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OPTIMAL OPTIONS FOR RENDEZVOUS AND IMPACT MISSIONS TO NEOS

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In this paper some potentially interesting transfer options for missions to Near Earth Objects have been studied. Due to the high number of potential targets and to the large variety of possible missions that can be considered, especially if resorting to low-thrust propulsion, an extensive analysis of transfer options requires a preliminary approach oriented toward an effective global search, and an appropriately simplified trajectory transcription. Low-thrust options have been modeled through a novel shape-based approach and a global optimization method has been used to look for globally optimal transfers. Different targets have been identified and various mission scenarios have been considered: rendezvous, sample return missions both with and without Earth gravity assist and impact missions.

Keywords: NEO, impact missions, sample and return.

1. INTRODUCTION

The exploration of the Solar System is currently widening its scope to new targets; beside the continuous interest in planetary missions, new targets are being considered in the current and future plans of the space agencies [1, 2].

Near Earth Objects (NEO), which include both asteroids and comets, are in fact a very high priority objective for science, but at the same time, and this mainly is the case of asteroids, they draw the attention of scientists for the potential hazard they can represent for our planet [3, 4]. As an example it has been recognized that there are at least 160 impact craters on Earth and more than 100 tons of interplanetary material fall down to Earth daily. Luckily most of the objects are too small to cause damages, but objects in the range of 1-2 km, although statistically quite rare, could produce damages on a regional or global scale, while objects in the range of 100 m could cause enormous local damage.

For certain types of investigation, related to the assessment of the threat, space mission offer unique opportunities with respect to ground-based observation. Due to the difficulties related to the observation of these objects and to the high uncertainty in the knowledge of their orbits, space missions oriented toward in-situ measurements or transponder release turn out to be of high importance in the definition of the risk and of a mitigation strategy. It is therefore likely that future missions will be oriented either toward science objectives or hazard prevention or mitigation, or in some cases the two rationales will be present at the same time. Various missions to NEO have been investigated and implemented so far [5, 6]; however, differently from the case of the transfers to the planets, a wide range of possible alternative missions and transfer options still remain to be further analyzed. Additionally the recent availability of low-thrust propulsion systems, turns out to be the natural propulsive candidate for those types of mission, since they offer a flexible, cheaper and more effective way to fill the energy gap between the earth orbit and the asteroid orbit in case of rendezvous missions [7].

However due to the large number of possible targets and the wide variety of mission options, it seems clear that traditional trajectory design approaches may turn out to be unsuitable for an extensive analysis of mission opportunities maximizing both scientific and prevention objectives. The design problem can in fact be properly stated as a global optimization, which requires a proper preliminary search.

This paper intends to present a novel methodology for the preliminary investigation of interplanetary transfers, and to provide an analysis and characterization of different transfer options for possible missions to Near Earth Objects. Starting from a classification of potential targets, different kinds of missions are considered: single-target missions, which can have science or hazard mitigation purpose (which can be used for tracking of dangerous objects or for deflection missions), multiple-target missions (tour of a high number of NEOs) and sample and return options, both applied to single and multiple targets. Furthermore, an analysis of the best swing-by sequences is performed in the case of impact trajectories. Those missions in fact can typically require several gravity assists from the planets. Particular attention is also given to multiple-target missions with the inclusion of sample and return options, since those missions would represent an interesting avenue for new scientific investigations. Interplanetary flights targeting a certain number of asteroids, useful to release transponders to track the motion of hazardous asteroids could be achieved by a combination of multiple swing-by maneuvers and low-thrust propulsion. In addition, if a recurrent swing-by of the Earth is included among the gravity assist maneuvers, then sample and return missions could be considered. In the paper an analysis of both multiple target and sample return opportunities is presented.

The search for different transfers and best sequence options, with respect to the different types of mission, is performed by resorting to a preliminary design tool, based on [8] a global search by an incremental branch and prune method combined with an agent-based technique [9]. This methodology has proven to be an effective way of looking for optimal transfer options in the case of complex multiple swing-by problems and particularly when the definition of the optimal swing-by sequence is required. Low-thrust transfers are instead modeled by resorting to a methodology called the inverse method, which has already proven to be effective in the preliminary assessment of solutions for low-thrust trajectories.

2. TARGET SELECTION AND MISSION DESIGN

The high number of NEO already catalogued and the high rate of discovery of new objects every year (more than one hundred) would make a comprehensive investigation of all possible targets impossible. For this reason in this paper an extract from the JPL catalogue has been considered. Table 1 reports the asteroids considered in the following and the relative reference number assigned, where the semi-major axis a is in astronomical units and the inclination *I*, argument of the pericenter ω, longitude of the ascending node Ω and mean anomaly *M* are in degrees, while the codes *P*, *T* or *M* respectively stand for the Apollo, Athen and Amor classes.

The list considered in this work contains some bodies that recently have become the object of interest for the scientific community. 2004MN4, 2004VD17 and 1997XR2 are reported in the JPL catalogue of dangerous objects, having a Palermo scale ranging between -1.13 and -2.71 and value equal to one in the Torino scale, thus being worth of consideration. These targets are therefore of high interest since in-situ measurements and transponder release could be beneficial for an improvement of the available knowledge of potentially risky objects. Other asteroids like 1989ML, 2002AT4, 2003WP5, 1999AO10, instead have been recently considered as possible targets for both ESA and JAXA missions [6, 7]. More generally the list presents a variety of objects belonging to the Apollo, Athen and Amor classes.

The asteroids in the list above have been investigated as potential targets for different classes of missions: sample and return, multiple rendezvous with sample and return and impact missions. The feasibility of sample and return trajectories and multiple rendezvous transfers is here investigated for small mission classes. In the following it is assumed that the total mass of the spacecraft is not exceeding 600 kg, and that a light power and propulsion system is implemented. Power available at 1AU is assumed to be 3.88 kW, with a maximum thrust available of 70-80 mN and an I_{sp} = 3000 s. Solar array size is set to 11 m^2. Table 2 reports the spacecraft configuration.

