Office de la Propriété
Intellectuelle
du Canada

Un organisme
d'Industrie Canada

Canadian
Intellectual Property
Office

An agency of
Industry Canada

CA 2713363 C 2016/05/17

(11)(21) **2 713 363**

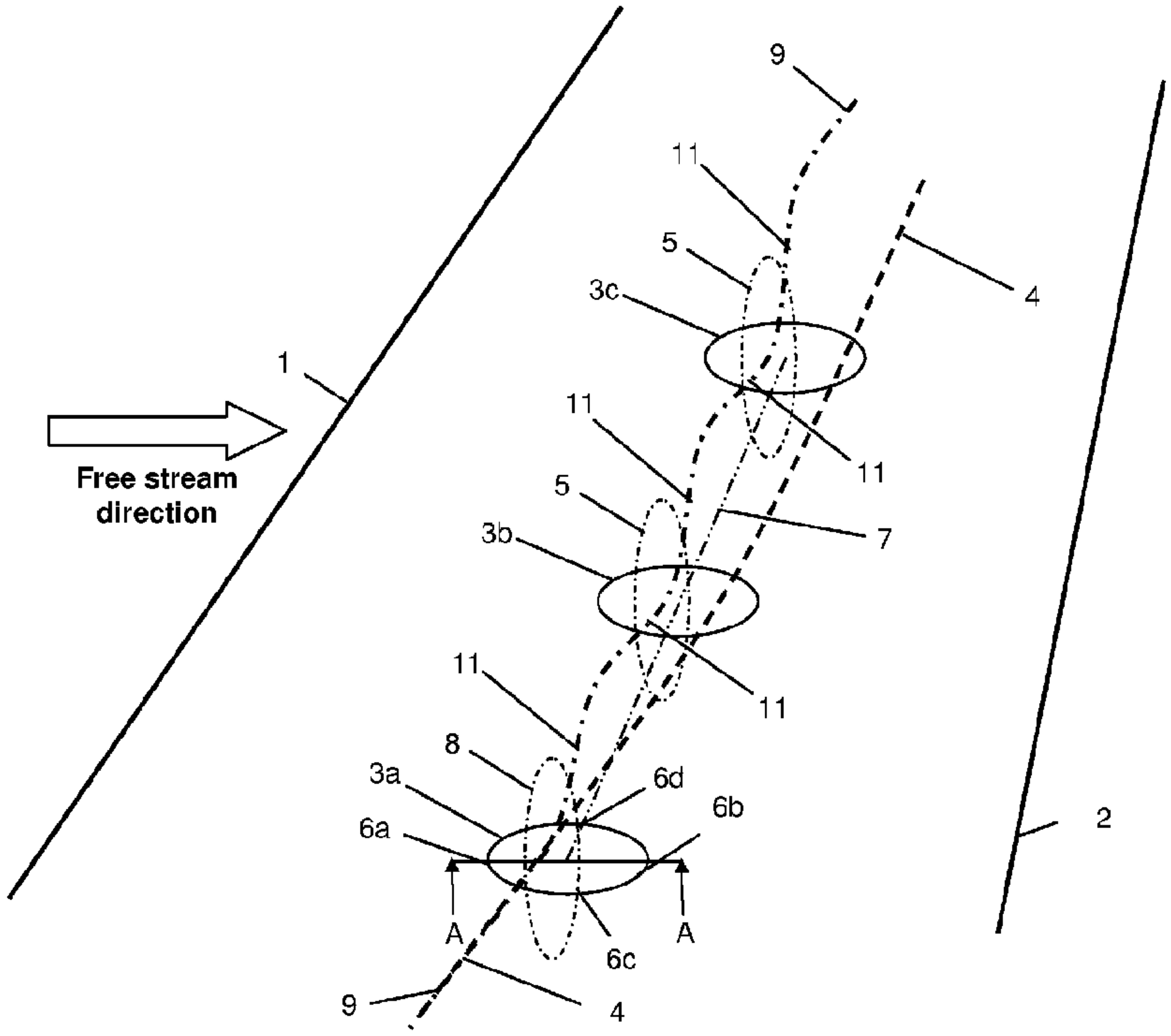
(12) **BREVET CANADIEN**
CANADIAN PATENT

(13) **C**

(86) Date de dépôt PCT/PCT Filing Date: 2009/02/17
(87) Date publication PCT/PCT Publication Date: 2009/09/03
(45) Date de délivrance/Issue Date: 2016/05/17
(85) Entrée phase nationale/National Entry: 2010/07/26
(86) N° demande PCT/PCT Application No.: GB 2009/050150
(87) N° publication PCT/PCT Publication No.: 2009/106869
(30) Priorité/Priority: 2008/02/29 (GB0803727.7)

(51) Cl.Int./Int.Cl. *B64C 23/04* (2006.01)
(72) Inventeur/Inventor:
WOOD, NORMAN, GB
(73) Propriétaire/Owner:
AIRBUS OPERATIONS LIMITED, GB
(74) Agent: OYEN WIGGS GREEN & MUTALA LLP

(54) Titre : STRUCTURE AERODYNAMIQUE COMPORTANT UNE SERIE DE BOSSES ANTICHOCS
(54) Title: AERODYNAMIC STRUCTURE WITH SERIES OF SHOCK BUMPS



(57) Abrégé/Abstract:
An aerodynamic structure (1) comprising a series of shock bumps (3a, 3b, 3c) extending from its surface. The shock bumps are distributed along a line (7) with a smaller mean angle of sweep than an unperturbed shock (4) which would form adjacent to the surface during transonic movement of the structure in the absence of the shock bumps. Instead of being distributed along the line of the unperturbed shock, the shock bumps are distributed along a line which is less swept than the mean angle of sweep of the unperturbed shock. When the structure is moved at a transonic speed; a shock forms adjacent to its surface and the shock bumps perturb the shock (9) so as to reduce its angle of sweep.

