\end{aligned} \quad (17)$$

where the indices p and c denote the transition matrices for the motion of the probe and the comet respectively.

Assuming the executive error of the manoeuvre n_i not to be correlated with the manoeuvre $\underline{v}_i$

we obtain for the covariance matrix of the manoeuvre

$$\delta \underline{v}_i = \underline{v}_i + \underline{n}_i \quad (18)$$

as it really is executed

$$S_i = E(\underline{v}_i \underline{v}_i^T) + M_i \quad (19)$$

As for the minimum variance estimator, the estimate $\hat{\underline{x}}$ is perpendicular to the estimation error $\underline{x} = \underline{x} - \hat{\underline{x}}$, we obtain after some calculations

$$S_i = \Lambda_i (C_i^- - P_i^-) \Lambda_i^T + M_i \quad (20)$$

Here, C_i^- and P_i^- are dispersion and covariance before manoeuvre execution. From this

$$P_i^+ = P_i^- + JM_i J^T \quad (21)$$

and using (15) after more algebra

$$C_i^+ = (I + J \Lambda_i)(C_i^- - P_i^-)(I + J \Lambda_i)^J + P_i^- + JM_i J^T \quad (22)$$

I is a n x n unit matrix. This completes the sequential algorithm.

The algorithm clearly does not consider the orbit determination process in sufficient complexity, so orbit determination results should not be drawn from it. It turned out to be sufficient for propellant estimation.

3.2 Probability densities for fuel consumption

The propellant consumption for a manoeuvre is approximately proportional to the modules $v_i = |\underline{v}_i|$ of the velocity increment.

In the omni-directional mode the density function of v_i can be derived from the diagonalized

$$\underline{S}_i = \begin{bmatrix} \lambda_1 & 0 & 0 \\ 0 & \lambda_2 & 0 \\ 0 & 0 & \lambda_3 \end{bmatrix} \quad (23)$$

by some typical transformations (ref. 4). It finally contains a hypergeometric function, evaluation of which has been done by numerical inversion of Laplace-transforms.

In the constrained mode the corresponding statistical transformations starting with a 2 x 2 matrix or a scalar, imply the evaluation of a Bessel function of the first kind and an error function.

For the Giotto mission the propellant consumption for orbit corrections was estimated at about 180 m/s 99-percentile under preliminary assumptions on the errors.

3.3 Monte Carlo statistics for injection error removal

To investigate how the simplified one-impulse correction propellant estimates compare with the optimum three impulse solution, the injection error was simulated for many random cases and the optimum correction sequence was calculated (in each case according to 2.4). It turned out that for the removal of the injection error, the optimization only leads to a considerable propellant saving if the execution mode of the first correction is combined (ref. 6).

3.4 Planned arrival time shifts

As only one ground station will be available to receive the high bit rate data stream during the 4 hour encounter period, the time of arrival has to be restricted such that this encounter period lies within the daily time interval in which the probe is visible from the ground station in Parkes (Australia). On March 13th/14th 1986, this period extends from 18.30 to 3.30 GMT, assuming a minimum elevation of 30°. Therefore, the probe has to arrive at the comet some time between 22.30 and 3.30. If the on-board hydrazine system could provide a ±9.5 hours shift of the arrival time, this constraint could be removed in the launch window calculations, which could then be extended to the end of July, assuming any solar aspect angle at injection is acceptable.

Inspection of the partial derivatives along the orbit shows that most efficiently, a 9.5 hour arrival time shift is accomplished by a manoeuvre of 127 m/s 77 days after injection. If the manoeuvre has to be decomposed along the prescribed attitude, 105 m/s radial and 85 m/s axial will be required. This decomposition at day 77, of course, is not optimal.

Propellant may be saved by combining part of this planned mid-course correction with the statistical correction manoeuvres which have to be executed anyway. In particular, the combination with the dominating first orbit correction after two days which removes the injection error has been studied. This combination will save only about 15 m/s, independent of the percentile which is looked at (ref. 5).

4. OPERATIONAL IMPLICATIONS

When preparing the operations it must be kept in mind that the success criterion of the Giotto mission is:

> *delivery of the spacecraft as close as possible to a properly chosen target point.*

This is different from the standard geostationary missions, where any propellant saving during the orbit insertion may extend the operational lifetime and, therefore, is the driving optimization criterion.

If the propellant tanks of Giotto are well sized, only in very rare cases or in emergency situations, will propellant optimization be necessary. Comparisons showed that optimum manoeuvre sequences do not lead to a propellant saving of more than 5 m/s below the primitive one-impluse solution for 99% of the injection errors.

This means the nominal operations can adopt a very simple scheme of orbit corrections, which fixed the manoeuvre times in advance without sophisticated optimisation, following criteria of operational convenience.

Nevertheless, optimization software must be available for any non-nominal case, e.g. malfunctioning of thrusters or operational problems in the initial phase, such that the first correction cannot be executed in omni-directional mode. Then only a well prepared complete package of optimization and mission planning software can guarantee mission success. Part of this package has to be a program for fuel budgeting. In particular, in optimizing the orbit injection, the propellant estimation during the cruise phase must be included. The mission analysis programs do not cope with the variety of operational constraints and possible cases, nevertheless the development of operational software may start with the experiences of mission analysis. In particular, it has been shown that the areas which are critical for mission success are the areas which directly influence the delivery error; these are the orbit determination and the cometary ephemeris estimation from earth-based astrometric observations.

Special attention must be paid to these latter problems within the next four years.

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