The purpose of this work is to show that even for complex missions, resulting in the rendezvous with more than one NEO, sample and return options are achievable from a trajectory design point of view. No further analysis is however performed in terms of subsystem and operational issues.

3. TRAJECTORY MODEL

Low-thrust arcs are modeled using a shape-based method: the Cartesian coordinates of the spacecraft are described by means of a set of pseudo-equinoctial elements α. The set of elements used to parameterize the Cartesian coordinates are here called pseudo-equinoctial because they do not satisfy exactly the

TABLE 1: *Orbital Parameters for Considered NEOs.*

#	name	*A*	*e*	*I*	ω	Ω	*M*
1	VF32-T	0.8	0.44	24.0	320.8	236.8	47.4
2	WR12-T	0.75	0.40	7.1	205.8	63	259.6
3	SG344-T	0.97	0.06	0.10	274.5	192.5	144.4
4	UG-P	1.22	0.24	4.5	225.9	12.4	131.8
5	GK-P	1.92	0.59	5.60	111.6	15.22	315.6
6	SB45-P	1.56	0.40	3.67	216.4	195.5	175.3
7	Florence-M	1.77	0.42	22.2	27.6	336.2	164.9
8	Castalia-P	1.06	0.48	8.9	121.3	325.7	152.1
9	Pan-P	1.44	0.59	5.5	291.5	312.1	205.7
10	1989UQ-T	0.91	0.26	1.3	14.9	178.4	321.6
10	2001CC21-T	1.03	0.22	4.80	179.2	75.8	340.2
11	1996FG3- P	1.05	0.35	2.0	23.9	300	125.6
12	1999YB-M	1.32	0.07	6.80	192.8	31.1	78.3
13	1994CN2-P	1.57	0.49	1.40	248.1	99.4	204.2
14	2003GG21-P	2.14	0.70	10.1	94.9	13.2	161.2
15	1989ML-M	1.27	0.14	4.37	183.3	104.4	302.6
16	1999JU3-P	1.18	0.19	5.88	211.3	251.7	259.1
17	1999AO10-T	0.91	0.11	2.63	7.4	313.5	124.0
18	2000LG6-T	0.91	0.11	2.82	7.7	72.8	283.0
19	2003WP25-T	0.99	0.12	2.52	225.5	42.4	217.0
20	2002AT4-M	1.86	0.45	1,51	202.7	323.8	51.0
21	2004MN4-T	0.92	0.18	3.33	126.4	204.5	359.7
22	2004VD17-P	1.51	0.59	4.22	90.7	224.2	74.1
23	1997XR2-P	1.07	0.20	7.17	84.6	250.9	218.9

TABLE 2: *Spacecraft Configuration*.

m_0	I_{sp}	Power@1AU
595 kg	3000s	3.88kW

Gauss planetary equations unless the thrust is zero. Each element is then developed as a parameterized function of the longitude L. This function is the shape of the pseudo-element.

If the position vector is parameterized by means of the nonsingular equinoctial elements as follows:

$$\mathbf{r} = \begin{bmatrix} \frac{r}{1+h^2+k^2}(\cos L + (h^2-k^2)\cos L + 2hk\sin L \\ \frac{r}{1+h^2+k^2}(\sin L - (h^2-k^2)\sin L + 2hk\cos L) \\ \frac{2r}{1+h^2+k^2}(h\sin L - k\cos L) \end{bmatrix} \quad (1)$$

with $\alpha^T = [p, f, g, h, k, L]$, being related to the keplerian orbital elements through the following equations:

$$\begin{aligned} p &= a(1-e^2) \\ f &= e\cos(\omega+\Omega) \\ g &= e\sin(\omega+\Omega) \\ h &= \tan\left(i/2\right)\cos\Omega \\ k &= \tan\left(i/2\right)\sin\Omega \\ L &= \nu+\omega+\Omega \end{aligned} \quad (2)$$

the velocity and accelerations can be computed by differentiation:

$$\begin{aligned} v &= \frac{dr}{dt} = \frac{dr}{dL}\frac{dL}{dt}; \quad a = \frac{dv}{dt} = \frac{dv}{dL}\frac{dL}{dt} \\ \frac{dr}{dL} &= \sum_{i=1}^{5} \frac{\partial r}{\partial \alpha_i}\frac{\partial \alpha_i}{\partial L} + \frac{\partial r}{\partial L} \end{aligned} \quad (3)$$

In order to obtain the set of pseudo-elements that satisfies exactly the conditions at boundaries, the following nonlinear programming problem must be solved:

$$\begin{aligned} &\mathbf{r}(\boldsymbol{\alpha}(L_0), L_0) = \mathbf{r}_0; \quad \mathbf{v}(\boldsymbol{\alpha}(L_0), L_0) = \mathbf{v}_0; \\ &\mathbf{r}(\boldsymbol{\alpha}(L_f), L_f) = \mathbf{r}_f; \quad \mathbf{v}(\boldsymbol{\alpha}(L_f), L_f) = \mathbf{v}_f; \end{aligned} \quad (4)$$

anyway for low values of the acceleration, as in the case of low-thrust engines, it is sufficient to solve the easier linear problem for the boundary condition of the pseudo-elements:

$$\boldsymbol{\alpha}(L_0) = \boldsymbol{\alpha}_0; \quad \boldsymbol{\alpha}(L_f) = \boldsymbol{\alpha}_f; \quad (5)$$

However, this just gives an approximation of the satisfaction of the boundary constraints on the velocity vector. For a more accurate trajectory description, the solution obtained by Eq. (5) is used as a first guess to solve Eq. (4) exactly.