(12) INTERNATIONAL APPLICATION PUBLISHED UNDER THE PATENT COOPERATION TREATY (PCT)

(19) **World Intellectual Property Organization**
International Bureau

(43) **International Publication Date**
3 September 2009 (03.09.2009)

PCT

(10) **International Publication Number**
WO 2009/106869 A3

(51) **International Patent Classification**:
B64C 23/04 (2006.01)

(21) **International Application Number**:
PCT/GB2009/050150

(22) **International Filing Date**:
17 February 2009 (17.02.2009)

(25) **Filing Language**: English

(26) **Publication Language**: English

(30) **Priority Data**:
0803727.7 29 February 2008 (29.02.2008) GB

(71) **Applicant** *(for all designated States except US)*: **AIRBUS UK LIMITED** [GB/GB]; New Filton House, Bristol BS99 7AR (GB).

(72) **Inventor; and**
(75) **Inventor/Applicant** *(for US only)*: **WOOD, Norman** [GB/GB]; Airbus UK Limited, Filton, Bristol BS99 7AR (GB).

(74) **Agent**: **SCOTT, Alistair**; IP Dept, Building 09A, New Filton House, Filton, Bristol BS99 7AR (GB).

(81) **Designated States** *(unless otherwise indicated, for every kind of national protection available)*: AE, AG, AL, AM, AO, AT, AU, AZ, BA, BB, BG, BH, BR, BW, BY, BZ, CA, CH, CN, CO, CR, CU, CZ, DE, DK, DM, DO, DZ, EC, EE, EG, ES, FI, GB, GD, GE, GH, GM, GT, HN, HR, HU, ID, IL, IN, IS, JP, KE, KG, KM, KN, KP, KR, KZ, LA, LC, LK, LR, LS, LT, LU, LY, MA, MD, ME, MG, MK, MN, MW, MX, MY, MZ, NA, NG, NI, NO, NZ, OM, PG, PH, PL, PT, RO, RS, RU, SC, SD, SE, SG, SK, SL, SM, ST, SV, SY, TJ, TM, TN, TR, TT, TZ, UA, UG, US, UZ, VC, VN, ZA, ZM, ZW.

(84) **Designated States** *(unless otherwise indicated, for every kind of regional protection available)*: ARIPO (BW, GH, GM, KE, LS, MW, MZ, NA, SD, SL, SZ, TZ, UG, ZM, ZW), Eurasian (AM, AZ, BY, KG, KZ, MD, RU, TJ, TM), European (AT, BE, BG, CH, CY, CZ, DE, DK, EE, ES, FI, FR, GB, GR, HR, HU, IE, IS, IT, LT, LU, LV, MC, MK, MT, NL, NO, PL, PT, RO, SE, SI, SK, TR), OAPI (BF, BJ, CF, CG, CI, CM, GA, GN, GQ, GW, ML, MR, NE, SN, TD, TG).

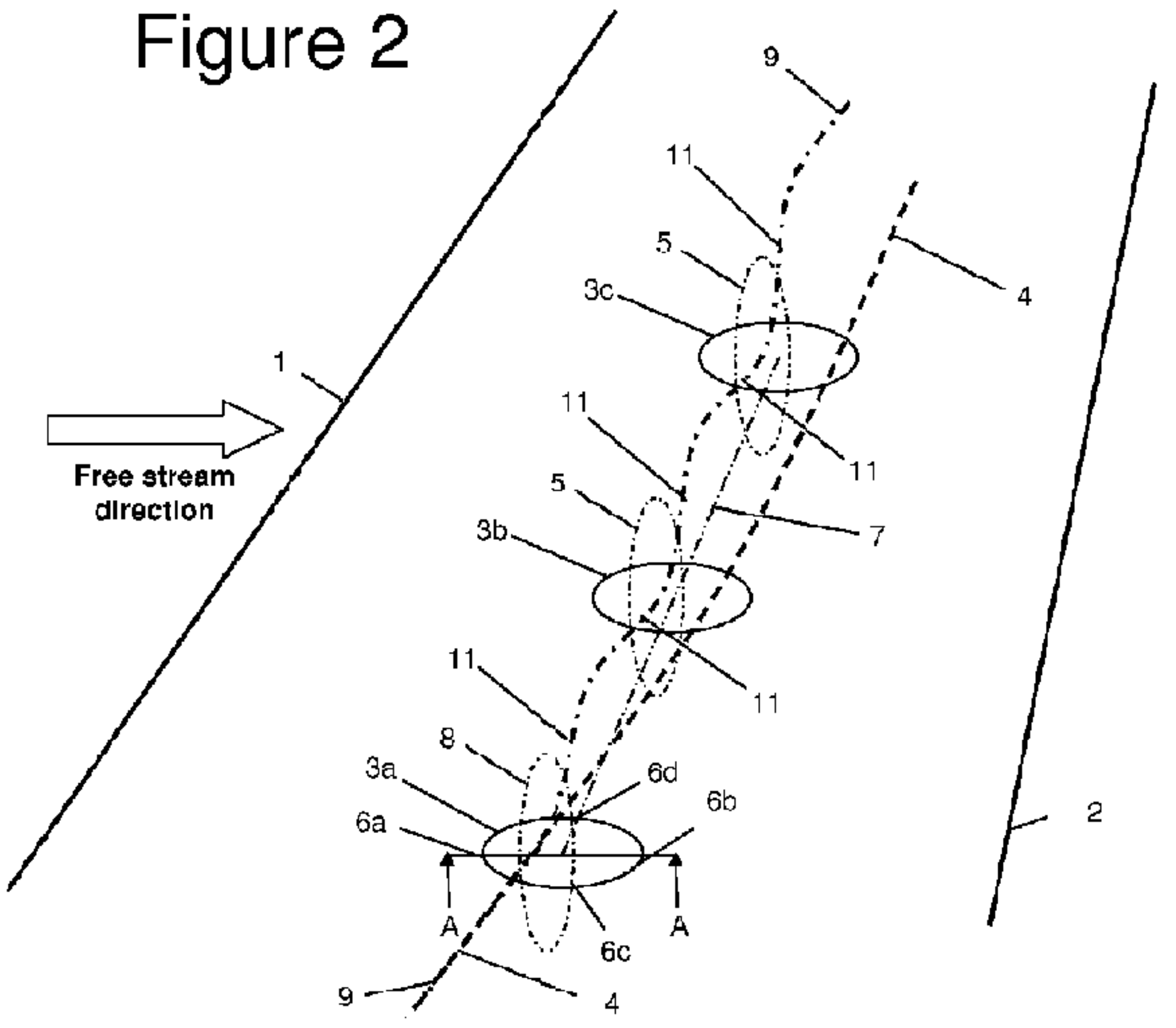
Published:

- *with international search report (Art. 21(3))*
- *before the expiration of the time limit for amending the claims and to be republished in the event of receipt of amendments (Rule 48.2(h))*

(88) **Date of publication of the international search report**:
22 October 2009

(54) **Title**: AERODYNAMIC STRUCTURE WITH SERIES OF SHOCK BUMPS

Figure 2



(57) **Abstract**: An aerodynamic structure (1) comprising a series of shock bumps (3a, 3b, 3c) extending from its surface. The shock bumps are distributed along a line (7) with a smaller mean angle of sweep than an unperturbed shock (4) which would form adjacent to the surface during transonic movement of the structure in the absence of the shock bumps. Instead of being distributed along the line of the unperturbed shock, the shock bumps are distributed along a line which is less swept than the mean angle of sweep of the unperturbed shock. When the structure is moved at a transonic speed; a shock forms adjacent to its surface and the shock bumps perturb the shock (9) so as to reduce its angle of sweep.

AERODYNAMIC STRUCTURE WITH SERIES OF SHOCK BUMPS

FIELD OF THE INVENTION

The present invention relates to an aerodynamic structure comprising a series of shock bumps extending from its surface, and a method of operating such a structure.

BACKGROUND OF THE INVENTION

Figure 1 is a plan view of the upper surface of an aircraft wing. The wing has a leading edge 1 and a trailing edge 2, each swept to the rear relative to the free stream direction.

At transonic speeds a swept shock 4 forms normal to the upper surface of the wing. As described in Holden, H.A. and Babinsky, H. (2003) *Shock/boundary layer interaction control using 3D devices* In: 41st Aerospace Sciences Meeting and Exhibit, January 6-9, 2003, Reno, Nevada, USA, Paper no. AIAA 2003-447, a 3-D shock bump can be used to induce a smeared shock foot with a lambda-like wave pattern.

Conventionally the chord-wise position of such bumps is dictated by the expected position of the shock 4. However for either laminar or turbulent wings the position is a complex function of Mach number and lift coefficient. The wave drag associated with a shock can be alleviated by the use of a 3-D shock bump that will exhibit maximum benefit when the shock is at a particular location on the bump. Hence as the flight conditions vary the shock may move away from this optimal location.

A traditional approach to solve this problem is to deploy trailing edge variable camber to modify the aerofoil shape and hence the shock location and this incurs additional weight and systems complexity. The challenge then is to find a way of fixing the shock wave independent of the shape of the wing section and the span load distribution.

US 2006/0060720 uses a shock control protrusion to generate a shock extending away from the lower surface of a wing.

SUMMARY OF THE INVENTION

A first aspect of the invention provides an aerodynamic structure comprising a series of shock bumps extending from its surface, the shock bumps being distributed along a line with a smaller mean angle of sweep than an unperturbed shock which would form adjacent to the surface during transonic movement of the structure in the absence of the shock bumps.

Instead of being distributed along the line of the unperturbed shock, the shock bumps are distributed along a line which is less swept than the mean angle of sweep of the unperturbed shock. That is, if the unperturbed shock is swept to the rear then the line is either not swept or is swept to the rear by a smaller angle of sweep. Equivalently, if the unperturbed shock is swept forward then the line is either not swept or is swept forward by a smaller angle of sweep. In other words, the shock bumps “un-sweep” the shock.

A second aspect of the invention provides a method of operating an aerodynamic structure comprising a series of shock bumps extending from its surface, the method comprising: moving the structure at a transonic speed; forming a shock adjacent to its surface; and perturbing the shock with the series of shock bumps so as to reduce its angle of sweep.

Typically the shock bumps cause the shock to form a stepped plan-form shape with a series of points of inflection.

Typically each shock bump induces a smeared shock foot with a lambda-like wave pattern.

Typically a first shock bump in the series is positioned in line with the position of the unperturbed shock, and the other shock bumps in the series are positioned either fore or aft of the position of the unperturbed shock (depending on whether the unperturbed shock is swept back or forward respectively).

Typically each bump has a leading edge, a trailing edge, an inboard edge and an outboard edge. The bumps may merge gradually into the surface at its edges or there may be an abrupt concave discontinuity at one or more of its edges.

Typically each bump has substantially no sharp convex edges or points.

Typically the shock bumps are shaped and positioned so as to modify the structure of the unperturbed shock. This can be contrasted with US 2006/0060720 which uses a shock control protrusion to generate a shock which would not otherwise exist in the absence of the shock control protrusion.

The structure may comprise an aerofoil such as an aircraft wing, horizontal tail plane or control surface; an aircraft structure such as a nacelle, pylon or fin; or any other kind of aerodynamic structure such as a turbine blade.

In the case of an aerofoil the shock bumps may be located on a high pressure surface of the aerofoil (that is, the lower surface in the case of an aircraft wing) but more preferably the surface is a low pressure surface of the aerofoil (that is, the upper surface in the case of an aircraft wing). Also each bump typically has an apex which is positioned towards the trailing edge of the aerofoil, in other words it is positioned aft of 50% chord. The apex of the bump may be a single point, or a plateau. In the case of a flat plateau then the leading edge of the plateau is positioned towards the trailing edge of the aerofoil.

BRIEF DESCRIPTION OF THE DRAWINGS

Embodiments of the invention will now be described with reference to the accompanying drawings, in which:

Figure 1 is a plan view of the top of an aircraft wing;

Figure 2 is a plan view of the top of an aircraft wing carrying a series of shock bumps according to a first embodiment of the invention;

Figure 3 is a cross-sectional view through the centre of one of the bumps taken along a line A-A;

Figure 4 is a plan view showing the mean sweep angles of the perturbed and unperturbed shocks, outboard of the first shock bump; and

Figure 5 is a plan view of the top of an aircraft wing carrying a series of shock bumps according to a second embodiment of the invention.