For each set of pseudo-elements a different trajectory can be generated, connecting two points in the state space. The controls necessary to achieve the imposed shape of the trajectory can be obtained by solving the following system:

$$\mathbf{a}_c = \mathbf{a} - \frac{\mu}{r^3}\mathbf{r}; \quad m_f = m_0 e^{\int_{t_0}^{t_f} -c|a_c|dt} \quad (6)$$

where **a** is the acceleration vector acting on the spacecraft, and $\mathbf{a}_c$ is the control acceleration vector due to the propulsion system, whereas m is the mass of the spacecraft. Additionally the following constraint on the time of flight must be satisfied:

$$t_f - t_0 = \int_{L_0}^{L_f} \frac{dt}{dL} dL \quad (7)$$

For this study the maximum deliverable thrust is a simple function of the inverse of the square of the distance from the Sun, and of the power P_0 delivered at 1 AU:

$$T_{\max} = \frac{P_0}{r^2} \quad (8)$$

This approach is extremely fast and the computational cost extremely low since no propagation or collocation is necessary.

Of course the thrust profile, though constrainable, is a direct consequence of the shape and must be considered only as a first guess useful for further, more refined optimization. For the analysis conducted in this paper the following shape has been used:

$$\tilde{\boldsymbol{\alpha}} = \tilde{\alpha}_0 + \tilde{\alpha}_1 e^{\lambda(L-L_0)} \quad (9)$$

where $\lambda = [\lambda_1, \lambda_2, \lambda_3]^T$ is a set of parameters shaping each pseudo-element.

A full trajectory made of low-thrust, coast arcs and multiple swing-bys is composed by patching conic arcs and shaped arcs together. For a sequence of N_P planets, the problem is divided into k phases, with $k = 1 \ldots . N_P$-1, then for each phase k the sequence of coast arcs and thrust arcs is fixed a priori: depending on the trajectory under study, each phase can be characterized by a pure thrust arc, a thrust-coast arc, a coast thrust arc or a pure coast arc with a deep-space maneuver. Therefore, two points in space can be connected either by a conic arc, solution of a Lambert's targeting, or by a low-thrust arc, solution of the proposed inverse method, which represents a sort of extension of the Lambert's algorithm.

The preliminary design of a general low-thrust multiple gravity assist trajectory is here formulated as a global optimization problem of the form:

$$\begin{aligned} &\min \quad F(\mathbf{y}) \\ &\text{with } \mathbf{y} = [\mathbf{z}, \mathbf{y}_r] \\ &\text{and } \mathbf{y}_r \in D_r \subseteq \Re^n; \mathbf{z} \in D_i \subseteq Z \\ &\textit{sub.to} \\ &\mathbf{h}(\mathbf{y}) = \mathbf{0} \\ &\mathbf{g}(\mathbf{y}) \le \mathbf{0} \end{aligned} \quad (10)$$

and the solution vector is:

$$
\begin{aligned}
&\mathbf{y} = [n_R^{(1)},..,n_R^{(k)},..,n_R^{(N_p-1)}, v_\infty^E, \psi, \theta,\\
&\mathbf{v}_{in}^{(1)},..,\mathbf{v}_{in}^{(k)},..,\mathbf{v}_{in}^{(N_p-1)},\\
&t_1,..,t_i,..,t_{N_p}, \boldsymbol{\lambda}^{(1)},..,\boldsymbol{\lambda}^{(k)},..,\boldsymbol{\lambda}^{(N_p-1)},\\
&r_p^{(1)},..,r_p^{(k)},..,r_p^{(N_p-2)},\\
&...,\eta^{(1)},...,\eta^{(k)},...,\eta^{(N_P-2)},\\
&\varepsilon^{(1)},...,\varepsilon^{(k)},...,\varepsilon^{(N_P-2)}]^T
\end{aligned} \tag{11}
$$

$n^{(k)}_R$ is an integer defining the number of revolutions around the Sun for each phase k, $r^{(k)}_p$ is the pericenter-radius for each flyby divided by the planet mean radius and $\eta^{(k)}$ represents an auxiliary angle used to identify the plane of the swing-by hyperbola [12], while ε represents the time of the deep space maneuver in the case of a coast arc, or the switching instant in the case of a mixed thrust-coast arc.

Constraints on the maximum level of thrust and on the terminal time are defined by:

$$g_j = \left|\mathbf{u}^{(k)}\right| \le u_{\max}^{(k)} \quad \mathrm{j = k = 1,..,N_p - 1} \tag{12}$$

$$h_j(\mathbf{y}) = ToF^{(k)}(\mathbf{y}) - (t_j - t_{j-1}) = 0 \quad \mathrm{j = 2,..,N_p} \tag{13}$$

The optimization problem in Eq. (10) would be typically reformulated through an indirect solution method, with static penalty parameters and a measurement of the constraints violation

$$f_p(\mathbf{y}) = c_1 \sum_{i=1}^{k} \left|h_i(\mathbf{y})\right| + c_2 \sum_{i=1}^{K} \left\langle g_i(\mathbf{y})\right\rangle + c_3 F(\mathbf{y}) \tag{14}$$

The constants c_1, c_2, c_3 would require to be empirically tuned in order to converge to a set of feasible solutions satisfying firstly the time constraint, which represents a physical constraint, and secondly the thrust constraint, which is related to technological limitations.

However, since the global optimization technique employed in this work is devised to handle inequality constraints directly, a more effective approach has been adopted. The constraint on the time of flight has been explicitly solved by translating it into an inequality constraint which requires the actual time of flight Eq.(7) to have a maximum two percent difference with respect to the required T, while the constraint on the maximum acceleration attainable has been weighted into the objective function with a penalty term. This approach has proven to be the best compromise for the effective solution of the constrained problem. In fact, the time constraint has a high priority in the design process, since it represents a physical constraint, while the thrust constraint, which is a technological constraint, is of secondary importance in the solution of a preliminary design problem. In this way, at first, solutions which are physically meaningful are found, and then among the feasible solutions the propellant mass ratio is optimized while satisfying the maximum acceleration available.

4. PROBLEM FORMULATION

In many practical cases it is required to identify multiple optimal solutions and therefore to reconstruct a set of values and not just a single one. Now if more than one solution exists within a required solution domain D the interest could be more to find a number of solutions forming a feasible set X, rather than finding the global optimum with a high level of accuracy.

Therefore problem in Eq. (10) can be translated in the more general problem of finding a feasible set X of solutions x such that a given property $P(x)$ is true for all $x \in X \subseteq D$:

$$X = \{\mathbf{x} \in D \mid P(\mathbf{x})\} \tag{15}$$

where the domain D is a hyper-rectangle defined by the upper and lower bounds on the components of the vector x:

$$D = \left\{x_i \mid x_i \in [b_i^l, b_i^u] \subseteq \Re, i = 1,...,n\right\} \tag{16}$$

All the solutions satisfying property P are said to be optimal with respect to P or P-optimal and X can be said to be a P-optimal set. As an example in the case of multi-objective optimization, if P is a dominance condition or Pareto optimality condition for the solution x then the solution is Pareto-optimal if $P(x)$ is true. In the case of a single objective function, the set X contains all solutions that are in the basin of attraction of a local minimizer.

Now property P might not define a unique set, therefore we can define an optimal set X_o such that all the elements of X_o dominate the elements of any other X:

$$X_o = \left\{\mathbf{x}^* \in D \mid P(\mathbf{x}^*) \wedge \forall \mathbf{x} \in X \Rightarrow \mathbf{x}^* \succ \mathbf{x}\right\} \tag{17}$$

where represents the dominance of the x^* solution over the x solution.

5. OPTIMIZATION APPROACH

Problem in Eq. (15) has been solved with an innovative hybrid global optimization method that combines a deterministic domain decomposition technique with a stochastic type search called multi-agent collaborative search.

5.1 Multiagent Collaborative Search

The global optimization problem formulated in the previous chapter has been solved using a novel procedure based on a hybridization of a domain decomposition scheme and an agent-based global optimization approach. A discussion about the performances of the algorithms is beyond the scopes of this paper and has already been developed in other works. In the following only a brief reminder of the basic working principles of the algorithm will be presented in order to better understand the results obtained in this paper.

The idea is to use a population of agents to explore the solution space $D = D_r \cup D_i$. Then a branching scheme, dependent on the findings of the agents, is used to partition the solution domain in subdomains. On each subdomain a new evolutionary search is performed. The process continues until a number of good minima and eventually the global one are found.

Each solution y is associated to an agent and is represented by a string, of length n, containing in the first s components integer values and in the remaining $n - s$ components real values. A hypercube S enclosing a region of the solution space surrounding each agent, is then associated to y. In analogy with neighborhood search techniques, the solution space is explored locally by each agent by implementing a set of actions. This allows to acquire information about the landscape within the associated region S. The local search space S is adaptively contracted or expanded during the search and for a portion of the population it extends to cover the entire domain D. All the agents form a population that explore globally the search space.

At every generation, a selected number of agents communicate their findings to the others in order to evolve the entire population towards a better status. The entire process of exploring locally-globally and then sharing information is here called multiagent collaborative search (MACS) [9].

During the MACS step a discoveries-resources balance is maintained: a level of resources is associated with each agent and is reduced or increased depending of the number of good findings.

5.2 Domain Decomposition and Pruning

In order to improve search space exploration, the initial domain D is progressively decomposed into smaller domains $D_l \subseteq D$ according to a branching scheme. A branching scheme is represented by a set I_s containing the indices of the coordinates that have to be split and a set C_B containing the cutting point for each coordinate. The initial set I_s is defined by the user while the set C_B contains the middle point of the interval defining each coordinate. After each run of MACS the branching scheme is adapted depending on the outcome of the MACS. If a coordinate has a high number of clusters and a low number of cuts then it is included in the set I_s. The cutting point is then recomputed as the middle point between the cluster containing the best fitness and the cluster containing the worst fitness. If just one cluster exists then one of the boundaries is used instead of the other cluster.

The subdomains D_l with the least number of collected samples among the ones containing elements of X is selected for further decomposition provided that the number of times n_b its parent subdomains have been branched without improvement is below a given threshold. This strategy, however, excludes from further exploration all the subdomains containing no elements of X. Therefore when an exhaustive search is required, the following merit function is used:

$$\psi_{D_l} = (1-\upsilon)\varpi_{D_l} + \upsilon\varphi_{D_l} \tag{18}$$

where ϖ_{D_l} is the density of function evaluations in D_l, φ_{D_l} is best fitness in D_l and n is a weighting factor used to favor either convergence or exploration. For multiobjective problems, in the following, we use the former strategy.

5.3 Incremental Search Space Reduction

In order to reduce the search space, an incremental pruning procedure has been applied to the search for optimal solutions. In case of multiple encounters, instead of solving all-at-once the entire trajectory, the problem is decomposed into subproblems corresponding to each planet-to-planet leg. Starting from the first direct transfer from the Earth to the first encounter, the algorithm search for all potentially interesting launch windows and transfer times. The rest of the search space is then pruned out. Once a reduced solution domain is obtained a second leg is added to the trajectory and optimization proceeds to look for a new optimal set of solutions. The other legs of the trajectory are then incrementally added to the list and optimized.

6. EARTH-NEO SAMPLE RETURN MISSIONS

Sample return missions require a rendezvous with an asteroid, a stay time in orbit and a return leg to the Earth. If more than one asteroid is considered a number of asteroid-asteroid transfers are required. This may be considerably demanding in terms of total impulse and suggests the use of low-thrust propulsion. For this analysis, at first, Earth-asteroid, asteroid-Earth and asteroid-asteroid direct transfers with zero departure and arrival velocity have been studied. The choice of the asteroid was free and the objective was to minimize the propellant mass. This preliminary analysis allows identifying potentially interesting combinations of asteroids for a sample return mission. After this preliminary investigation more complex trajectories involving a flyby of the Earth have been investigated.

Launch and encounter opportunities have been searched in a time frame of 13 years, starting from January 1st, 2012, for all the asteroids reported in Table 1.

This analysis has been performed considering two different groups of possible solutions. Group I collects all the solutions with an outbound transfer toward one of the asteroids from 19 to 23, an asteroid-asteroid transfer to one of the asteroids from 15 to 18 and a return leg from these asteroids.