DETAILED DESCRIPTION OF EMBODIMENT(S)

Figure 2 is a plan view of the upper surface of an aircraft wing in a transonic flow similar to the wing of Figure 1. The footprint of a series of shock bumps is indicated at 3a-3c and Figure 3 is a longitudinal cross-sectional view through the centre of one of the bumps taken along a line A-A which is parallel with the free stream direction. An unperturbed shock 4 would form adjacent to the surface of the wing during transonic cruise flight conditions in the absence of the shock bumps.

Each bump protrudes from a nominal surface 5 of the wing, and meets the nominal surface 5 at a leading edge 6a; a trailing edge 6b; an inboard edge 6c; and an outboard edge 6d. Each bump also has an apex point 6e. The lower portions of the sides of bump are concave and merge gradually into the nominal surface 5. For example in Figure 3 the lower portion 7 of the front side of the bump merges gradually into the nominal surface 5 at leading edge 6a. Alternatively there may be an abrupt discontinuity at one or more of the edges of the bump. For instance the lower portion of the front side of the bump may be planar as illustrated in dashed lines at 7a. In this case the front side 7a of the shock bump meets the nominal surface 5 with an abrupt discontinuity at the leading edge 6a. The apex point 6e of the fore/aft cross-section A-A is offset aft of the centre of the bump. The apex 6e of each bump 3a-3c is also positioned aft of 50% chord, typically between 60% and 65% chord. Note that, unlike vortex generators, the bumps have no sharp convex edges or points so the flow remains attached over the bumps when they are operated at their optimum (i.e. when the shock is positioned on the bump just ahead of its apex).

The shock bumps 3a-3c modify the structure of the shock by inducing a smeared shock foot 8 with a lambda like wave pattern shown in Figure 3. When the shock bumps 3a-3c are operated at their optimum with the shock 4 just ahead of the apex 6e of the bump as shown in Figure 3, the smeared foot 8 has a lambda-like wave pattern with a single forward shock 8a towards the leading edge of the bump and a single rear shock 8b positioned slightly forward of the apex 6e. Alternatively, instead of having only a single forward shock 8a, the smeared foot may have a lambda-like wave pattern with a fan-like series of forward shocks. As the local flow is generally just above the sonic condition the perturbation to the flow spreads sideways almost normal to the free stream direction and not along the unperturbed shock 4. This is illustrated in Figure 2 by a perturbed shock line 9 which is coincident with the unperturbed shock 4 until it reaches the first (most inboard) shock bump 3a. At this point the shock bump perturbs the shock so that the perturbed shock line 9 bends forwards as shown. At some span-wise distance from the first bump 3a the flow returns to its unperturbed state and attempts to return to its original chord-wise location. This results in a point of inflection 11 in the perturbed shock line 9. The second bump 3b is placed outboard of the bump 3a and forward of the line 4 to re-perturb the shock knowing that, independent of the original shock location 4, the first bump 3a will be dictating the path of the smeared lambda shock. Similarly the third bump 3b is placed at a suitable position outboard of the bump 3b and forward of the line 4 to re-perturb the shock. More than three shock bumps may be used to extend the process towards the wing tip.

The shock bumps 3a-3c cause the shock to form a stepped plan-form shape 9 with a series of points of inflection 11. Figure 4 is a plan view showing a line 9a representing the mean sweep angle of the perturbed shock 9 and a line 4a representing the mean sweep angle of the unperturbed shock 4 outboard of the first shock bump 3a. As shown in Figure 4, the line 9a is less swept than the line 4a.

The perturbed location 9 of the shock is determined as a function of the flow of the innermost bump 3a and not the lift coefficient or Mach number. This precludes the need for a variable camber system and maintains the bumps operating at or near their optimum for a variety of flight conditions.

The centres of the shock bumps are distributed along a line 10. This line 10 is also less swept than the line 4a. In the example shown in Figure 2, all of the shock bumps 3a-3c are centred on a straight line 10. However in other embodiments the centres of the bumps may not all lie on a straight line, an example being given in Figure 5. In this example the shock bumps 3a-3e are distributed along a zigzag line 10a-10d. The mean sweep angle (indicated by line 10e) of the zigzag line 10a-10d is swept to a lesser degree than the mean sweep angle of the unperturbed shock 4 outboard of the first bump 10a, in a similar manner to the line 10 in Figure 2. Note that the deviation of the zigzag line 10a-10d from the straight mean line 10e is exaggerated in Figure 5 for purposes of illustration.

Although the shock bumps are shown on an upper surface of a wing, similar arrangements could be used in a variety of other applications e.g. around pylons and nacelles. They may also provide a reduction in profile power and noise when applied to the tips of helicopter rotors and propeller blades.

Although the invention has been described above with reference to one or more preferred embodiments, it will be appreciated that various changes or modifications may be made without departing from the scope of the invention as defined in the appended claims.

Claims

1. An aerodynamic structure comprising a series of shock bumps extending from its surface, the shock bumps being distributed along a line with a smaller mean angle of sweep than an unperturbed shock which would form adjacent to the surface during transonic movement of the structure in the absence of the shock bumps.

2. The structure of claim 1 wherein a first shock bump in the series is positioned in line with the position of the unperturbed shock, and the other shock bumps in the series are positioned fore or aft of the position of the unperturbed shock.

3. The structure of claims 1 or 2 wherein each bump has a leading edge, a trailing edge, an inboard edge and an outboard edge.

4. The structure of claim 3 wherein each bump meets the surface at the leading edge, trailing edge, inboard edge and outboard edge.

5. The structure of any one of claims 1 to 4 wherein each bump has substantially no sharp convex edges or points.

6. The structure of any one of claims 1 to 5 wherein the shock bumps are shaped and positioned so as to modify the structure of the shock.

7. The structure of claim 6 wherein the shock bumps are shaped and positioned so as to induce a smeared foot in the shock with a lambda like wave pattern.

8. The structure of any one of claims 1 to 7 wherein the aerodynamic structure is an aerofoil and the surface is a low pressure surface of the aerofoil.