Figure 1 reports all the solutions with an estimated (non-optimized) propellant consumption for each leg below 150 kg, for the spacecraft configuration in Table 2. Dots represent outbound trajectories, circles asteroid-asteroid transfers and diamonds inbound transfers. For a given identification number, Fig. 1 shows how many transfer opportunities with a cost lower than 150 kg are available. The launch date must be intended at the departure point, therefore for inbound at the asteroid, for outbounds at the Earth and for asteroid-asteroid transfers at the former asteroid.

For example, asteroid 17 has a wide range of return opportunities to the Earth and a number of transfer options from both asteroid 21 and 23. These last two asteroids present a number of outbound solutions from the Earth. Thus asteroid 17 is a good candidate for a multiple-asteroid sample return mission in which the first asteroid is either the number 21 or the number 23.

Group II collects all solutions with an outbound leg toward asteroids from 15 to 18, an asteroid-asteroid transfer to asteroids from 19 to 23 and a return leg from these celestial bodies. In this case there is a higher number of mission opportunities, even though only asteroid 21 and 23 seems to be feasible targets (see Fig. 2). However, the latter has just one return opportunity before 6000 (MJD2000), which does not allow, in the required time frame, an asteroid-asteroid transfer.

This preliminary analysis suggests that with no swingby the most convenient mission options in the required time frame are: Earth-1997XR-1999AO10-Earth, Earth-2004MN4-1999AO10-Earth, Earth-1999AO10-2004MN4-Earth, Earth-1989ML-2004MN4-Earth. Additional interesting sequences are Earth-1999AO10-2003WP25-Earth and Earth-1999AO10-2000LG6-Earth.

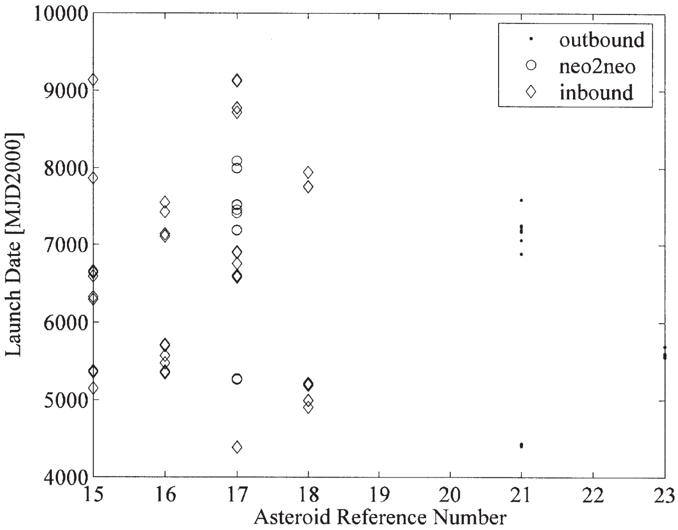

Fig. 1 Earth-NEO-NEO-Earth options: group I.

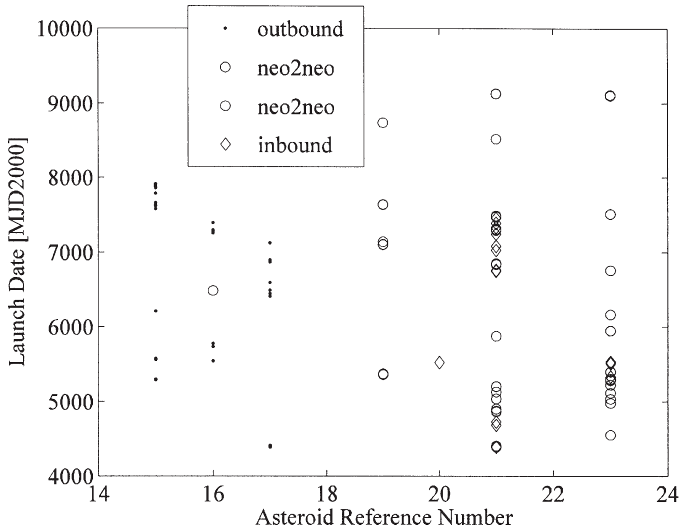

Fig. 2 Earth-NEO-NEO-Earth options: group II.

In Table 3 some of these options are listed, considering an arrival velocity at the Earth of 3.3 km/s and a departure $C_3 = 0$. Notice that the propellant mass is not optimized and from experience [9] a significant reduction of up to half of the value reported here is expected. The mass ratio in the Table is that between the propellant and the spacecraft mass at the beginning of the transfer leg. It is remarkable that two of the solutions found with this procedure, and possibly more with a free target search, are in a very good agreement with two solutions presented in work by JAXA [7], thus confirming the investigation capability of the proposed approach. However, our result is more conservative since zero departure and arrival velocity is considered.

Finally as it can be seen from the third solution in Table 3 and from the summary of different transfer options presented in figs. 1 & 2, the approach presented is suitable for locating a large number of different options, with a low computational cost, which makes such a methodology appealing for an extensive preliminary investigation.

TABLE 3: *Sample Return Options.*

Transfer Leg	t_0 (MJD2000)	TOF (day)	m_p/m_0
E-1999AO10	4407.8	433.4	0.16
1999AO10-2003WP25	5022.5	820.9	0.16
2003WP25-E	6208.0	867.0	0.06
E-1999AO10	4383.5	480.5	0.15
1999AO10-2000LG6	4969.8	750.8	0.24
2000LG6-E	5810.0	745.0	0.12
E-1999AO10	4407.8	433.4	0.16
1999AO10-2004MN4	5126.3	769.8	0.165
2004MN4-E	6054.1	946.0	0.06

7. EARTH-NEO IMPACT MISSIONS

Impact missions aim at crashing a spacecraft onto an asteroid in order to observe and analyze the fragments generated during the impact. Three different transfer categories have been investigated: direct Earth-asteroid, Earth-Earth-asteroid, Earth-Venus-asteroid. In all the three cases the set X is made up of all those solutions that have an impact velocity higher than 10 km/s and the minimum total Δv for launch and deep space maneuvers.

For impact missions only chemical options have been considered since a low total Δv is expected, with respect to rendezvous and sample return missions, and the spacecraft has to be expendable.