9. The structure of any one of claims 1 to 8 wherein the aerodynamic structure is an aerofoil having a leading edge and a trailing edge, and wherein each bump in the first series has an apex which is positioned towards the trailing edge of the aerofoil.

10. A method of operating an aerodynamic structure comprising a series of shock bumps extending from its surface, the method comprising:

moving the structure at a transonic speed;

forming a shock adjacent to its surface; and

perturbing the shock with the series of shock bumps so as to reduce its angle of sweep.

11. The method of claim 10 wherein the perturbed shock has a stepped plan-form shape with a series of points of inflection.

12. The method according to claims 10 or 11 wherein the shock bumps are used to modify the structure of the shock.

13. The method according to any one of claims 10 to 12 further comprising inducing with each shock bump a smeared shock foot with a lambda-like wave pattern.

14. The method according to any one of claims 10 to 13 wherein the flow over at least one of the shock bumps is substantially fully attached.

Figure 1

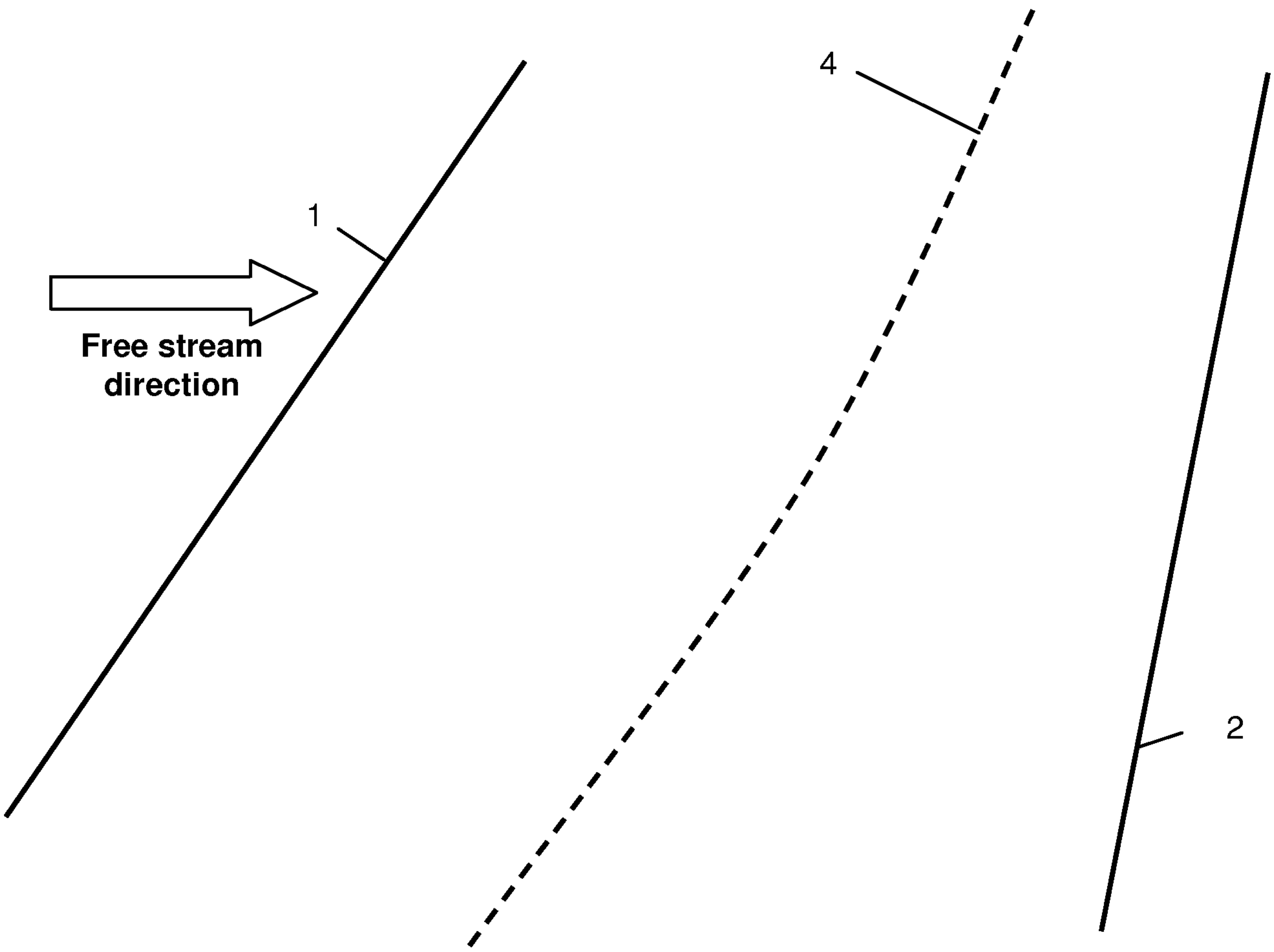
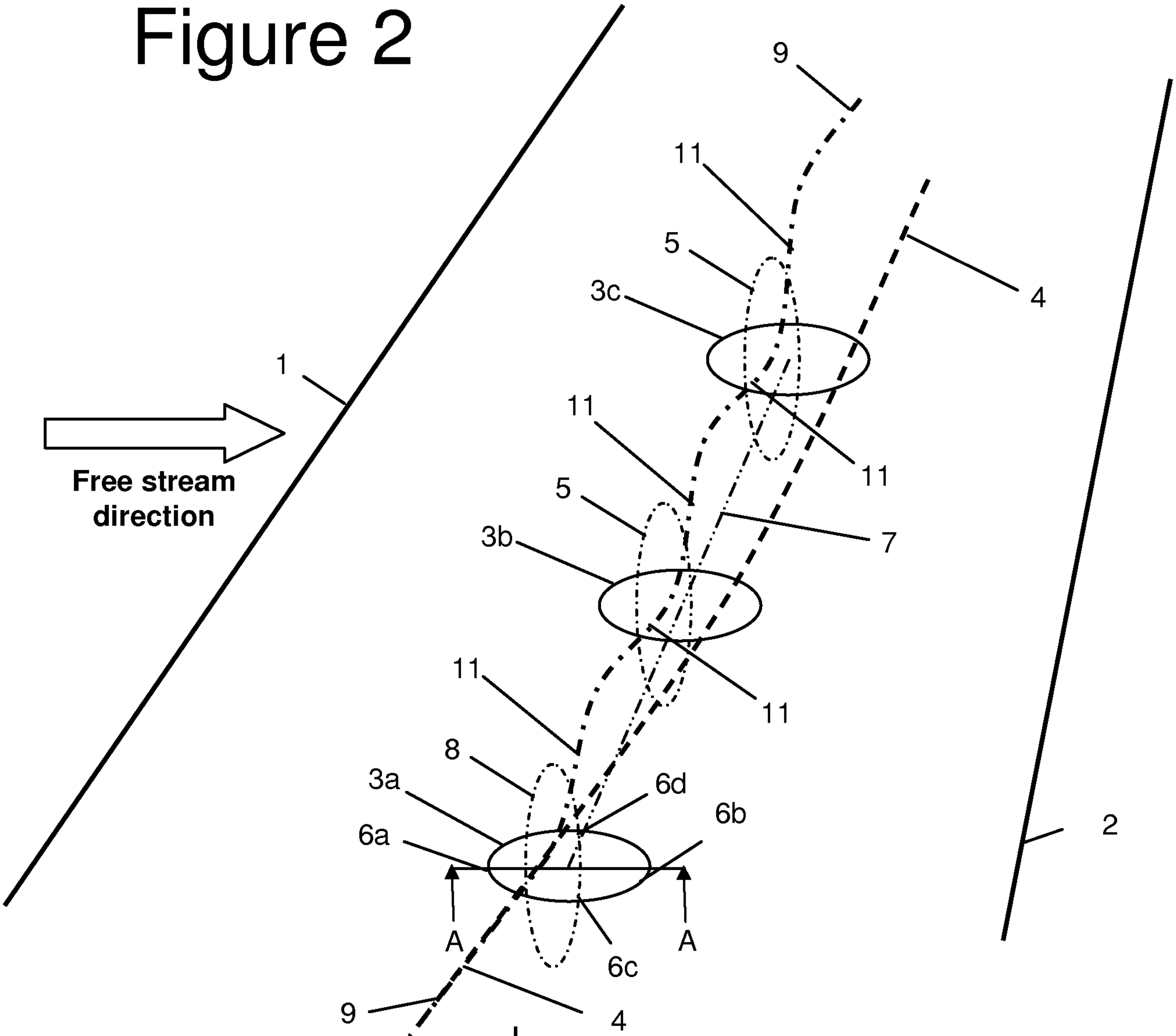


Figure 2



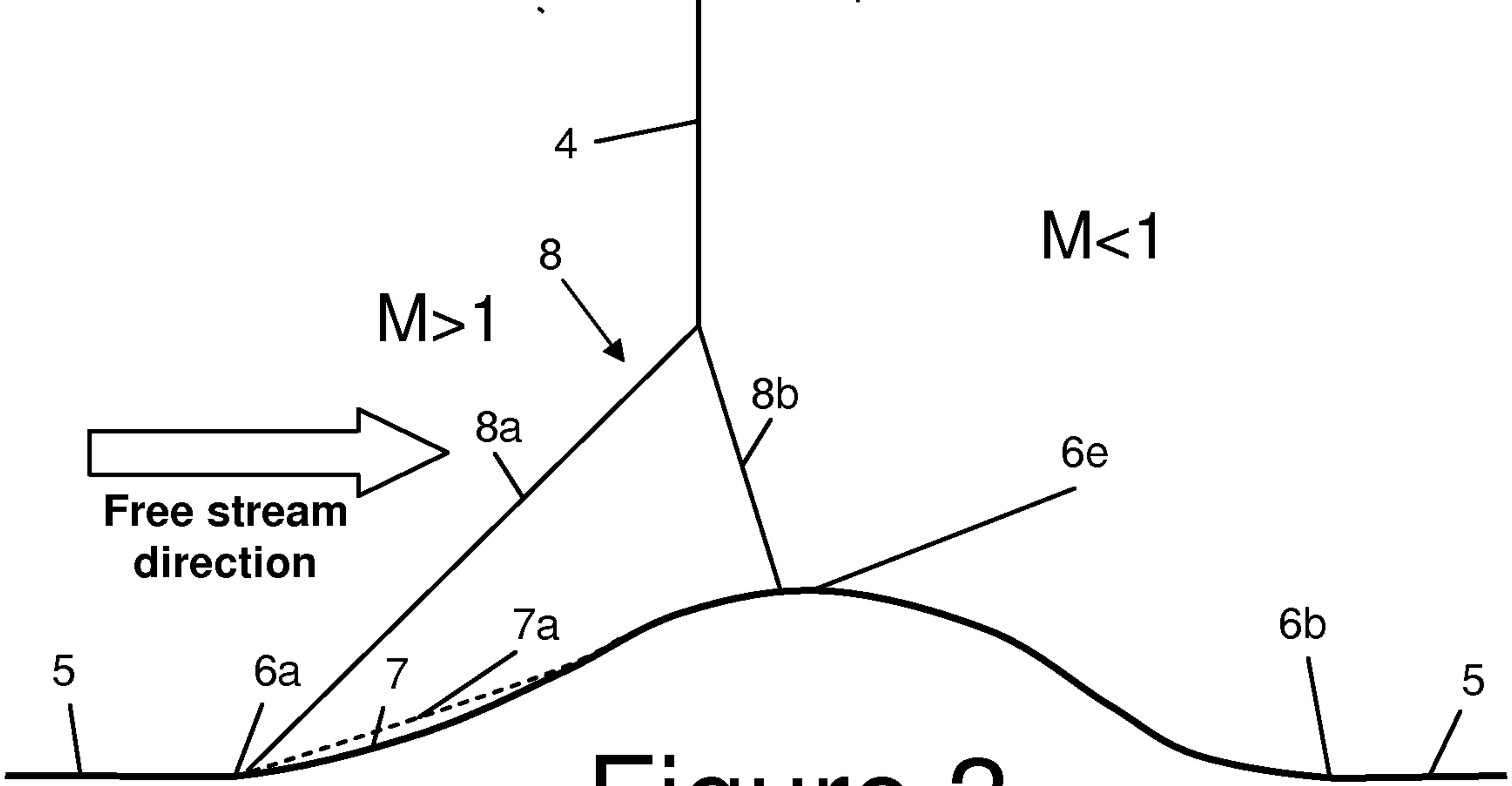


Figure 3

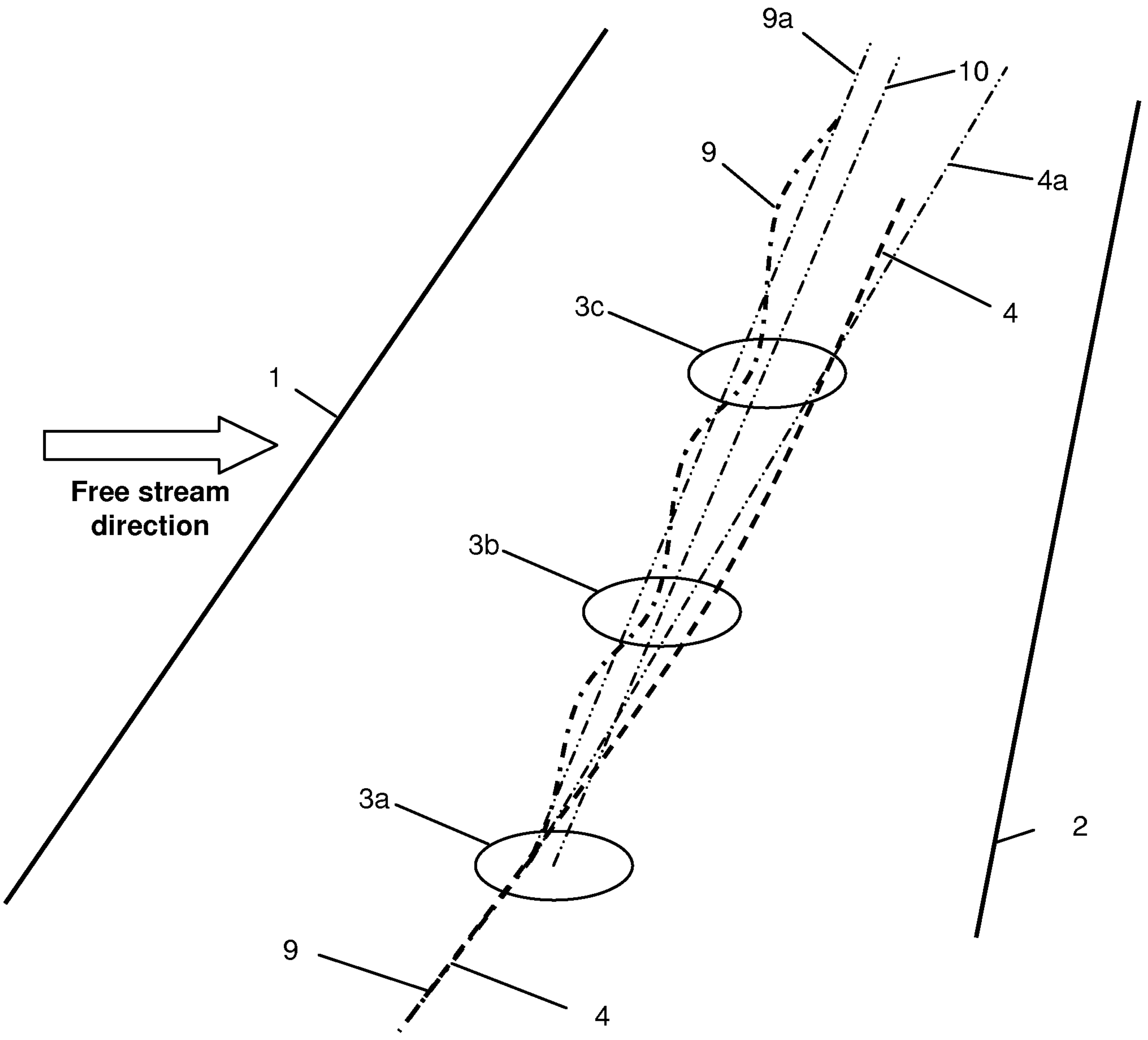