In order to find the most appropriate transfer for a given set of asteroids, at first a free final target analysis has been performed. In this analysis the target asteroid is a free parameter. The result for asteroids from 15 to 23 has been plotted in fig. 3 where the total Δv of the mission is plotted against the identification number of the asteroids. Dots represent solutions with a direct transfer from the Earth to the asteroid, while circles represent transfers via a Venus swingby and diamonds via an Earth swingby. Figure 3 shows, for the asteroids having an identification number ranging from 15 to 23, the best transfer options in order to achieve an impact having a relative velocity greater than 10km/s. As an example, for the last three asteroids (id.21-22-23) in the considered time frame a direct impact is more convenient than a transfer via Venus or the Earth. This is consistent with these NEOs having a high level of impact risk.

All the others benefit from a gravity assist maneuver, in particular asteroid number 20 (2002at4) can be efficiently reached with a swing-by of the Earth while asteroid 18 (2000LG6), which is not a convenient target for a direct impact, becomes interesting if a swing-by of Venus is considered. Notice that asteroid 15 (1989ML) benefits from a swing-by of Venus as correctly proposed by the Don Quijote mission.

The most convenient asteroid for a direct impact is number 22 (2004VD17), reachable with a relative departure velocity of few hundreds meters per second. This target is quite challenging for a rendezvous due to its high eccentricity and semi-major axis. Figure 4 proposes a possible solution for an all-chemical transfer to asteroid 22. The launch is at 4455.2 (MJD2000) and requires a launch $C_3 = 11.76$ km^2/s^2, three swingbys of Venus and two deep space maneuvers after the first two swing-bys respectively of 0.53 km/s and of 0.29 km/s. The arrival velocity at the asteroid is $v_f = 1.79$ km/s with a total transfer time of 2097.7 days. The cheapest impact solution for asteroid 22, with a $\Delta v = 0.31$ km/s, reaches the asteroid at 7011 (MJD2000), that is to say 459 days after the rendezvous mission which allows more than one year of operations and scientific investigations before the impact.

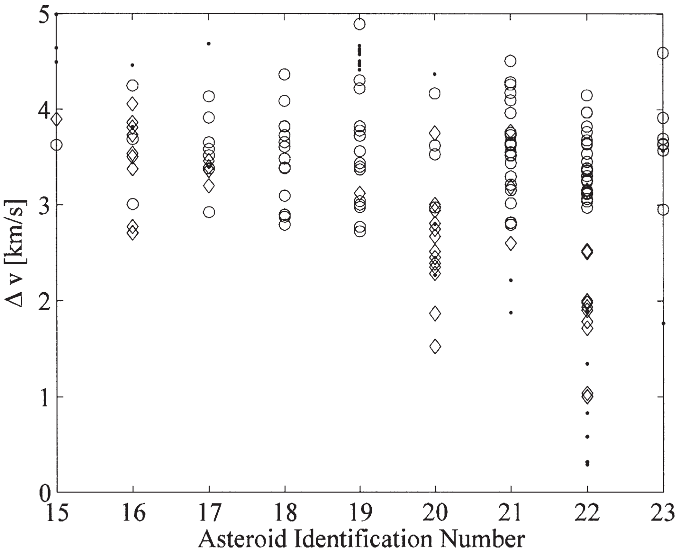

Fig. 3 Required Δv for an impact at v>10 km/s

The high total Δv required for this asteroid and in particular the high arrival velocity would suggest the use of electric propulsion. The electric option will be investigated in a future work.

Another critical asteroid for a rendezvous mission is 2002at4. In this case a multiple swing-by of Venus gives little benefit since the perihelion is at the Earth distance; therefore the proposed solution is a multispiral trajectory with low-thrust propulsion (see Fig. 5). For a specific impulse of 3000 s the estimated (non-optimized) propellant ratio is 0.39 and a transfer time of 1683.6 days with a launch date at 9131.6 (MJD2000).

8. MULTIPLE SAMPLE AND RETURN MISSIONS WITH EARTH SWINGBYS

As a further step in the analysis of sample return missions, the possibility of having multiple encounters have been investigated. In order to consider small class missions, and to keep launch velocity sufficiently low, i.e. lower than 1.3 km/s, a strategy consisting of Earth flyby and asteroid encounter has been devised. In the following the sequence E-E-NEO-E-NEO-E, has been analyzed for all targets available in Table 1. Exploiting the incremental and pruning approach first the subsequence E-E-NEO has been investigated for a set of launch dates in the range [3650, 7200] looking for potential targets for a first encounter. It resulted that at least 8 asteroids, belonging to the Amor, Apollo and Athen classes, can be reached with propellant mass ratio lower than 22%. Thus, assuming this value as a pruning criterion, the domain has been pruned and the incremental search procedure has been further applied in order to look for further potential targets, to extend the mission. The procedure is then incrementally iterated in order to identify targets for a rendezvous with a second asteroid, and for the final return to Earth. It is important to remark that this approach is segmented into subsequences consisting of three bodies, where the last body is always a rendezvous with an asteroid or the final transfer to Earth that closes the tour. This approach, although computationally heavier than a single transfer incremental approach, turns out to be better for multiple rendezvous trajectory, since the flyby manoeuvre is never split into two different phases, thus avoiding to select options having a low propellant consumption on the first leg, but resulting into too low relative velocity at the flyby, which would prevent reaching the asteroid in the following arc.