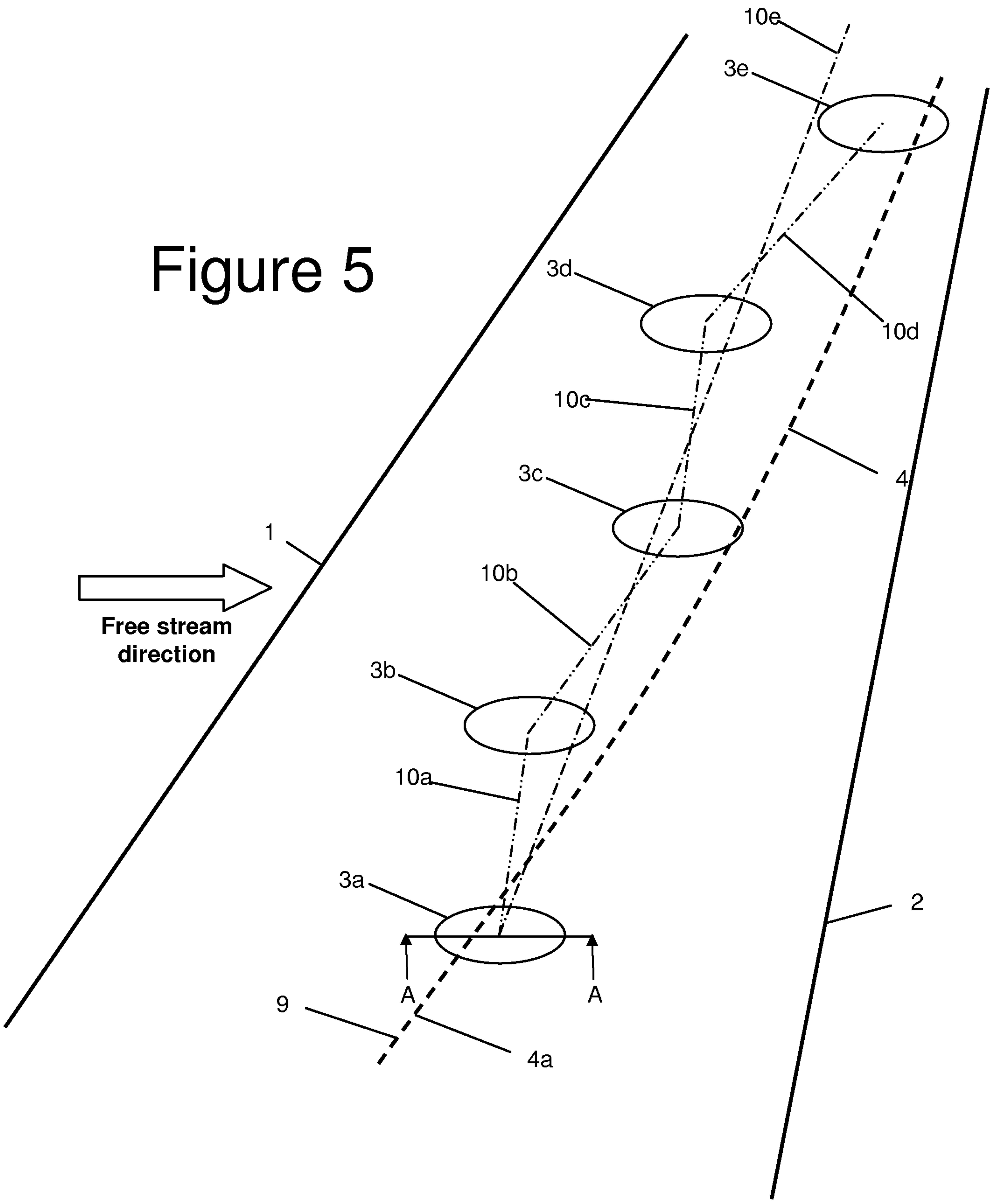
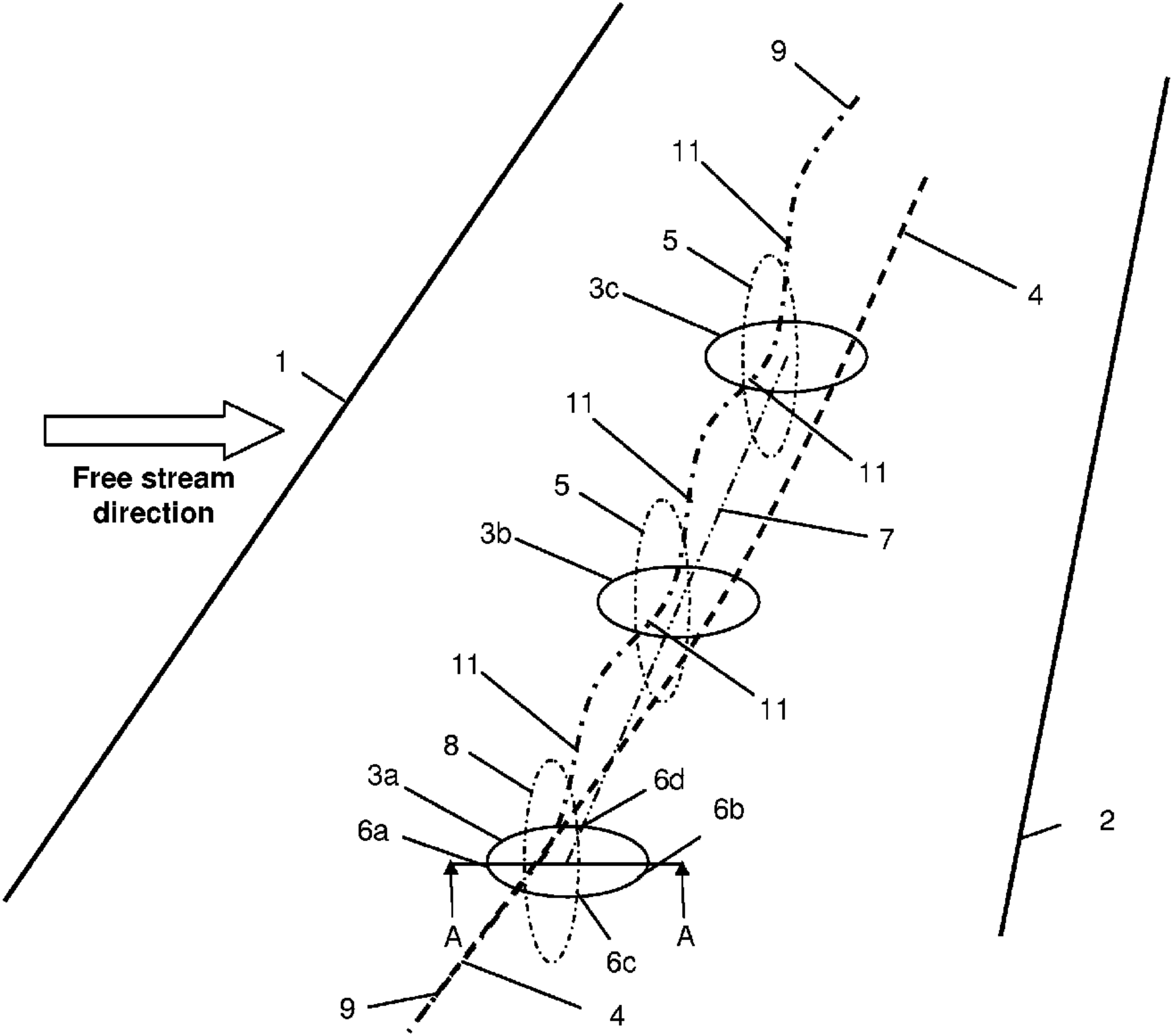


Figure 4

Figure 5


10e
3e
3d
10d
10c
4
3c
1
10b
Free stream direction
3b
10a
3a
2
A
A
9
4a


9
11
5
4
3c
1
Free stream direction
11
11
5
7
3b
11
11
8
6d
3a
6b
2
6a
A
A
6c
9
4