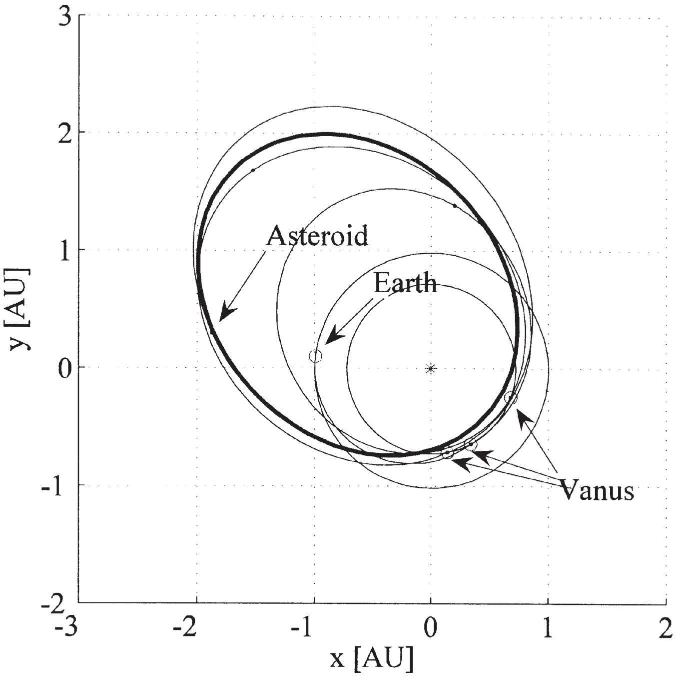

Fig. 4 2004VD17 rendezvous trajectory.

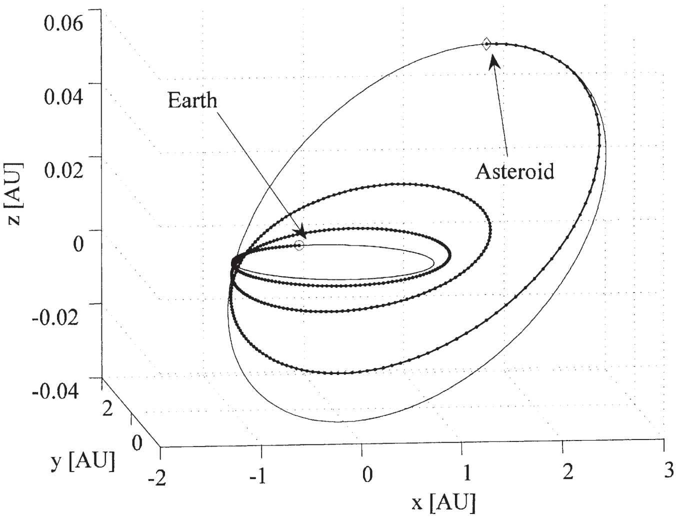

Fig. 5 Low-thrust transfer to 2002at4.

As an example of this search methodology a series of multiple rendezvous and sample and return mission has been designed, looking for solution having a thrust level in the range of 50-60 mN at 1 AU. As a result of the first investigation asteroid 2003WP5 (id. 19) has been selected as a potential target for a multiple rendezvous sequence, and six different tours were designed. Stop-time at each asteroid has been introduced as an optimization parameter and it can vary between 45 and 360 days. Table 3 shows the potential sequences for a double sample and return mission, with a rendezvous of either SG344 (id.3) or 1999YB (id.12), or 1989ML (id.15), 1999AO10 (id.17) or 2002LG6 (id.18) and 2004MN4 (id.21). It is interesting to notice that with a multiple rendezvous tour it is possible to go from Athen- to Amor-class asteroids, thus enhancing the scientific value of this kind of missions.

It can be noticed that all these tours have a total duration between ten and twelve years, and a quite low propellant mass ratio. Additionally it should be said that since a shape is used to transcribe low-thrust arcs, the resulting control is sub-optimal, thus providing a conservative estimation of both the propellant mass ratio, and of the maximum thrust required.

In order to show that this preliminary solution found by the inverse method are physical, and since it is sub-optimal it can be improved if optimized with a refined local optimization method, the tour Earth-Earth-2003WP5-Earth-SG344-Earth has been further optimized with DITAN a direct local optimization software based on a direct transcription by finite elements in time [12]. Table 4 compares the features of the preliminary tour with the optimized solution. In particular we compared the preliminary sub-optimal solution found with the inverse method to a couple of different fully optimized solutions found with DITAN. One of them minimizes the integral of the square of the thrust modulus, and the other maximizes the final mass.

As it can be seen, the agreement between the preliminary solution obtained with the inverse method and the refined optimized solution is quite good, and this confirms that the methodology used to transcribe low-thrust arcs, by a reduction of free parameters, is powerful and well describes on average the optimal control solution. This can be seen comparing the value of the propellant mass ratio found preliminarily with the value found in the accurate optimization with objective function quadratic control (the first value in the optimized column); in this case the difference is just 3% of the total mass. This means that the proposed simplified transcription is well suited to find first guesses, which are very close, in the solution space, to those solutions minimizing the integral of the thrust modulus. On the other hand a larger improvement can be achieved when the final mass is optimized (the second value for the propellant mass ratio in last column of Table 5). In this case the difference with respect to the preliminary results are larger, due to the sub-optimality of the shape approach. In fact, although it correctly locates the optimal launch dates and transfer times, it can only approximately reproduce a bang-bang (on-off) structure.

As a final remark the inverse method, combined with the global search, proves to be highly efficient in providing reliable preliminary information on the optimal solution, in terms of dates, velocities and optimal control structure. Thus the results obtained should be considered as a good first guess for further optimization and a conservative estimation of the feasibility of a complex mission, within a given mass and thrust level. Figures 6, 7 and 8 show the various phases of the sample and return tour to 2003WP5 and SG344.

9. CONCLUSIONS

In this paper an analysis of different transfer options to asteroids has been presented. Three mission scenarios have been investigated: multiple-asteroid sample-return missions with low-thrust, ballistic impacts, multiple-asteroid sample-return missions via an Earth swingby.

In order to perform the investigation of many different mission options efficiently, a novel computational approach has been proposed. It combines a simplified trajectory model with a particular hybrid global search method. The two-point boundary value problem associated with the design of a generic low-thrust transfer is solved by imposing a particular shape to the orbital elements and by deriving the required thrust control law. This inverse approach provides feasible and sub-optimal solutions without the need for any numerically expensive propagation or collocation of the dynamic equations. Since the solution of every two-point boundary value problems takes few seconds, a large number of feasible trajectories can be generated in a very short time.

TABLE 4: *Comparison Between Preliminary and Optimized Tour.*

#	Preliminary	Optimized
Launch date [MJD]	5464	5479
Escape Vel [km/s]	1.18	1.18
E-E ToF [day]	348	355
E-2003WP5 Tof [day]	863	869
2003WP5 Op. time [day]	185	156
2003WP5-E Tof [day]	576	572
E-SG344 ToF [day]	890	870
SG344 Op. Time [day]	92	97
SG344-E ToF [day]	903	862
Earth Hyp Vel. [km/s]	1.0	1.9
Total ToF [year]	10.51	10.35
m_p/m_0	0.26	0.23/0.18

TABLE 5: *Summary of Different Tours.*

#	Tdep [MJD]	Duration [y]	Prop. Mass Ratio	Max Thrust [mN]
19-3	5464	10.51	0.26	50
19-12	5464	9.82	0.36	52
19-18	5464	12.48	0.21	52
19-21	5464	11.22	0.49	64
19-15	5464	11.55	0.34	64
19-17	5464	10.19	0.42	60

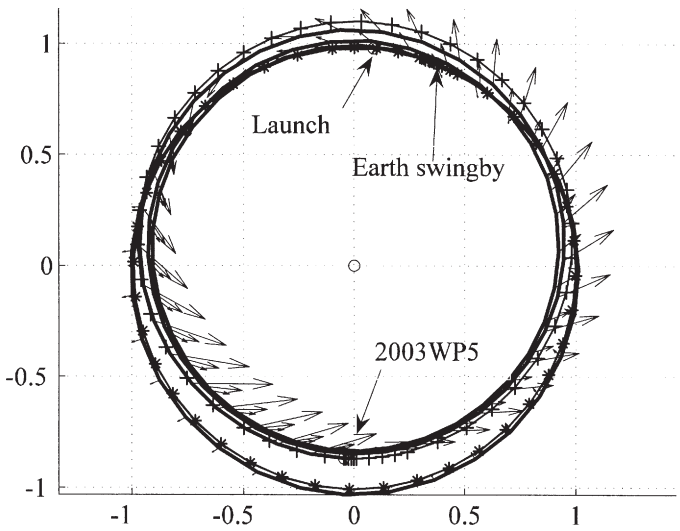

Fig. 6 Earth-Earth-2003WP5.

The comparison between the preliminary solutions and the optimized solutions, have demonstrated how the proposed approach can provide results that are accurate enough to allow a credible assessment of a large number of mission options. The requirement for a large number of potential targets and the wide variety of mission options requires a global exploration the search space. However, traditional enumerative approaches would have been too computationally expensive. On the other hand basic stochastic approaches, such as Evolutionary Algorithms, would not have been sufficiently exhaustive.

The proposed global search, instead, exploits the benefits of evolutionary methods at exploring complex domains, and the systematic reduction of the search space by a branch and prune procedure. Although the presented results cannot be considered

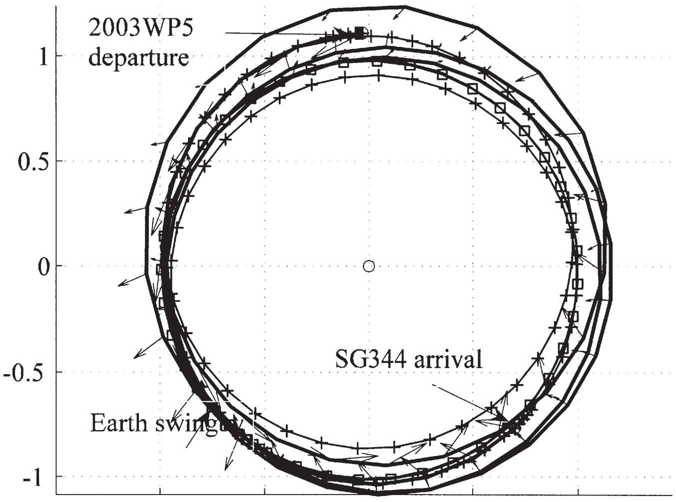

Fig. 7 2003WP5-Earth-SG344.

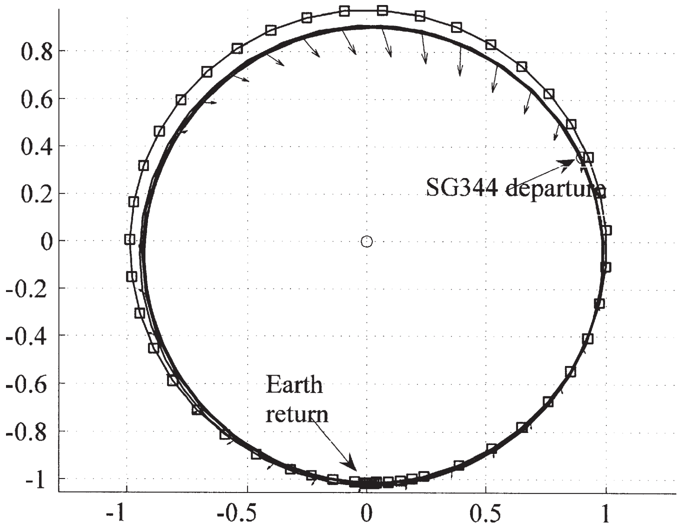

Fig. 8 SG344-Earth.

completely exhaustive, this combined approach has demonstrated to provide an efficient solution to the problem of designing multiple target, low-thrust trajectories.

Finally it has been shown that small-class spacecraft can accomplish, through the use of low-thrust propulsion, very different and complex missions, such as sample and return, multiple encounters and combined multiple rendezvous and Earth gravity assist. Additionally some of the results presented might be valuable for further studies for those interested in mission to NEO.

The effectiveness of the proposed approach suggests that it may be interesting to proceed in this field of research toward the objective of providing an extensive and deep characterization of transfer types to NEO. For this reason future work will approach the problem in a more systematic way, for example defining the best sequence of swing-bys or multiple-rendezvous missions, with respect to the classes of asteroids in terms of physical or orbital properties.

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